Guidelines for the Development of a Critical Composite Maintenance and Repair Issues Awareness Course

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16. Abstract

This report documents the results of a Federal Aviation Administration Cooperative Agreement with Edmonds Community College to develop a standard for teaching an introductory course on critical composite maintenance and repair issues. The course will serve as an introduction and provide an awareness of safety issues regarding the maintenance and repair of composite materials used in aerospace. The course will also provide a background for further study for those interested in becoming qualified practitioners. This course is intended for engineers, technicians, inspectors, and other personnel involved with the maintenance and repair of composite structures.

The framework for the awareness course is defined by Terminal Course Objectives, which are summarized further into terminal course modules. The four main areas at the highest outline level include base knowledge, teamwork and disposition, damage detection and characterization, and repair processes. Content includes text, laboratory instructions, and videos in support of the course objectives. Materials for evaluating the effectiveness of the course content in meeting the standards represented in this report are provided with reference to industry documents, especially CACRC AIR report 5719, which provides a checklist of detailed teaching points. An instructor's guide is also included to assist in developing the course. Additionally, several collaborative workshops and other forums that were held during the development process, involving industry, academia, and government regulatory agencies are described.

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LIST OF ACRONYMS

3D	Three-dimensional
AC	Advisory Circular
AD	Airworthiness Directive
ADL	Allowable damage limits
AMC	Acceptable means of compliance
AMM	Aircraft Maintenance Manual
AOG	Aircraft on ground
ATA	Air Transportation Association of America
AWM	Airworthiness Manual
BAA	Bilateral Airworthiness Agreement
BASA	Bilateral Aviation Safety Agreement
BVID	Barely visible impact damage
CACRC	Commercial Aircraft Composite Repair Committee
CAR	Canadian Aviation Regulation
CATT	Computer-assisted tap test
CFR	Code of Federal Regulations
CFRP	Carbon fiber-reinforced plastic
CMM	Component Maintenance Manual
CS	Certification Specification
DER	Designated engineering representative
DVD	Double vacuum debulk
EASA	European Aviation Safety Agency
EC	European Commission
emf	Electromotive force
FAA	Federal Aviation Administration
FEP	Fluorinated ethylene propylene
GA	General aviation
GFRP	Glass fiber-reinforced plastic
IPA	Implementation Procedures for Airworthiness
ISO	International Organization for Standards
LSP	Lightning strike protection
MAUS	Mobile automated scanners
MIA	Mechanical impedance analysis
MIBK	Methyl isobutyl keytone
MOR	Maintenance Occurrence Report
MPD	Maintenance planning document
MRO	Maintenance and repair organization
MIL HDBK-17	Military Composite Material Handbook 17
MRB	Material Review Board
NDI	Nondestructive inspection
OEM	Original equipment manufacturer
P/E	Pulse-echo
PMB	Plastic media blast
PSE	Principal structural element

PVF	Polyvinyl fluoride
QC	Quality control
QMS	Quality Management System
rpm	Revolutions per minute
RTM	Resin transfer molding
SAE	Society of Automotive Engineers
SB	Service Bulletin
SCRIMP	Seemann Composite Resin Infusion Molding Process
SRM	Structural repair manual
TBD	To be determined
TCCA	Transport Canada Civil Aviation
TCO	Terminal course objective
T_g	Glass transition temperature
TRM	Training repair manual
TTU	Through transmission ultrasonic
UT	Ultrasonic
UV	Ultraviolet
VID	Visible impact damage

EXECUTIVE SUMMARY

This report documents the results of a Federal Aviation Administration Cooperative Agreement with Edmonds Community College to develop a standard for teaching an introductory course on critical composite maintenance and repair issues. This course will serve as an introduction and provide an awareness of safety issues regarding the maintenance and repair of composite materials used in aerospace. The course will provide a background for further study for those interested in becoming qualified practitioners. This course is intended for engineers, technicians, inspectors, and other personnel involved with the maintenance and repair of composite structures.

This report establishes a standard awareness course content, through industry, academia, and regulatory collaboration, concerning the critical safety issues of composite materials maintenance and repair in commercial aerospace. This report is organized according to the awareness course framework and focuses on the terminal course objectives (TCO), divided into the four critical areas: base knowledge, teamwork and disposition, damage detection and characterization, and repair processes. Base knowledge is a prerequisite to the other three categories that represent the awareness course.

Base knowledge content provides a background for the awareness course. Students must have a grasp of terminologies and basic composites' technology to fully appreciate and understand critical issues in maintenance and repair. Therefore, teaching organizations should develop assessment tools to determine whether the student levels of understanding are sufficient prior to advancement to the awareness course.

The teamwork and disposition section of the class is organized to highlight the importance of teamwork in the disposition of damaged composite structure before proceeding to maintenance and repair. Trained engineers, inspectors, and technicians will work as a team from the time of damage detection until an approved repair is properly executed. Approval of a new repair design and process for critical composite structures will require an interface with engineers that have the necessary skills and data to substantiate structural integrity of the repair per regulations. Students will also be exposed to the documentation used in composite maintenance and repair that includes acceptable field disposition procedures and the associated regulatory rules that must be followed. Finally, a laboratory is described that will provide hands-on experience for students while working as a team in damage assessment, disposition, and repair.

Curriculum discussing damage detection and characterization familiarizes students with the typical types of composite damage, characteristics of the damage, and appropriate inspection procedures. It integrates technical information and laboratory experience to ensure students understand what causes damage in composite materials and the inspection methods needed for detection and complete characterization. Students will be able to identify hidden features that need to be characterized for damage disposition and repair quality control, as well as the more obvious visual assessments. Source documents will be used to identify the design details and appropriate inspection procedures for damage or repair in a particular location of a structural component. Once damage is characterized, source documents will also be used for disposition and repair. Damage and repairs that are beyond the limits of source documents will require special approvals that substantiate that the repair meets structural requirements.

Students will be introduced to bonded and bolted composite repair processes. The importance of conducting a composite repair per established processes, as specified in repair documentation, will be emphasized. Repair technicians and inspectors must have training that qualifies them for the specific tasks they will need to perform. A combination of technical information and laboratory exercises that provide exposure to the steps involved in composite repair processes will be used to help build this awareness.

The principal course objective is to provide an industry standard for awareness of composite maintenance and repair safety issues that reflect the insights of a worldwide body of experts in the field. A complete and balanced standard was achieved by developing the course primarily through collaborative workshops and other forums involving industry, academia, and government regulatory agencies. Composites technology lacks standard practices, and thus, these practices vary among manufacturers and maintenance and repair organizations. This cooperative agreement promotes the development of a standard awareness of composite technology and its safety implications in industry.

The awareness course development has resulted in a course framework with safety messages, assessments for course developers, and teaching resources. Combined, these outcomes provide an industry standard that represents the consensus of experts from around the world. TCOs and teaching points are the result of an extensive consensus process by practitioners from throughout the composite aerospace industry and, as a result, are considered industry standards. Detailed teaching-point descriptions were developed jointly with the Commercial Aircraft Composite Repair Committee (CACRC) and documented in AIR Document 5719, which is accessible through the Society of Automotive Engineers (SAE). Course developers can use the teaching points in this CACRC report as a checklist. The instructor's guide was not developed through the same rigorous consensus process represented by other elements of the course and, therefore, is not considered an industry standard. The instructor's guide is intended to provide teaching resource content that enhances the TCOs; it is a tool that instructors and curriculum designers can use at their discretion for course development and teaching. It should be regarded as an alternative to other sources of information, as deemed appropriate.

Course design is at the discretion of the teaching organization using this document, which provides examples. The order and time requirements of material presentation are flexible, as well as the teaching format. One format alternative, which involves a blend of traditional classroom instruction and laboratory exercises, has the advantage of immediate hands-on practice of principles in the laboratory. However, students have to be on-site for the duration of the course.

Another format alternative is the use of distance learning for teaching principles coupled with attendance at 3-day regional laboratories. Specific advantages of distance learning include reduced travel costs, better access to the course from remote regions, and the ability to use subject matter experts in the discussion sections of the course, which would be difficult in many traditional classroom locations. The methodologies and advantages of distance learning satisfy a pressing need for out-reaching to global students in a cost-effective fashion. The combination of distance learning and regional laboratories is anticipated to attract many more students, fulfilling the need to educate critical composite maintenance and repair principles to a wide, global audience.

1. INTRODUCTION.

This report documents the results of a Federal Aviation Administration Cooperative Agreement with Edmonds Community College to develop a standard for teaching an introductory course on critical composite maintenance and repair issues. This course will serve as an introduction and provide an awareness of safety issues regarding the maintenance and repair of composite materials used in aerospace. The course will provide a background for further study for those interested in becoming qualified practitioners. This course is intended for engineers, technicians, inspectors, and other personnel involved with the maintenance and repair of composite structures.

1.1 REPORT ORGANIZATION.

Section 2 is an overview of the processes involved in the maintenance of composites and provides a baseline for all subsequent discussions.

Section 3 summarizes and focuses on the critical safety issues that influence the course content.

Section 4 provides a list of terminal course modules and objectives that outline the background information given in sections 2 and 3. Terminal course objectives (TCO) provide the consistency of training among organizations and individual course instructors. Assessment of student learning is directly related to the objectives described in section 4. TCOs represent the consensus of course expectations from industry experts.

Section 5 provides an example of a course structure and design, which can be modified at the discretion of curriculum designers. Specific reference is made to a publication by the Society of Automotive Engineers (SAE) (AIR Document 5719), which is an industry publication describing detailed teaching points within the course objective structure in this report.

Section 6 contains proposed content that a course developer can use to populate the course objectives. Use of this content is optional and not considered part of the course standard, but should be a valuable resource to the instructor or the developer.

A detailed description of activities and outputs resulting from the many workshops that are the basis of this report is described in appendix A.

1.2 INDUSTRY STANDARD FOR A COURSE IN COMPOSITES MAINTENANCE.

As a relatively new technology, composite material development and practice continues to evolve. The absence of public domain standards is reflected in a lack of education standards, resulting in institutions and training organizations frequently focusing on specific areas of composite materials technology. This sacrifices balance and breadth, and usually omits valuable details from course curricula.

This lack of standards in composite technology exists for many reasons. As the percentage of composite materials dramatically increases in new aircraft development, key primary structures

are being designed using composites such as wing spars in a large aircraft wing for the Airbus A400M and wings and fuselage of the Boeing 787.

In commercial aerospace, the intense competition among airframe manufacturers, notably Airbus and Boeing, has restricted information flow. Composite technologies are evolving in real time, with next-generation design and manufacturing technologies for large composite structures being researched and developed. Coupled with property developments, manufacturing techniques are also changing. Evolving composite technologies are frequently considered proprietary and are not available in the public domain. As a result, composite property standards are in the early stages of development and are in flux, requiring special skills and awareness of safety implications of composite maintenance and repair.

Elements of the curriculum were developed through an extensive consensus process among industry, academia, and regulatory organizations. This process, described in appendix A, is one way an industry standard is developed. Key elements of this process include:

- Curriculum development industry standards
- TCOs and modules
- Teaching points
- Curriculum development aspects that are not considered industry standards
- Instructor's guide
- Storyboards
- Course evaluation using AIR 5719 teaching point industry standards

Teaching points can be used as a checklist to (1) allow a training organization to evaluate course effectiveness, (2) ensure course completeness, and (3) balance with the course standard. The inclusion of teaching points in this curriculum development process is intended to provide the benefit of consistency among different training institutions. This report contains TCOs and modules. AIR 5719 provides a detailed list of teaching points for this purpose.

The teaching points were developed with the involvement of industry, academia, and regulators, and are, therefore, considered to represent industry standards and expectations for fulfilling the intent of this awareness course. However, the instructor's guide is a tool for educators, but was not developed through the same consensus process as the teaching points, and does not qualify as an industry standard.

2. OVERVIEW.

This section represents an overview of important safety issues associated with commercial aircraft composite materials maintenance and repair.

2.1 PRODUCT DEVELOPMENT AND SUBSTANTIATION.

The development and substantiation of composite aircraft structure requires close coordination between design, manufacturing, maintenance, and operations personnel. Tasks performed during type design and production certification are used to integrate structural and manufacturing

details, yielding quality control (QC) processes that ensure a repeatable product with reliable performance. In addition, manufacturing defects, environmental exposure, service damage, and inspection and repair procedures need to be considered in the structural substantiation to support subsequent product production and service.

Product substantiation usually involves a combination of tests and analyses at different scales of study, ranging from coupons to structural details to full-scale aircraft structure. A building-block approach is often used for this substantiation. Such an approach, which is not unique to composites, relies on repetitive testing at lower scales to capture material and process variability and large-scale tests to validate load paths and the structural design. Depending on product performance goals (e.g., weight savings) and specific design and manufacturing details, the amount of building-block analyses and tests used for a particular structure may differ from program to program. However, structural tests are generally needed to supplement analyses in substantiating static strength and damage tolerance, particularly when considering the unique composite issues for new designs and processes.

The process of type certification implies the aircraft design meets the airworthiness standards prescribed in Title 14 Code of Federal Regulations (CFR). Manufacturing aircraft via a facility holding a valid production certificate implies the aircraft can be fabricated as designed. The aircraft is considered airworthy and in safe operating condition. Thus, a standard certificate of airworthiness can be issued. At this stage, the term "initial airworthiness" is often used to indicate the status of aircraft.

2.2 CONTINUED AIRWORTHINESS AND OPERATIONAL SAFETY.

The term "continued airworthiness" is often used to monitor the safety of the aircraft when it enters service. There are a number of factors affecting the continued airworthiness of composite structure. Unlike metal structures, where fatigue cracking can be a primary threat to structural integrity, accidental damage (e.g., foreign object impact) is a critical threat for composites. Hidden deficiencies incurred in manufacturing also need to be considered. For example, weak bonds caused from surface contamination may not be detected by initial inspection methods. As a result, other QC procedures and redundant design features are needed to ensure the continued airworthiness of bonded structures.

The different levels of degradation and damage that can occur during service must be considered for structural substantiation of static strength, flutter, and damage tolerance. This can start with an evaluation of environmental effects and fluid compatibility for the particular composite material. Matrix-dominated composite properties, such as compressive strength, are most sensitive to moisture absorption over time and high temperatures. Static strength substantiation includes the smaller damages that will not be detected in production or maintenance inspection, while damage tolerance addresses larger damages that need to be repaired once discovered. In addition to meeting the static strength and damage tolerance requirements, sufficient data development is needed to facilitate timely engineering support for airworthiness assessments and repair of manufacturing flaws or damage found during production and service.

Repairs and continued airworthiness procedures must be provided in service documents, including approved sections of the maintenance and instruction manuals for continued

airworthiness. The resulting repairs and maintenance procedures must be shown to provide structure that continually meets the processing standards and structural performance substantiated during type certification. These procedures include requirements and substantiation for material acceptance, repair fabrication, QC, environmental resistance, lightning protection, static strength, damage tolerance or fatigue and stiffness, and weight and balance.

Bonded repairs performed in the field require special care due to added complexity and lack of controlled fabrication facilities. Reliable procedures are needed to ensure sufficient cleanliness and environmental controls for proper bonding. Some special considerations (e.g., drying of core materials subjected to water ingression) also need to be taken in the field. Limits on the size of bonded repairs performed in the field relates to the same needs for structural redundancy as established in the design of bonded structure. All repairs should have supporting data based on tests or analyses substantiated by test evidence.

Documentation identifying all critical inspection items should be gathered to support maintenance. For example, control surface gaps between fixed and movable surfaces should be identified. Information on the weight and balance of control surfaces should also be documented. Maintenance instructions should include material and process controls, fabrication steps, cured-part tolerances, nondestructive inspection (NDI), and other QC checks for bonded repair.

2.3 OTHER ESSENTIAL ELEMENTS.

There are other elements that have an impact on composite product certification and continued airworthiness management. Lack of engineering standards for composites can affect the associated costs and timelines. This effect can be minimized, depending on skills of the engineering team. The mentoring and training of new engineers and technicians is also essential to a successful aircraft service use.

A good balance of team members with engineering experience in composite design, analysis, manufacturing, and maintenance practice is needed to coordinate a product development and certification program. The tasks performed by each discipline must be coordinated to avoid adding costs and risks in meeting schedule milestones. For example, integrated design and manufacturing process definition with sufficient QC is needed to ensure the proper conformity of structural and flight test articles. The engineering team should also have specialists on the specific composite material and processes that are used to troubleshoot problems.

Good communication must exist between the engineering disciplines involved in the continued airworthiness management of composite products in service. Maintenance and operations personnel should have knowledge of factors affecting the performance of a composite structure. This is important when working with structures engineers on the disposition of anomalous events (e.g., structural overloads and ground vehicle collisions) and damage found in service.

2.4 TECHNICAL ISSUES.

Technical issues associated with the maintenance (repair) of composite materials used in aircraft products start with a realization of how to gain knowledge and acquire skills for safe industry practices. Since the technology has not been standardized, textbooks and reports documenting the working knowledge needed to be proficient in the field do not exist. Experience must be gained from working in the industry and using methods and procedures that are often proprietary for a given product. Some references help to clarify the critical technical issues, to put the lack of standard industry practices into context, and to illustrate the additional training necessary for the maintenance (repair) of composite structures on a given aircraft.

In the discipline of composite materials, it is important to realize that the material structural properties are set during the fabrication processes of parts or repairs. This differs from metals where in most part and repair fabrication processes do not alter the base material properties of the raw material form (exceptions include heat treatments, welding, and some forming processes). Special skills are needed for composite engineers, inspectors, and technicians involved in production and maintenance. These skills depend on the specific details of a given structural design, processing specification, QC procedure, tooling, inspection method, and repair. If any of these details are not followed properly, the database and analyses used for structural substantiation may not be representative of the fabricated part or repair.

2.5 TEAMWORK AND DISPOSITION.

Teamwork is essential to composite maintenance, particularly as associated with the steps involved in aircraft structural inspection, disposition, and repair. Team members should have some awareness of the different skills needed to successfully perform each step. This awareness serves to understand personal skill limits better and to recognize the appropriate personnel when assistance is needed. Approved documentation for a particular aircraft structure should have the necessary supporting databases and sufficient details to guide a team in the field through the steps of inspection, disposition, and repair.

Whenever damage found in service is beyond that included in approved documents, inspection and repair will require special instructions. Such cases involve other team members with the skills needed to determine the full extent of damage and to develop a repair that meets the airworthiness requirements that were originally posed on the base structure. In the case of primary structures, the data and analyses needed to substantiate that the repair meets such requirements may be extensive (e.g., stiffness, strength, fatigue, and damage tolerance for critical load cases). Team members not versed in making such engineering judgments for the specific structure in question should never assume that the inspection and repair is similar to previous experiences on other structures.

2.6 ROLES AND RESPONSIBILITIES.

Each team member involved in damage disposition, inspection, and repair should have the training needed to complete their tasks. Three general types of individuals are needed: engineers, inspectors, and repair technicians. Team members used for engineering tasks require a minimum of a Bachelor of Science engineering degree, or equivalent, from an accredited

academic institution, including some training in aircraft composite structural design and analysis. The latter training is best received while gaining industry experience. Engineers also need a good understanding of the regulatory documents and procedures to follow, if damage and repairs are beyond those given in source documentation. Inspectors must have training in the use of a variety of inspection techniques for composites. In addition, inspectors must have good eyesight and hearing. Repair technicians must have training in composite repair processing, including the use of the associated tooling and equipment. In addition, technicians generally need good hand-eye coordination.

In addition to training for the general skills used in composite damage disposition, inspection, and repair, the personnel (engineers, inspectors, and technicians) working on a given structural component need to have detailed knowledge of the particular part. This includes an understanding of approved source documentation such as the original equipment manufacturer (OEM) specifications, drawings, inspection procedures, and repair processes. Some of this information may be referenced or may exist in the OEM structural repair manual (SRM), or its equivalent. In addition to an understanding of the source documentation, the team must have numerous technical skills for the damage disposition, inspection, and repair of a particular structure. Sometimes such skills may only be attained in training provided directly by the OEM.

A complete disposition process is needed after composite damage is first discovered. Operations personnel become the first line of defense in reporting known service events that may have damaged the aircraft. They may also discover clearly visual damage while servicing the aircraft or in-frequent, walk-around inspections. Operations personnel must report possible damage to maintenance personnel before the appropriate inspection and disposition is initiated. Maintenance personnel must understand limits of inspection methods used to disposition the composite damage. Organizations that perform heavy composite maintenance inspections while aircraft are grounded for a significant period of time usually have an understanding of these methods. However, it is likely that possible damage discovered in the field will be reported to field maintenance personnel who lack the special skills needed for a more thorough composite inspection. Since damage to composite structure often includes characteristics that are not visibly obvious (e.g., delamination and debonded elements), operations and maintenance personnel in the field must be informed that special skills are needed for inspection and disposition. The right people must be contacted to avoid putting an aircraft with severely damaged composite structure back in service.

Operations personnel are critical to protecting the safety of aircraft exposed to anomalous service events that may lead to damage not considered in design. Their important role is limited to the realization that such an event may have damaged the aircraft and therefore needs to be reported for maintenance action. Some anomalous service events that need to be reported include: (1) high-energy impacts due to service vehicle collisions, (2) flight excursions outside the design envelope (3) severe landing loads, (4) other abnormal flight, landing, or ground events outside the scope of that substantiated during type design certification, and (5) heavy rain or hail damage. Damage caused in such anomalous events is a valid safety threat for all types of aircraft structures. A particular concern for composite structures comes from the nature of damage, which may not include obvious evidence of severe loading or material distress (e.g., dents or other permanent distortions that indicate metal yielding). Safety management principles for

communication between the type certificate holder, owner, maintenance organization, and operations personnel is paramount to mitigating risks. Any damage suspected due to an anomalous ground, flight, or landing event must be reported to maintenance engineers and inspectors to help determine the severity of damage. This should be the case whether or not exterior damage is visibly detectable.

Composite damage that can be reliably detected using visual inspection methods will usually require additional NDI methods to determine the full extent of damage. Composite inspectors must be trained to recognize that small visual dents and other indications of surface failures often come with additional hidden damage that requires NDI for disposition. Source documentation must be reviewed to determine the recommended NDI method for given structural details. The different composite damage types and NDI methods that can be used to fully characterize the extent of damage will be reviewed later in this section.

Once the damage is completely characterized, engineering personnel will need to consult source documentation to determine if it is within allowable damage limits (ADL). Component records should also be reviewed for previous repairs and existing ADL in the proximity of damage. These may be restrictions as to the proximity between composite repairs and ADL. When damage is within ADL for a given location, some maintenance action may still be needed to replace protective surface layers and seal the damage. Damage beyond the ADL must be repaired. Source documentation also provides limits on what can be repaired in the field without further consultation with the OEM. Use of repair limits highlights the importance of an accurate damage disposition process.

The design details, material types, tooling and process instructions used in SRM should be clearly outlined in source documentation. These steps must be closely followed by maintenance (repair) technicians. Any deviations require approval. Options for both permanent and temporary repairs are often available for composite structures. The lack of environmental resistance and long-term durability for most temporary composite repairs dictates that permanent repairs be added before exceeding time limits. Temporary composite repairs in the field may translate to the long down time for the aircraft while permanent repairs, such as those involving laminate patch curing and bonding, are made.

Depending on the particular composite part and damage type, a bonded or bolted repair may be selected. The removal of damage and surface preparation for each type of repair is different. Bolted repairs to composite structure are similar to those used for metals, but there are unique design and process details that need to be considered. For example, hole drilling in composite laminates is distinctly different. Unique fastener installation processes may also be needed for composite bolted repairs to gain the necessary lightning strike protection. Bonded composite repairs contain a series of process steps that must be carefully performed by well-trained technicians. In-process controls are essential to ensure adequate bonding is achieved. Postprocess NDI of bonded repairs provides useful information on the quality of the bonded joint but does not ensure that the full bond strength is achieved. Cleanliness and environmental controls are essential for bonded composite repairs, bringing unique challenges when the part cannot be removed from the airplane. Drying of the part is also an important step before bonding a repair.

2.7 INFORMATION CONTAINED IN DOCUMENTATION.

As is the case for original part fabrication and assembly, all materials and processes used to repair composite structural aircraft components must meet approved specifications. Additional instructions for damage disposition, repair design, and processing, which are documented to control maintenance inspection and repair in the field, must also be approved. Approval is gained through sufficient research, processing trials, testing, and analysis to demonstrate that inspection, damage disposition, and repair can be reliably controlled. As a result, all structural repairs in approved documents have been substantiated to meet the regulatory requirements.

The need for approved documentation in composite repair is an important safety issue. Since composites have not reached the same level of standardization as metal materials, there are very few standard material and process specifications available in the field. Even those that exist are limited because the repair substantiation database and analysis may relate to different materials and specifications preferred by the OEM. As a result, far more constraints exist for composite repair than exist for metals. These constraints directly limit maintenance organizations from defining their own preferred materials and suppliers. In general, maintenance organizations will not be able to purchase materials and perform composite repair in a mode similar to the procedures they use for metals. As a result, composite repair costs are generally higher.

The science of reverse engineering for composite structures is not mature. In addition, repairs used for similar structures on other models of aircraft should never be assumed to be acceptable for a particular structure. The specific structure in question will likely have loads and structural strength margins that are different despite similar appearance. In addition, the damage in question may be beyond that in which data and analysis for repair substantiation exists. Composite design relies heavily on test data correlated with engineering analyses of structural load paths for a given structure. Generic theoretical approaches to strength and damage tolerance analyses have been found to be very limited in applications with real structure. Much of this relates to the complexity of damage and design details that must be included in the analysis of composite structures, because they have different properties in different directions and are quasi-brittle materials. Such a class of materials must account for all possible stress concentrations to limit working stresses to a safe level.

Composite materials generally are not interchangeable and very few material forms are sold as commodities. This is different than the case for metal alloys, which have multiple suppliers for the same material. Even though two different composite materials may appear to have similar properties (or the one desired for use appears to be better), there will often be subtle differences in structural properties (e.g., notched strength and damage tolerance), which are not routinely part of base material databases. Different composite materials will also have different chemical compatibilities with the base composite part and adhesive selected for a bonded repair.

Current trends in the composite industry indicate that international standards for material and process specifications will eventually be established. Major OEM are supporting this for the routine, noncritical repairs used for some structures. In general, shared data should be viewed with caution because the transition to such a state will likely take a long time for the many materials and processes used for composite repair. One of the biggest hurdles relates to the data and analyses needed for structural repair substantiation. The current available data in this area is

part-specific and empirical. Therefore, the information is limited to design details (i.e., specific geometries, loads, and criteria applied for a given part). As a result, expanded shared databases are also needed to allow more freedom in composite repair. To date, most shared databases have been limited to the base properties used in controlling materials and test sampling to show that new users of the material understand the specifications. As shared databases expand to include structural properties critical to repair design and analysis, the users may be able to design more complex repairs.

There are many source documents that contain maintenance, modification, rework, and repair information. The SRM, or equivalent, is often the most complete maintenance document in terms of instructions for damage disposition, inspection, and repair. A SRM typically contains previously approved data but this should always be confirmed. Service Bulletins (SB) issued by an OEM are the means for sharing modifications to previous maintenance instructions. These include supplemental inspection, rework, and repair instructions for a given composite part. Many composite SBs used to date relate to composite manufacturing problems or service degradation in the field. The latter case can often be traced to a bad composite design or process detail, which has less than adequate environmental or aviation fluid resistance. There are also some cases where unanticipated secondary loads cause part delamination or debonding in the field. Service newsletters are issued by OEM to address troublesome components or systems. These help make owners, operators, and maintenance groups aware of potential problems. Other documents available in the field include maintenance planning data, aircraft maintenance manuals, and component maintenance manuals. All of these typically contain specific inspection requirements and overhaul instructions.

Whenever an approved repair design and process is not available for a given damage, the maintenance engineer has several options. First, the OEM may be contacted for help in defining an approved repair. Although time consuming, this is often the most efficient and safe option available to the maintenance organization because the OEM has key information for the solution (databases, substantiated analyses, loads, design details, and repair process procedures). Another option is to develop a new repair design and process and gain the necessary substantiation for regulatory approval. Without some help from the OEM, this is a very difficult task for composite primary structures, where internal loads and structural details directly relate to the surrounding structures and their design details. The final option is to replace the part with an approved part that meets the regulatory requirements.

2.8 DAMAGE DETECTION AND CHARACTERIZATION.

Composite defects or damage come from manufacturing or service exposure. Defects and handling damages that occur during manufacturing are controlled by in-process and postprocess QCs. Despite stringent controls, some defects and damage are likely to occur, requiring factory disposition. Most processing anomalies that are allowed to enter the field are much smaller than damage considered from service. This relates to advanced NDI procedures used in the factory. Factory NDI methods are more stringent than those that can be practically applied in the field. Hence, design criteria must account for larger field damage to accommodate practical maintenance practices.

One type of manufacturing defect that has posed field problems relates to weak bonds, where bond surface contamination, tooling, or curing problems lead to insufficient bond strength. This manufacturing defect is best controlled in-process because factory NDI performed after cure typically will not detect the problem. Composite design criteria have protected against this problem by making sure there is redundant design detail and damage tolerance to ensure the associated debonding can be found in service.

There are many types of composite defects and damage that can arise in the field. Field damages can result from (1) dropped tools, (2) service vehicle, jetway, or work-stand collisions, (3) aircraft-handling accidents, (4) dropped parts, (5) improperly installed fasteners, (6) bird strikes, (7) foreign object impacts (e.g., runway debris), (8) overheating, (9) fluid contamination, (10) flight overloads, and (11) sonic fatigue. Note that many of these damages are due to foreign object impact, which is one of the primary safety issues for composite structures. Repeated loads, by themselves, typically do not lead to service damage because of relatively flat composite fatigue curves and a need to account for accidental damage in design criteria, which reduces the working strain levels. Exceptions where fatigue has been a problem usually relates to bad design detail, where secondary out-plane loads occur in service, damaging the weak direction of the composite.

Other field damages are due to environmental conditions, including (1) hail, (2) lightning, (3) ultraviolet (UV) radiation, (4) high-intensity radiated fields, (5) rain erosion, (6) moisture ingression, and (7) ground-air-ground cycles (temperature, pressure, and moisture excursions).

2.8.1 Damage Types and Sources.

There are a number of damage types that significantly reduce the residual strength of composites. The drops in residual strength are related to damage type and size. In the case of impact damage, compression, shear and tensile strength can all be reduced. High energy and large impactor diameters are needed to produce the greatest drops in residual strength. There are also some damage types that have very little effect on residual strength but, depending on the design detail, some of these may combine with environmental effects and ground-air-ground cycling to cause further damage. The following sections will cover the different composite damage types and their sources.

2.8.1.1 Matrix Imperfections (Cracks, Porosity, Blisters, etc.).

This type of defect usually is not a concern for most applications. Matrix cracking occurs parallel to fibers due to thermal and mechanical loading. Isolated matrix cracks can occur in processing as local fiber and matrix volumes change with part geometry. Matrix cracks have little effect on residual stiffness and strength but are not a design driver when localized. Most composites used in aircraft applications do not matrix crack over wide areas at working strain levels. Due to high-thermal residual stresses, aramid/epoxy materials are prone to matrix cracking over wide areas in an aircraft environment. Aramid/epoxy composite facesheets and sandwich construction have had problems because moisture or aviation fluids ingress through the cracks and into the sandwich core due to ground-air-ground cycling. Fluid ingression can further degrade the sandwich facesheet to core bond. Porosity is usually controlled in the factory below levels that significantly reduce structural properties.

2.8.1.2 Delaminations and Debonds.

This form of composite damage occurs at the interface between the layers in the laminate, along the bondline between two elements, and between facesheets and the core of sandwich structures. Delaminations can form due to stress concentrations at laminate-free edges, matrix cracks, or structural details (e.g., radii and ply drops). Delaminations may also form from poor processing or from low-energy impact. Debonds may also form similarly. Since delaminations and debonds break the laminate into multiple sublaminates and reduce the effective stiffness of bonded structural assemblies, they reduce structural stability and strength, posing a safety threat.

2.8.1.3 Fiber Breakage.

Broken fibers can be critical because composite structures are typically designed to be fiber dominant (i.e., fibers carry most of the loads). Fortunately, fiber failure is typically limited to the zone of impact contact and is constrained by the impact-object size and energy. The resulting loss in residual strength is controlled by a relatively small damage size. One exception can be a high-energy, blunt impact over a large area, which breaks internal structural elements such as stiffeners, ribs, or spars, but leaves the exterior composite laminate skin relatively intact.

2.8.1.4 Cracks.

These types of damage are defined as a fracture of the laminate through the entire thickness (or a portion of the thickness) and involve both fiber breakage and matrix damage. Cracks typically are caused by impact events, but can be the result of excessive local loads (either in the panel acreage or at a fastener hole). In most cases, cracks can be thought of as a more general category than fiber breakage because matrix cracking is also included. As a result, the same thoughts shared for fiber breakage apply.

2.8.1.5 Nicks, Scratches, and Gouges.

These types of damage are not critical if the damage is limited to the outer layer of resin without any damage to the fibers. If the fibers are damaged, they must be treated as a crack in the affected plies. Unlike metals, composite matrix nicks, scratches, and gouges are not likely to grow under repeated loads.

2.8.1.6 Dents.

Dents are typically caused by an impact event. The dent is usually an indication of underlying damage. Damage can consist of one or more of the following: sandwich core damage, facesheet delaminations, matrix cracks, fiber breakage, and debonds between facesheets and core. Dents in thin facesheets of sandwich core often only involve core damage. Dents in solid laminate areas fastened to a substructure (e.g., edge-band areas of sandwich panels) can have associated damage to the substructure.

2.8.1.7 Puncture.

A puncture is defined as an impact damage that causes a penetration of the facesheet or laminate. A puncture is more likely to occur than a dent if the impact energy is higher and the impactor is of small diameter. Also, thin facesheets of sandwich structure are puncture-susceptible. The edge of the puncture may be relatively clean, or may be ragged, depending on the type and energy of the impact event. In either case, there may be associated delaminations, matrix damage, and fiber breakage outside of the puncture edges.

2.8.1.8 Damaged Fastener Holes.

Composite damage can occur due to improper hole drilling and poor fastener installation. Fastener hole elongation damage can occur due to repeated load cycling in service. Damage to fastener holes can also happen during maintenance while removing or replacing screws or quick-release fasteners. When this type of damage is localized to one or two holes, it will have limited effect in multifastener joints.

2.8.1.9 Erosion.

Erosion can occur at the edge of a laminate panel or at a sandwich edge band as a result of airflow over the structure or the impingement of debris, rain, etc. Erosion can expose surface fibers to reduce local strength and lead to moisture ingression. In most cases, erosion is not a safety threat because damage is found before becoming serious.

2.8.1.10 Heat Damage.

This type of damage is possible near sources of high temperature (e.g., engines, air-conditioning units, or other systems). There are usually visual indications of heat damage caused by exhaust or charring of the part surface, but it may be difficult to determine the extent of heat damage.

2.8.1.11 Lightning Strike Damage.

This type of damage is usually constrained to surface layers of the skin panel. Degradation to the lightning protection system could pose potential for greater damage threats. Rare, high-energy lightning strikes may also cause considerable damage. Lightning damage to the composite is usually easily detected visually.

2.8.1.12 Combinations of Damages.

Dents or penetrations may include matrix cracks, fiber breaks, delaminations, and element debonding. This combination of damage usually requires foreign object impact. The size and detectability of combined damage determine the safety threat. Small damages that are covered by composite design criteria (e.g., barely visible impact damage) are usually not a safety threat because the structure is substantiated with such damage present for the life of the aircraft. Larger damages may remain hidden depending on the nature of the impact event. Penetrations by turbine blades and rotor disks are known events causing damage. This damage is repaired once the aircraft lands. Significant bird strike also leads to visibly obvious damage that is fixed

shortly after the event. Very high energy levels and blunt impacts are needed to create large hidden damages. This is one reason that communications with operations groups are so critical to safety. Known service vehicle collisions must be reported to engineers and inspectors capable of determining the extent of damage.

2.8.1.13 Damage From Fluid Ingression Into Sandwich Panels.

This type of damage usually requires another damage to be present, allowing a leak path into the sandwich core. Some design details (e.g., porous fabric weave styles used for facesheets, square-edge sandwich close-outs) may also allow fluids to enter the core through leaks. Once the fluid gets into the sandwich part, it can degrade the core or its bond with the facesheets. Damage growth can be caused by freeze-thaw cycles, a pressure differential through the facesheet and fluid degradation of the bond.

2.8.2 Damage and Repair Inspection Procedures.

Methods used in the field for composite part damage detection, damage characterization, and postrepair inspection are typically less sophisticated than those employed by the OEM for their postprocessing inspection. Operators and maintenance organizations use visual inspection as their main technique for initial detection of field damages, unless NDI techniques are specified by the specific maintenance planning manual or aircraft maintenance manual. Once damage is detected visually, other NDI methods are needed to map the full extent of damage for proper disposition. As discussed previously, visual detection methods are possible, assuming a composite structure was designed to carry loads with nonvisible damages occurring in service. The exception relates to anomalous events that occur with knowledge to pilots or operations personnel. Since all such events are difficult to cover through design criteria, safety is managed by reporting the event to maintenance personnel that are trained to inspect a part for underlying damage.

Despite the use of visual inspection to first detect damage, NDI methods are essential to the subsequent damage disposition and repair processes. Many of the damage types described in section 4.2.1 have both visual and hidden damages. Hidden damage in composites usually covers a larger area than visual indications of damage and dominates the lost residual strength. It is essential that the proper NDI methods are applied to damage found on composite structure to map the full extent of the damage, which is needed to determine whether damage is below the ADL or whether repairs are required. Since a disposition of repair size limits also depends on accurate mapping, decisions on whether the repair substantiation database is sufficient also relies on a complete inspection with the proper NDI. Typical NDI methods used for composites are described in sections 2.8.2.1 through 2.8.2.7.

2.8.2.1 Tap Testing.

There are many different tap-testing devices ranging from a simple coin tap, where the human ear is used to audibly sense damaged structure, to automated methods that make a recording of changes in the sound. Tap testing has been used for damage inspection of composite and metal-bond components. In general, the tap test works well for inspection of damages in thin skins of any type. The method is especially useful on sandwich structure with thin facesheets and

honeycomb core. It can work on solid composite laminate structure if the first few plies are delaminated, but it cannot reliably detect defects or anomalies deeper in the laminate.

2.8.2.2 Ultrasonic Inspection.

This type of inspection uses ultrasonic or stress waves transmitted through a part. It basically compares the trace of a standard undamaged laminate of similar thickness with the part being inspected. An inspector using ultrasonic methods must interpret any differences found and, therefore, needs a thorough knowledge of the structure being inspected. There are generally two types of ultrasonic inspection.

- Through-transmission ultrasonics (TTU), which uses two transducers (one to send the ultrasonic wave and one to receive it after traveling through the part), is typically limited to the factory because access to both sides of the part is required (exception: TTU can be applied to repair when the part has been removed from the aircraft).

- Pulse-echo (P/E) ultrasonics uses a single transducer and requires access to only one side of the part. It is suitable for different facets of fieldwork (i.e., damage detection and postrepair inspections).

Both TTU and P/E inspection can detect small defects through the thickness of a laminate and debonds between elements or facesheets and honeycomb core. Special training is usually needed for TTU and P/E. Test standards and a detailed understanding of the part design features (internally bonded elements and ply drops) are essential for determining the extent of damage.

2.8.2.3 X-Radiography (X-Ray).

The use of x-ray on composite parts that are constructed of carbon fiber-reinforced epoxy is difficult because the absorption characteristics of the fibers and resin are similar and the overall absorption is low. The properties of glass and boron fibers are more suited to the use of x-ray as an inspection method for composites. X-ray is often used to detect moisture ingression in honeycomb core of sandwich parts and is sometimes used to detect transverse cracks in laminates.

2.8.2.4 Eddy Current.

Eddy current has very limited use in detecting composite damages and for inspecting repairs for integrity. It is commonly used to detect cracks emanating from fastener holes in metal structures without removing the fasteners.

2.8.2.5 Thermography.

Two forms of thermographic inspection methods are currently available: (1) the passive method measures structure response to an applied heating transient, and (2) the active method monitors heating produced by applying cyclic stress to the structure. In both forms, surface temperature of the structure is monitored, usually with an infrared camera, and anomalies in the temperature distribution reveal the presence of composite damage. It can also detect moisture in honeycomb

sandwich structures and has been used by airlines for detecting moisture in the form of ice or water. Thermography may replace the moisture meter for detecting moisture in composite parts. Equipment needed for this method is expensive, but large areas can be inspected quickly.

2.8.2.6 Moisture Meters.

These devices are often used to detect the presence of moisture when making repairs to glass fiber-reinforced plastic or aramid materials. They can also detect moisture within aramid honeycomb core. However, the technique cannot be used with carbon or any other conductive material or with antistatic coatings that contain carbon.

2.8.2.7 Bond Testers.

This NDI uses instruments based on mechanical impedance measurements. Bond testers are typically used to detect composite delaminations and adhesive debonds. Bond testers are portable and well suited for inspection of facesheet core separation in sandwich structures when small anomalies are not considered to be important. Gross defects, such as wide-spread environmental degradation and facesheet debonds in sandwich structure, produce readily measurable changes in resonant frequencies.

2.9 REPAIR PROCESSES.

Many of the NDI techniques described in sections 2.8.2.1 through 2.8.2.7 are important for postrepair inspection, but the quality of the repair will not be completely determined from NDI. Instead, in-process inspections are needed to complement postrepair NDI. Some processing defects that occur during bolted and bonded repairs can be detected through NDI. This includes debonds, delamination, porosity, and any impact-handling damage. However, some processing defects cannot be detected using NDI. Adhesive bonding surfaces that are contaminated or improperly cured may not be reliably detected by postcure NDI using ultrasonic or other methods. Instrumented NDI is also not good at detecting errors in the ply stacking sequence for a laminated repair patch. To verify that all repair process steps are performed correctly, a QC plan is recommended so a technician or inspector can ensure that the approved repair instructions were followed during the process. Due to the limitations of current postrepair NDI techniques, it is essential that repairs are performed per the approved repair documentation, and that the separate repair steps are checked for having been performed correctly.

The two basic types of composite repair processes are bonded and bolted. The latter has processing steps that are similar to bolted metal repairs. However, there are several differences that need to be understood to successfully perform a composite bolted repair. The important technical issues for bonded repairs are similar to those for composite part fabrication. However, the issues become more difficult when addressed for bonded repairs made in the field, sometimes performed on-airplane. Since bolted repairs may start with laminate fabrication of repair patches, some of the issues that need to be addressed are common with the following discussions on bonded repairs.

Approved repairs for control surfaces must evaluate the effects on overall part stiffness, weight and balance, and flutter characteristics. Many composite control surfaces use a sandwich panel

design. Damaged sandwich panels typically require local core removal and a bonded repair to the facesheet. Following repair, the mass balance of a control surface must be checked against operational limits before returning the part to service. One possible issue when using composite curing for bonded repairs is that the part distorts or warps due to uneven cure or tooling problems. This issue can further cause problems with control surface clearances and deployment mechanisms. As a result, the clearances with adjacent fixed and movable structure should also be checked using the full range of deployment, including possible deflections when under load.

Other considerations for the repair of composite parts include the restoration of coatings and exterior layers used for various types of protection. A conductive coating, such as a copper mesh used for lightning protection, must be restored during repair. The primer and paint used to protect composite parts from UV degradation must also be restored. To protect against corrosion, fiberglass isolation plies are often used to separate carbon composite from aluminum parts.

2.9.1 Bonded Composite Repair.

Basic laminate fabrication involves creating fiber-reinforced composite parts from uncured material. The most common fabrication techniques use epoxy pre-impregnated tape and fabric materials (prepreg). The composite component is formed into a final shape using heat and pressure during the cure process. When including adhesive bonding in part fabrication (i.e., bonding used to attach precured elements), special care is needed to prepare the precured surfaces for bonding. Since much of the composite part has already been cured, bonded repair requires adhesive bonding on at least one of the repair surfaces. As is the case for a laminated part fabrication that includes bonded elements, bonded repair surface preparation is one of the most critical processing steps.

Some OEMs use autoclaves for curing large epoxy prepreg laminate components. This method provides vacuum, heat, and pressure to the bagged composite part. The addition of autoclave pressure provides ply consolidation that helps minimize internal defects, such as porosity. Autoclaves are generally not available for bonded composite repairs in the field. Repair materials are selected to create a repair that is both sufficiently strong and practical to process. Repair patches use materials that match the parent component or compatible nonparent materials. Parent material properties are usually not attained in a bonded repair due to reduced consolidation, fiber volume inconsistencies, and thermal anomalies during typical vacuum cures. Nonparent materials may be more appropriate for creating a nonautoclave bonded patch. However, all materials used for bonded repair must have approved specifications and data that substantiate the repair.

There are generally two types of bonded composite repairs. The first is called a prepreg repair, and the second is called a wet lay-up repair. Prepreg repairs can be made using either the original part prepreg material or a substitute prepreg material that has been approved for a specific prepreg repair. Shipping, handling, and storage of the prepreg must be controlled because it is perishable. To maximize life, prepreg must be stored at low temperatures in sealed bags until use and then allowed to reach room temperature before being removed from the bags and applied to laminate lay-up. Prepreg repairs can be performed using an autoclave if the damaged part can be easily removed from the aircraft. Approved prepreg repairs have used a

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vacuum bag and local heating source for on-airplane repair. Such processing does not enable the repair ply consolidation that autoclave processing provides. The resulting porosity reduces material strength.

Wet lay-up repairs are bonded repairs that use special two-part epoxy resins and dry fiber fabrics. The resin ingredients are kept sealed in separate containers at room temperature until mixed to requirements. The dry fiber fabrics, woven similar to fabric prepreg materials, are kept at room temperature. Wet lay-up repairs provide some storage advantages when approved for a given application. In the case of a wet lay-up repair, technicians must accurately mix the resin components and impregnate the dry fiber fabric layers.

The correct processing of bonded composite repairs is critical to the elimination of defects. This includes all processing steps, such as damage removal, surface preparation, material handling and storage, patch material lay-up, part bagging and cure, and postprocess inspection. In the case of a wet lay-up repair, resin mixing and dry fabric impregnation are added steps. In the case of a prepreg repair, the material must be removed from the freezer and allowed to thaw before opening the bag and starting the lay-up process. If a prepreg is not allowed time to reach room temperature, moisture will condense on the prepreg surface, potentially causing cure problems. Depending on the specific adhesive type used for a bonded repair, it will also have storage, handling, and mixing requirements that need to be followed. It is essential to realize that in-process inspections are at least as important as postprocess NDI for bonded composite repairs.

Before a bonded repair, all fluids must be removed from the damaged component using vacuum and heat. Failure to remove all moisture and fluids from the repair region of the component may cause a patch bondline failure. This may be particularly troublesome for sandwich construction where fluid ingression into damaged core (e.g., honeycomb) may cause internal vapor pressures when heated during cure, blowing facesheets off the core. It is also essential that the protective coating (e.g., conductive coating, if present, paint enamel, and primer) should be completely removed from an area larger than the bonded repair using a prescribed method such as abrading or sanding.

Damage removal and surface preparation must be performed prior to a bonded repair. The damage is typically removed using a specified scarfing technique. Scarfing removes the damaged material and creates a tapered surface so the repair material plies can be laid down in a stepped process. The tapered repair bond surface must be abraded and cleaned with a soft cloth moistened with an approved solvent. If this surface is not cleaned sufficiently, contaminates (e.g., dust) may be present in the repair bondline. The approved method used for abrasion and cleaning must provide a chemically activated surface that will bond with the repair patch and adhesive. The abraded surface must also have an accurate taper to closely match the geometry of the adhesive and repair patches applied or problems will occur during cure. Good surface preparation requires (1) an approved process shown to reliably work for the specific adhesive and composites included in the bonded repair, and (2) a technician with the skills needed to properly execute the process. Deviations may cause a poorly bonded repair that appears acceptable when inspected using a postbond NDI method. This highlights the importance of stringent in-process controls for the bond surface preparation steps.

Fresh prepreg material and film adhesive is essential for consistent and high-quality cured laminates. Porosity, delaminations, and debonds are often the result of using old material. Wet lay-up repair materials must be mixed to the correct proportions and used before the workable out-time expires. All materials must also be kept free of debris and fluids during processing to prevent strength degradation through contamination.

The lay-up of adhesive and plies in a repair patch also requires careful attention to detail. The adhesive is placed down first, then each repair ply is placed down in proper sequence, and last, a final ply (or plies) is placed over the tapered repair. All plies must be laid up per the dimensions and orientations specified in the patch-stacking sequence. Most postrepair NDI methods are unable to determine individual repair ply orientations, again highlighting the importance of in-process QCs. Each ply is swept to remove any wrinkles during lay-up. If required by the SRM or approved repair documents, the repair plies are compacted (debulked) using a temporary vacuum bag. Composite repairs may need to be debulked every 5 or 6 plies to remove any air that gets trapped between plies during lay-up. Air trapped in a laminate can cause wrinkles and porosity during cure.

Following patch lay-up, a bonded composite repair is bagged with the requisite bleeder and breather plies, and the thermocouples are placed as specified. Bagging systems must be vacuum tight to ensure proper ply consolidation during cure. Assuming a nonautoclave cure, a specified minimum level of vacuum is applied, and the repair cured using a heat blanket or other heating device per the specified cure cycle. It is essential that the vacuum does not fall below the specified minimum level (e.g., 22 inches of mercury). The specified heat-up and cooldown rates, dwell temperature, and vacuum pressure are monitored periodically throughout the cure cycle to ensure requirements are met.

A repair cure cycle must be controlled on-airplane or in an autoclave. Substructure heat sinks for on-airplane bonded repairs can cause the cure temperatures to vary by drawing heat away from the repair zone. For this reason, it is important to be cognizant of the substructure when placing the thermocouples. If underlying structure or equipment will be adversely affected by adjacent heating, the equipment must be either removed prior to the repair, or protected from excessive heat. During the cure cycle, any loss of vacuum, autoclave pressure, or temperature can result in anomalies such as voids, porosity, and delamination. These problems can be detected by postrepair NDI. An improperly cured part may also have lower than required thermal stability in addition to lower mechanical properties. If such problems occur without indications of porosity or delamination, the NDI may not detect an issue. Instead, in-process control measurements of temperature and vacuum are needed to identify a possible problem associated with under- or overcure.

Postprocess NDI of a bonded repair is performed after specified cooldown and removal of bag and cure materials. Approved repair documents should specify NDI procedures to be used. Qualified inspectors are typically needed for most postprocess NDI methods. The NDI can find processing anomalies, such as voids, delaminations, and porosity, which occur during the cure process, and may be the result of poor tooling, insufficient ply consolidation, low autoclave pressure, or loss of vacuum during the cure cycle. The NDI can also detect handling damage on laminate edges, impact damage and delaminations from poorly machined parts (i.e., drilled holes

or edge trim), or improper assembly. The NDI measurements combine with in-process quality checks to indicate that the repair is satisfactory. Once such a determination is made, protective coatings can be restored over the repaired area, per approved documents, and the component can be returned to service.

Many OEMs have a factory process called the Material Review Board (MRB). The MRB is a process that is intended to make team dispositions concerning reported defects or unsatisfactory raw material and take corrective actions as necessary. A similar process should be established whenever questions arise in composite field repair. Factory or field dispositions may include repair or rework, scrapping the material or part, or using the material, part, or repair as is. Corrective action may be taken to reduce the number of repetitive errors or defects in the fabrication or repair process. The MRB process starts with an inspection report (rejection notice) provided by an inspector. The rejected material, part, or repair is reviewed to determine its disposition. It may be used as is, scrapped, reworked, or, if raw material, returned to the vendor.

2.9.2 Bolted Composite Repair.

The use of bonded composite parts in aircraft structures enables the elimination of thousands of mechanical fasteners that exist in similar metal components. However, mechanical fasteners are still used for joining the more highly loaded composite elements and components. Benefits from the use of composite bolted joints include the higher joint reliability of discrete fasteners, the improved inspection capability, and the ability for possible disassembly during maintenance. It is important to understand the effect that holes and loaded fasteners can have on the strength of the composite laminates being joined. An open hole in a composite laminate produces stress concentrations that can significantly increase the stresses at the edges of the hole compared to the stresses in the unnotched section of the laminate. The bearing stress of the fastener, which transfers load from one part to the other, must be added to the stresses at the edge of the hole. All of these stresses cause significant reductions in strength of the laminate in the joint area.

The use of mechanical fasteners to assemble airframe structural components or elements is a mature technology. Composite part joining is no exception to this. Failure modes for composite-fastened joints are similar to those for metallic-fastened joints. Despite their similarities, the behavior of composite-fastened joints differs significantly from that of a metal. Composite-fastened joint behavior is lay-up dependent, with the best performance typically coming from a laminate that is quasi-isotropic (e.g., same number of plies in 0°, 45°, -45° and 90° directions). Most composites have low tensile strength in the through-thickness direction. This necessitates the use of specialized fasteners with longer tail areas. Fastener flexibility under loading can have a more significant effect on the strength of a composite-fastened joint than for metals. Increased local bearing stresses on outer plies of a laminate due to bolt bending are more critical for composites. This can lead to delamination and microbuckling of the fibers, crushing the outer plies and reducing joint strength. Composites are more sensitive to fastener clearance. Any kind of close tolerance or transition-fit fasteners will cause damage to the holes in the composite laminate parts when installing the fasteners. Lack of fit can have a serious effect on the efficiency of composite-fastened joints. This leads to a need for careful hole drilling and fastener installation.   Care must also be taken when mechanically assembling composite components. Using the fastener clamp-up to close gaps between parts can lead to delamination.

Metal parts can bend to accommodate moderate gaps. It is essential to use shimming if part tolerances or warpage cause composite fit-up problems.

Bolted repairs can be smaller than bonded repairs, and thus are often used for the repair of composite parts when the thickness of the part requires a very large scarfed out area for bonding. Typically a 30:1 scarf angle is used for bonded repairs, which can cause the area of the scarf to be more than double the size of a bolted repair area. For example, for a 2-inch-diameter hole in a 0.5-inch-thick laminate, a bolted repair might cover 144 in^2, while the area for a bonded repair with a 30:1 scarf angle might cover 800 in^2. Also, bonded repairs to a component with complex geometry (excessive ply drops, cobonded or cocured stiffening structure, or bolted spars and ribs) can be quite difficult to implement due to the intricate scarfing and patch lay-up requirements.

The main consideration for any aircraft component repair is that, in general, aircraft components, such as wings, stabilizers, and fuselage skins, are loaded in multiple directions. A bolted composite repair has to be carefully designed, and knowledge of the component design loads is essential. The quality of holes drilled in a component that is loaded in multiple directions can have significant effects on the capability of that component. Most repair technicians do not have knowledge of the component design loads when repairing a damaged component with a bolted repair. Hence, it is essential to follow the approved repair instructions in source documentation. Consistent fastener fit is essential for good load sharing in a repair fastener pattern. The most efficient composite bolted joints use fasteners and holes drilled for a near net or a very slight interference fit. If holes are drilled with varying tolerances or at off-angles, fasteners will not all fit the same, and some fasteners will unload before others, increasing the peak bearing stresses. In extreme cases, holes with variable fits can lead to failure of the bolted repair.

There are distinct differences in fastener hole drilling for composites versus metals. This includes differences in drill types, lubrication, speed, and feed rates. Hole drilling of metal parts is typically performed using steel drill bits, a relatively slow drill speed, and a lubricant. As an example, the speed for drilling a 0.25-inch-diameter hole in a titanium metal part is usually in the order of 450 revolutions per minute (rpm) with the drill bit lubricated. Drilling holes in composite parts requires a totally different approach. Fiber breakout and/or delamination can occur during drilling of carbon parts if proper procedures are not followed. Drill bits for composites are usually solid carbide or diamond-tipped, and drill speeds need to be much higher than for metals. Drilling a 0.25-inch-diameter hole in a carbon composite part is often performed without a lubricant and requires a drill speed on the order of 5000 rpm. To eliminate defective holes, high drill speeds and a slow, controlled feed rate are used for composite laminates. Higher hole-drilling speeds are used to reduce pressure and allow low feed rates. Excessive pressure (i.e., high feed rates) during hole drilling causes delamination and fiber breakout. In addition, fiber breakout of the last ply to be penetrated by the drill bit can be avoided by using a block of wood as a backing block.

Special efforts are needed to keep machining and drilling temperatures down when working on composite parts. Temperatures should be kept below the glass transition temperature (T_g) of the material being worked on. The T_g of an epoxy material is the temperature at which the material transitions form an elastic state to a viscoelastic state. Drilling or cutting at temperatures

approaching the material T_g may cause clogging. High speeds and pressures cause high temperatures at the cut edges. High speeds used for machining or drilling composites require that pressure be kept low by significantly reducing the feed rate below that typically used for machining or drilling metals. It is important to follow drilling instructions contained in source documentation, because they typically detail feeds and speeds for drilling composite parts that will ensure temperatures at the cutting edge, well below the material T_g. Some composite materials require additional care during drilling and machining. Composite parts made with aramid fibers require special cutting tools and techniques. Aramid fibers tend to fuzz when cut. The same is true when cutting or machining aramid honeycomb core.

When machining, cutting, or drilling metals, the material debris is in the form of shavings or chips. These metal shavings are often carried away by the lubricant or forced air that is used in the process. This does not happen when performing the same operations on parts made of composite materials. Due to higher cutting and machining speeds, the composite debris is in the form of a fine dust. To prevent repair technicians from ingesting this dust, approved vacuum equipment should be used with all composite cutting, grinding, or drilling tools. Pneumatic tools with rear exhausts are recommended so air is directed away from the work and dust is not blown around the shop. Vacuum attachments can be used to provide dust extraction.

When both metal and composite parts are included in a bolted repair, hole drilling is usually performed separately on each part. Low speeds and high feed rates for metal cause overheating and fiber breakout in composites. The high speeds and low feed rates needed for composites are not effective for metal hole drilling. A final hole reaming is typically used to ensure proper hole quality.

The surface of damaged composite parts must be prepared for bolted repairs. The surfaces should be clean and it is essential that there is no protruding damage (e.g., fibers) that may prevent the repair doubler and base composite part from mating properly. If a repair doubler is not in proper contact with the part being repaired, fastener installation may be affected (e.g., effectively changing the pull-up forces or fastener grip length).

Special fasteners are used for composite bolted repairs. Due to corrosion considerations, corrosion-resistant steel, inconel, or titanium fasteners are needed for bolted repairs of carbon composites. Most fastener types work for fiberglass and aramid composites. The fastener type, size, grip length, and installation instructions specified in approved repair documents must be used to ensure a bolted repair is performed properly. If a fastener has insufficient grip length, concentrated bearing stresses on the fastener shank may result. Also any threaded portion of the fastener that bears on the laminates being joined may cause damage inside the holes, as well as creating a potential for incorrectly clamped sleeves. Fasteners with an incorrect grip length have contributed to bolted repair failures.

Fasteners must be installed with proper clearance, sealant, sizing, and clamp-up per specifications. If the fastener is incorrectly installed (i.e., with insufficient clamp or unsatisfactory sleeve formation), the designed repair strength may not be attained. In cases where the back face of the part being repaired is not easily viewed, it is very important to perform the fastener installation correctly, using the specified tool (e.g., blind lockbolt

installation). If the sealant between the repair parts and the edge seal has not been applied correctly, moisture paths may result, which can lead to fastener corrosion and freeze-thaw damage and laminate moisture absorption through the damaged area.

As with bonded repairs, bolted repairs benefit from in-process QCs. The use of two technicians to share in the bolted repair tasks can provide the in-process checks needed to avoid defects. On any given step, one technician can serve as the inspector for proper use of tools, equipment, and procedures, while the other performs processing tasks.

3. SAFETY MESSAGES.

The following is a summary of critical safety issues related to composite maintenance and repair, and emphasizes the key roles of engineers, inspectors, technicians, and Federal Aviation Administration (FAA) personnel and their inter-relationships in the repair process.

Table 1. Critical Safety Issues

Understand Roles and Responsibilities of Key Teammates	Recognize Composite Damage Types and Sources	Identify and Describe Information Contained in Documentation	Composite Laminate Fabrication and Bonded Repair Methods
Engineers are responsible for always authorizing repairs that will ensure flight safety.	Engineers must ensure that each disposition requires a damage assessment well beyond the initial damage site.	Documentation is the only way a repair can be shown to be flightworthy.	Engineers are responsible for creating repair designs and repair processes that will ensure flight safety.
Inspectors are responsible for only allowing repairs to enter service that meet necessary standards.	Inspectors must assess all visible and nonvisible damage in an area well beyond initial damage site.	Maintenance engineers and inspectors must know the documentation that defines the proper repair procedures.	Inspectors must correlate repair data and records as the primary method for knowing that composite repairs are flightworthy.
Technicians are responsible for processing a repair exactly as stated in the repair documentation.	Technicians must remove all damage prior to processing the composite repair.	Technicians must know the limits of their authorization to process composite repairs as detailed in approved documentation.	Technicians must always adhere to approved processes and methods so that each composite repair is flightworthy.

Table 1. Critical Safety Issues (Continued)

Understand Roles and Responsibilities of Key Teammates	Recognize Composite Damage Types and Sources	Identify and Describe Information Contained in Documentation	Composite Laminate Fabrication and Bonded Repair Methods
Managers are responsible for ensuring that all repair materials, designs, processing, inspection, and certification are conducted per the approved procedures and documentation.	Managers must ensure that people and processes always detect and remove damage from the entire damage site.	Managers must instill processes and only employ personnel who always follow instructions exactly as stated in approved documentation.	Managers must enable training and the proficiency of all repair personnel. Managers are responsible for employing capable, qualified repair personnel.
FAA personnel must only authorize repair systems at OEMs, airline repair stations, and MROs that ensure flightworthy composite repairs.	FAA personnel must qualify inspectors and only authorize inspection methods that are fully capable of detecting and removing all damage to composite structure.	FAA personnel must ensure that all repair professionals adhere strictly to approved documentation.	FAA personnel must authorize only repair processes and methods that produce flightworthy repairs.

MRO = Maintenance and repair organization

The following safety message descriptions focus on the major elements involved to ensure safe aircraft operation after composite maintenance and repair.

3.1 ALL ASPECTS OF COMPOSITE MAINTENANCE ARE INTERLINKED.

- The damage to a composite aircraft component must be discovered by inspection (if discovered by operations personnel, it must be reported to maintenance)

- The damage must be assessed by qualified maintenance personnel

- Damage disposition must include an interface with engineering personnel familiar with approved data for the structure in question

- If repair is necessary, the repair design must be approved by the appropriate authority (previously approved data such as what is documented in an SRM or other approved repair method)

- The chosen repair method must also be of an approved type (previously approved data such as what is documented in an SRM or approved by an FAA designated engineering representative (DER))

- The approved repair process must be performed by a qualified technician, using qualified materials, and strictly adhering to the appropriate in-process controls

- The completed repair must be inspected for approval by a qualified inspector

- If any of the above maintenance actions contain errors or flaws, then a deficient repair may result. It is essential for safety that qualified inspectors, technicians, and engineers are involved in composite maintenance and repair.

3.2 DAMAGE DISPOSITION MUST BE BASED ON APPROVED DATA.

Composite aircraft structural components can be damaged by many different sources. In many cases, these components are designed to withstand damages up to levels within the bounds of performance and cost.

When damages are above these levels and approved data exists for a repair, source documentation (e.g., SRM) will typically provide the necessary instructions for maintenance actions. This will include the appropriate NDI methods to accurately map the extent of the damage and an approved repair that can be performed to allow the aircraft to be returned to service. If the damage is of a level not covered by previously approved data that has been documented and made available, then the OEM must be contacted for repair disposition or an FAA DER will be needed to develop a repair design and generate the data needed to substantiate the repair.

Damage disposition and subsequent repair designs and processes must be based on approved data that substantiates the structural integrity. Without such data, the airworthiness of the structure is in question.

3.3 USE OF APPROVED SOURCES IS VITAL TO PROPER AIRCRAFT MAINTENANCE AND REPAIR.

The approved sources of technical data, maintenance and repair instructions, guidelines, and regulatory requirements contain information vital to proper aircraft maintenance and repair. It is essential to be familiar with the specific aircraft structure drawing system and approved maintenance methods for the particular component in question. The use of inappropriate or unapproved maintenance instructions for a given part, including inspection and repair processes or alternate repair materials used on other structures, is not allowed. While inspectors and technicians are not required to be fully cognizant with the Code of Federal Regulations, Advisory Circulars, and Airworthiness Directives, it is appropriate that all members of the repair team be aware that these regulatory requirements exist and that they are understood by at least one person in the repair process.

Any lack of understanding of the structural detail or deviation from the approved data, maintenance and repair instructions, or regulatory requirements can lead to unacceptable maintenance procedures and a defective repair.

3.4 CORRECT PROCESSING OF COMPOSITE COMPONENTS IS CRITICAL.

The correct processing of composite components is critical to the elimination of processing defects. This includes all steps in processing, such as surface preparation, material handling and storage, material lay-up, part bagging and cure, and postprocess inspection. Inspection of bonded composite and metal-bond assemblies, using appropriate NDI techniques together with in-process QC, is essential to the delivery of defect-free composite components for assembly to the aircraft. Equally, adherence to the correct repair processing steps, in-process QCs, and postprocess inspection is crucial to successful composite repairs. In addition to the above steps for the original part fabrication, composite repair procedures will require damage removal and acceptable surface preparation for bonding. Note that these procedures may also include drying. The particular fabrication and repair methods for a given structural part will often include specific processing details that are unique to that part.

In the event of processing mistakes, misuse of equipment, or misinterpretation of in-process QCs and postprocess NDI results, composite part fabrication or repairs may be compromised. Parts delivered with unaddressed defects and compromised repairs may not be airworthy.

3.5 IN-SERVICE INSPECTIONS OF COMPOSITE COMPONENTS IS NECESSARY FOR SAFETY.

In-service inspections of composite components are necessary for safe flight operations just as they are for those components fabricated from metals. In-service damages to composite parts from various sources are likely to occur during an aircraft's operational life. Per maintenance instructions, damages may be detected using visual inspection or by directed NDI. Visual indications of outside-surface damage should be followed up with a back side inspection if accessible. If damage is first detected using visual methods, NDI techniques, such as P/E or even a simple tap hammer, will generally be needed to determine the full extent of the damage and make the correct disposition. The correct use and interpretation of NDI are required to accurately define the extent of damages so that correct damage dispositions can be made.

In the event of in-service damages, it is crucial for safe flight operations that these damages are discovered, either by operations personnel or directed maintenance inspections, before they become critical. After damage has been discovered, the correct damage disposition must be made so the damage can either be determined to be acceptable or the damaged component can be repaired and the aircraft returned to safe flight operations.

3.6 CORRECT PROCESSING IS CRITICAL TO THE ELIMINATION OF DEFECTS IN BONDED COMPOSITE REPAIRS.

As in the fabrication of composite bond assemblies, correct processing is critical to the elimination of defects in bonded composite repairs. Bonded repairs are more difficult to inspect for structural integrity than bolted repairs. As a result, correct procedures in surface preparation,

moisture elimination, proper material storage and handling, and in-process controls are essential for defect-free repair bondlines and patches. A poorly bonded repair may be more detrimental to flight safety than the original damage, due to the potential for repair separation during flight. If the bonded patch is not effective or separates in service, the resulting damage size may be larger and more critical than the original damage. A repair that separates in flight can cause damage to other components. If a repair to a flight-critical component does not completely separate, it could cause flutter of that component or other components to which it is attached.

Strict adherence to repair bond process procedures and in-process QCs are vital for flight safety as well as to prevent the need for costly reworks. Current postrepair NDI methods are necessary but not sufficient proof of a good bond.

3.7 MISUSE OF EQUIPMENT OR MISINTERPRETATION OF INSPECTION RESULTS MAY BE DETRIMENTAL TO SAFETY.

Defects may be present in the bondline or within the repair patch due to poor surface preparation, material storage and handling, and cure process mistakes. It is essential to use the appropriate inspection methods for specific types of bonded repair (i.e., sandwich, laminate-stiffened or metal bond components). Postrepair inspection should be visual, followed by using an appropriate NDI technique. Visual inspection can be, in some cases, just as valuable as NDI methods such as P/E for detecting flawed bonded repairs.

Misuse of equipment during postrepair inspections or misinterpretation of inspection results may be detrimental to safety.

3.8 IMPROPER BOLTED REPAIR MAY IMPAIR FLIGHT SAFETY.

In general, there will be differences in the machining and fastening steps needed to complete a bolted repair of composite structure. Specific differences in bolted repairs should also be expected for different composite structures. The tolerances for installation of fasteners during composite bolted repairs are, in many cases, more critical than for installation of fasteners during the repair of metal parts. Drilling of composites is more difficult than drilling metals, and the selected fasteners should not cause delamination during installation. Details of the bolted repair design are also of great importance. A repair design that is not substantiated by approved data may cause problems for component margins of safety while repairing the damaged area adequately. Fastener installation equipment, such as torque wrenches, must be correctly calibrated and standards kept up to date.

The design of the bolted repair, selection of incorrect fasteners, poorly drilled holes, and incorrect fastener installation can all contribute to a defective or unapproved repair and impaired flight safety.

3.9 EACH MEMBER OF THE REPAIR TEAM HAS A ROLE AND RESPONSIBILITY.

All aspects of a composite repair are interlinked, and teamwork is essential to provide a safe damage and repair disposition. It is essential that each team member recognizes their limitation to (1) disposition a damage, (2) determine the repair design and process that have been

substantiated by approved data, (3) perform the approved repair, and (4) inspect and approve the completed repair.

If any of the above repair aspects contain errors or flaws, then the aircraft may be dispatched with a deficient or unapproved repair.

4. TERMINAL COURSE MODULES AND OBJECTIVES TO THE AWARENESS COURSE.

The course was developed through the use of TCOs. The teaching profession found this method to be useful in constructing new courses that require inputs from different sources. The TCOs were developed with participation of industry experts for the needed course content and thus represents industry consensus (the details of which are given in appendix B). The TCO process is a bottoms-up process where the lowest level contains the TCOs. The 60 or so agreed-on TCOs were apportioned to 13 modules, which were further grouped around 4 topics to form the complete course. The four topics are base knowledge, teamwork and disposition, damage detection and characterization, and repair processes. The course on base knowledge is a prerequisite to the courses containing the other three topics and must be successfully completed before proceeding further. As this course meets the intent of the TCOs, which are in compliance with experts' consensus input, the course becomes an industry standard.

For clarity, course modules are provided at the third level (e.g., 4.1.1), and the TCOs are bulleted.

An instructor's guide is provided in appendix A. It reflects suggested content corresponding to the TCOs listed in this section. The numbering of the paragraphs is consistent with the TCOs. Using the instructor's guide is optional, and it can be used however the teaching organization deems useful.

4.1 BASE KNOWLEDGE.

Base knowledge content provides a basis for the awareness course. Students must have a grasp of the terminology and basics of composites' technology to fully appreciate and understand critical issues in maintenance and repair. Understanding the basics of this technology, having an elementary familiarity of composites' maintenance, and the awareness of other important issues are prerequisite to further study in the course. Teaching organizations should develop assessment tools to determine the level of student understanding prior to advancement to the awareness courses.

4.1.1 Understand the Basics of Composite Materials Technology.

- The student will be able to distinguish among adhesive, resin, fiber, and core applications and uses.

- The student will be able to describe various composite processing parameters.

- The student will be able to describe composite material properties, failure modes, and effects of processing.

- The student will be able to describe various composite machining, assembly, and finishing processes.

- The student will be able to describe stiffened laminate and sandwich applications and structural properties.

- The student will be familiar with information sources that have a glossary of terms for reference.

4.1.2 Understand the Basics of Composite Materials Maintenance and Repair.

- The student will be able to list the basic steps in maintenance procedures from damage detection through repair completion.

- The student will be able to list key composite and expendable materials needed for a simple laminate structure repair, including the storage requirements.

- The student will be able to list the necessary tooling and equipment to accomplish a simple laminate structural repair.

- The student will understand personal and equipment safety requirements.

- The student will be able to describe the differences between repairing composite and metal structures, including discussions on the special issues related to composite and metal bonding.

- The student will be able to describe the process of metal bonding.

- The student will be able to discuss the importance of approved data, methods, and procedures used in product-specific composite maintenance and repair.

4.1.3 Understand Other Critical Elements of Composite Maintenance and Repair.

- The student will be able to discuss basic issues effecting the selection of bonded or bolted repairs.

- The student will be able to describe various electrical requirements and effects, including prevention of corrosion, hazards of electromagnetic interference and electrostatic discharge (lightning protection systems), and how they need to be considered during repair processes.

- The student will understand the need for protective coatings and surface-finishing steps used for composite aircraft structure.

- The student will be able to describe typical paint and surface layer removal techniques for finished composite parts.

- The student will know the issues related to the proper disposal of wastes from the composite repair process.

- The student will know why understanding personnel skill limits and where to receive assistance during maintenance is essential.

- The student will be able to discuss the importance of documenting and sharing information about damage scenarios discovered in service between OEM, maintenance and repair organizations, and regulators.

4.1.4 Be Aware of Composite Maintenance Research and Industry Developments.

- The student will be aware of the general lack of available standards for composite engineering practices and realize that composite maintenance and repair requires special skills.

- The student will discuss emerging advances in repair process technologies that are in development for bonded and bolted repair and QC.

- The student will discuss emerging damage and repair inspection technologies, such as bond testing, moisture meters, and interferometer (three-dimensional characterization).

- The student will discuss advances in composite repair analysis and design that may be used by engineers to develop a repair for an aircraft structure.

- The student will be able to outline a process for the structural substantiation of a repair and approval to meet regulatory requirements.

4.2 TEAMWORK AND DISPOSITION.

This section of the class is organized to highlight the importance of teamwork in the disposition of damaged composite structure during maintenance and repair. Each team member must have training that qualifies them for the specific tasks to be performed. The training would start with an awareness of critical composite maintenance and repair issues, such as those provided by this course. Additional training is needed to develop specific skills and perform tasks on the structural components of a given aircraft type that has unique repair design, process, and inspection details. It is unlikely that one person will attain all the special skills needed for engineering disposition, inspection, and repair of damage to a given structure. Instead, trained engineers, inspectors, and technicians will work as a team from the time of damage detection until an approved repair is properly executed. Approval of a new repair design and process for critical composite structures will require an interface with engineers that have the necessary skills and data to substantiate structural integrity of the repair, per regulations. Students are also exposed to documentation used in composite maintenance and repair, including those describing acceptable field disposition procedures and the associated regulatory rules that must be followed. Course work is supplemented by a laboratory to provide hands-on experience for students to work as a team in damage assessment, disposition, and repair.

4.2.1 Underscore Roles and Responsibilities.

This part of the class will help students understand the roles and responsibilities of engineers, inspectors, and technicians involved in composite maintenance and repair. It also discusses the vital interface between personnel trained in composites and personnel that may be unaware of a need for damage disposition, such as flight crew members, operations staff, and line maintenance. The content introduces several subjects covered in greater depths by other sections of the course. These include an overall view of the composite inspection, damage disposition, and repair process. It also covers approved documents used to guide the team on acceptable designs, inspection methods, and repair processes. The content of this section should be presented at the start of the course to help link different facets of composite repair and maintenance.

- The student will be able to describe critical safety aspects of repair design, process planning, and approval.

- The student will be able to briefly describe the steps in composite damage detection, inspection, and repair processes.

- The student will be able to list basic NDI methods used for damage assessment and postrepair inspection.

- The student will be able to distinguish between the skills needed for structure engineers, inspectors, and technicians dealing with composite maintenance and repair.

- The student will be able to recognize his or her skill limits in practice and where to find assistance.

4.2.2 Identify and Describe Information Contained in Documentation and Regulatory Rules.

This part of the course is intended to expose students to documents used in composite maintenance and repair, including those describing acceptable field disposition procedures and the associated regulatory rules that must be followed. Source documents for specified structural components outline the inspection procedures and the repair design and process details, which were substantiated to meet regulations. Limits on the damages and repairs covered by such documents are needed to control field disposition. Knowledge of the regulatory rules and the test data and analysis needed for structural substantiation provides a basis for gaining approval of repairs outside the limits. The content of this section is best taught after the student has gained an awareness of roles and responsibilities and is exposed to composite damage types and inspection procedures.

- The student will be able to describe requirements in material and process specifications and approved repair information.

- The student will be able to demonstrate the use of source documents.

- The student will be able to identify and demonstrate the use of regulatory documents.

- The student will be able to describe the requirements and engineering approvals necessary for valid sources of technical information and maintenance instructions.

4.2.3 Demonstrate Awareness of Course Principles Applied to Composite Damage Disposition and Repair.

This part of the class, which is organized as a lab, helps integrate principles previously learned on key aspects of teamwork and the use of source documentation. In particular, the special team skills needed for damage inspection and disposition, as well as those applied to execute and evaluate the quality of an approved repair, will be applied in a case study. This lab is envisioned for teams of students, which include at least one member for each of the three major functions (inspector, technician, and engineer). Even if the individuals that comprise the team do not have a career path that is leading to the specified discipline, they should be aware of the roles and can take that position for purposes of the lab. The content of this lab is best shared at the end of the course, after students have been exposed to all other areas.

- A team of students will identify a damaged structural component and understand the specific configuration and materials used for fabrication of the damaged component.

- A team of students will perform a damage assessment and map the damage as accurately as possible using visual inspection, the tap test, and P/E ultrasonic equipment.

- A team of students will use source documentation represented by the training repair manual (TRM), which is a generic version of an OEM's SRM, to understand the component-allowable damage limits, and review any repair options that are based on the mapped damage contained in the TRM.

- A team of students will write an appropriate repair procedure and in-process QC plan based on the chosen repair option.

- A team of students will write an appropriate postrepair inspection and approval plan.

4.3 DAMAGE DETECTION AND CHARACTERIZATION.

This section of the class is intended to familiarize students with the typical types of composite damage, characteristics of the damages, and appropriate inspection procedures. It integrates technical information and labs to ensure students understand what causes damage in composite materials and the inspection methods needed for detection and complete damage characterization. Although most composite damages and manufacturing anomalies have some visual indications, it is essential to realize that there will also be hidden features that need to be understood for appropriate damage disposition and repair. Source documents are used to identify the design details and appropriate inspection procedures for damage and repair in a particular location of a structural component. Once damage is adequately characterized, source documents will be used for disposition and repair. Damage and repairs that are beyond the limits of source documents will require special approvals, often from the OEM, to substantiate that the repair meets structural requirements.

4.3.1 Recognize Composite Damage Types and Sources.

This part of the class will help students learn about the types of damage that can occur to sandwich and skin-stiffened composite structures. This will include some understanding of the most critical damage features affecting structural integrity. Some initial exposure on how to interpret damage characteristics, disposition, and repair versus data provided in source documents will provide a basis for subsequent course content. This will include a brief introduction to damages and repairs outside of those covered by source documents and the need to follow the regulatory process in seeking approval. A lab is provided to allow students to damage composite laminates in a controlled environment. The content of this section and the associated lab is best provided after exposure to roles and responsibilities but before parts of the course that provide more detailed content on composite inspection, disposition, and source documents.

- The student will identify sources and characteristics of damage to composite sandwich and laminate-stiffened structures.

- The student will describe damage types and their significance to structural integrity.

- The student will understand the information and analysis necessary for repair design and process development and substantiation.

- The student will distinguish differences in repair disposition procedures for damages covered by source documentation and those that are not. A TRM will be used to represent source documentation for the purpose of this course only.

- The student will describe the regulatory approval process for damages not covered by source documentation.

- A team of students will damage laminate coupons in a controlled laboratory environment and visually inspect the extent of any front and back side surface damage. (laboratory)

4.3.2 Describe Composite Damage and Repair Inspection Procedures.

This part of the course is intended to help students understand basic NDI techniques for detecting damage in composite laminate and sandwich components and for bonded composite repair patch postprocess inspection. Emphasis is placed on describing the NDI techniques that are currently available in the field. Sessions are taught on the types of damage dispositions that are often encountered and on the steps required for postprocess inspection of a bonded repair. Two laboratory sessions are conducted—one demonstrating damage assessment and one demonstrating postrepair verification—to give the student an appreciation for the intricacy associated with NDI processes.

- The student will describe NDI techniques currently available in the field, including an assessment of their strengths and weaknesses.

- The student will describe the critical steps necessary for making damage dispositions, including inspection and a draft process of the QC plan for repair.

- The student will describe the critical steps necessary for inspecting a completed bonded repair, including NDI procedures and interpretation of the results.

- A team of students will demonstrate and perform various damage assessments, including visual inspection, tap test, and ultrasonic P/E inspection. (laboratory)

- A team of students will demonstrate and perform various postrepair acceptance inspections, including visual inspection, tap test, and ultrasonic P/E inspection. (laboratory)

4.4 REPAIR PROCESSES.

This section of the course is organized to introduce students to bonded and bolted composite repair processes. The importance of conducting a composite repair per established processes as specified in repair documentation is emphasized. Repair technicians and inspectors must have training that qualifies them for the specific tasks they will perform. The training starts with an awareness of critical issues for composite bonded and bolted repair processes, such as provided by this course. A combination of technical information and laboratory exercises, which provide some exposure to the steps involved in composite repair processes, is used to help build this awareness. Additional training is needed to develop specific skills and perform tasks on the structural components of a given aircraft type that has unique repair design, process, and inspection details. Engineers and others involved in repair design, planning, and oversight also need the basic composite process awareness training provided by this section of the course as a basis for future study.

4.4.1 Describe Composite Laminate Fabrication and Bonded Repair Methods.

This part of the course is intended to help students understand basic composite laminate and sandwich component fabrication methods. It is also intended to help the students understand the basics of composite bonded repair. Emphasis is placed on correct processing for fabrication and repair to highlight the correlation between processing and component performance. The course will cover the key characteristics and typical defects that are encountered in the fabrication and in the repair of composite structures.

- The student will understand the basics of composite laminate fabrication.

- The student will understand the basics of composite bonded repair.

- The student will describe the detailed processing steps necessary for laminate fabrication in the factory, bonded repair (maintenance base or line station), and disposition by the original manufacturer's MRB.

- The student will describe the key characteristics and processing parameters for laminate fabrication.

- The student will identify typical processing defects that occur in composite laminate fabrication and bonded repair.

- The student will describe differences between wet lay-up and prepreg bonded repairs to sandwich and laminate parts.

4.4.2 Perform a Simple Bonded Composite Repair.

This part of the class is intended to help students understand the principles of a bonded composite repair through the hands-on experience of performing a repair in a laboratory environment. All steps of the bonded repair process are demonstrated, including damage removal, drying, surface preparation, patch lay-up, vacuum bagging, cure, and inspection. Key repair process parameters are discussed to help students appreciate the intricacies of making a successful repair. A comparison between metal bond repair and composite hot-bond repair will also be discussed.

- The student will demonstrate and apply common drying and surface preparation techniques, including how to inspect for acceptability.

- The student will demonstrate and apply material lay-up and compaction processes for a simple laminate panel repair.

- The student will demonstrate how to prepare and cure a simple bonded repair to a laminate panel and will explain the types of errors to avoid.

- A team of students will prepare a bonded repair for cure, including bagging and heating apparatus and cure. (laboratory)

- The student will describe process parameters, which effect bonded repair quality, and in-process controls necessary to avoid defects.

- The student will demonstrate critical in-process QCs during laboratory bonded repair process trials.

- The student will describe metal bonded repairs and how they differ from composite bonded repairs.

4.4.3 Describe Composite Laminate Bolted Assembly and Repair Methods.

This part of the course is intended to help students understand basic composite bolted assembly methods. The differences between bolted assembly for composites and metals will be highlighted. The students will be given an understanding of basic composite bolted repair compared to a bonded repair.

- The student will describe the basics of composite bolted structural assembly and show the differences between composites and metal bolted assembly.

- The student will describe the basics of composite bolted repair and show the differences between drilling and cutting composites and metals.

4.4.4 Perform a Simple Bolted Composite Repair.

This part of the course is intended to help students understand the principles of a bolted composite repair through hands-on experience while performing a repair in a laboratory environment. The differences between metal and composite drilling and machining are demonstrated to instill the idea that composites must be treated differently than metal components. Key repair process parameters are discussed, including correct fastener selection and inspection, to help students understand the requirements for making a successful repair. All steps of the bolted repair process are demonstrated, including damage removal, patch machining, drilling, sealing, fastener installation, and inspection.

- The student will demonstrate and show the differences between composite drilling and metal drilling.

- The student will describe process parameters, which effect bolted composite repair quality, and in-process controls necessary to avoid defects.

- A team of students will demonstrate and apply common damage removal, surface preparation, and drilling and fastening techniques used for bolted composite repairs and how to inspect the repairs for acceptability. (laboratory)

- A team of students will verify correct fastener selection, inspect drilled holes, and check whether the fasteners were properly installed during bolted composite repair laboratory trials. (laboratory)

5. COURSE STRUCTURE AND DESIGN.

This section is provided to assist curriculum developers. While the framework of any course development is the establishment of student and instructor expectations through TCOs, it is ultimately the teaching organization that will provide the format and sequence in which the institution believes best achieves these objectives.

5.1 THE TCO FRAMEWORK.

Course development begins with terminal course modules that contain various TCOs. Together, modules and objectives provide the backbone of the course and represent the learning expectations of the students. The course objectives can be assembled, or sequenced, at the discretion of the teaching organization.

5.1.1 Example of TCO Arrangement in a Traditional Classroom.

At the discretion of the curriculum designer, the objectives should be arranged in a time sequence and flow consistent with the requirements of the teaching institution. Figures 1 to 10 show, in a storyboard format, examples of how the TCOs can be arranged in the context of a

traditional classroom once students have gained a basic knowledge of composite materials technology. Note that the order of the objectives does not necessarily have to follow the numbering sequence used in this report. Note also that this sequence assumes that students have studied and understand the base knowledge portion of the course prior to the regular class.

Figure 1. Course Structure: Monday Morning

Figure 2. Course Structure: Monday Afternoon

Tuesday — Critical Composite Maintenance and Repair Issues

Morning 8:00 to 9:50	Primary Mode[s]: Lecture / Supplemental Mode[s]: P. Pt Presentation, Streaming Video, Testimonial from Practitioner	Module: 6.3.1 Recognize composite damage types and sources • Identify damage to composite sandwich and laminate stiffened structures • Describe damage types • Understand technical data & structure analysis necessary for design and process • Distinguish differences in repair disposition procedures for those damages covered or not covered by source documentation • Describe regulatory approval process for damages not covered by source documentation
Morning 9:50 to 10:10	Break	Safety Message 3.2 Approved Data — Total Time: 1hr 50min
Morning 10:10 to 12:00	Primary Mode[s]: Lecture, Lab / Supplemental Mode[s]: Support Technician, Student Participate, P. Pt Presentation, Testimonial from Practitioner	Module: 4.3.1 Recognize composite damage types and sources • [LAB]: Damage laminate coupons; visually inspect extent of the front and back side surface Module: 4.3.2 Describe composite damage and repair inspection procedures • Describe NDI techniques including strengths & weaknesses • Describe steps for making damage dispositions Safety Message 3.5 In-service Inspections — Total Time: 1hr 50min
Afternoon 12:00 to 1:00	Lunch	Total Time: 1 hr

Figure 3. Course Structure: Tuesday Morning

Tuesday — Critical Composite Maintenance and Repair Issues

Afternoon 1:00 to 2:50	Primary Mode[s]: Lecture, Lab / Supplemental Mode[s]: Support Technician, Student Participate, P. Pt Presentation	Module: 4.3.1 Recognize composite damage types and sources • [LAB]: Prepare bonded repair for cure, including bagging and heating apparatus and cure • Demonstrate & apply common drying & surface preparation techniques, including inspection Total Time: 1hr 50min
Afternoon 2:50 to 3:10	Break	Total Time: 20 min
Afternoon 3:10 to 5:00	Primary Mode[s]: Lecture / Supplemental Mode[s]: P. Pt Presentation, Video, Testimonial from Practitioner	Module: 4.4.1 Describe composite laminate fabrication and bonded repair methods • Understand basics of composite laminate fabrication • Understand basics of composite bonded repair • Describe processing steps used by factory, MRO & MRB • Describe processing parameters for laminate & sandwich panel fabrication • Identify typical processing defects which occur in composite laminate fabrication and bonded repair Safety Message 3.4 Correct Processing — Total Time: 1hr 50min

Figure 4. Course Structure: Tuesday Afternoon

Wednesday　　　**Critical Composite Maintenance and Repair Issues**

Morning	Primary Mode[s]:	Module 4.4.2: Perform a bonded composite repair
8:00 to 9:50	Lecture	•Describe in-process quality controls during laboratory bonded repair process trials
	Supplemental Mode[s]:	•Describe metal bond repairs & differences from composite bonded repairs
	Demonstration	
	P. Pt Presentation	Total Time: 1hr 50min

Morning		
9:50 to 10:10	Break	Total Time: 20 min

Morning	Primary Mode[s]:	Module 4.4.2: Perform a bonded composite repair
10:10 to 12:00	Lecture	•Demonstrate & apply material lay-up and compaction processes
	Lab	•Demonstrate how to prepare & cure a simple bonded repair to a laminate panel; explain the types of errors to avoid
	Supplemental Mode[s]:	
	Support Technician	
	Student Participate	Total Time: 1hr 50min

Afternoon		
12:00 to 1:00	Lunch	Total Time: 1 hr

Figure 5. Course Structure: Wednesday Morning

Wednesday　　　**Critical Composite Maintenance and Repair Issues**

Afternoon	Primary Mode[s]:	Module 4.4.2: Perform a bonded composite repair
1:00 to 2:50	Lab	•Describe process parameters affecting bonded repair quality & in-process controls necessary to avoid defects
	Supplemental Mode[s]:	•[LAB]: Prepare bonded repair for cure, including bagging and heating apparatus and cure
	Support Technician	
	P. Pt Presentation	**Safety Message 3.6** Total Time: 1hr 50min
	Student Participate	**Correct Processing**

Afternoon		
2:50 to 3:10	Break	Total Time: 20 min

Afternoon	Primary Mode[s]:	Module 4.4.2: Perform a bonded composite repair
3:10 to 5:00	Lab	•[LAB]: Prepare bonded repair for cure, including bagging and heating apparatus and cure
	Supplemental Mode[s]:	•Describe critical in-process controls necessary to avoid defects
	Support Technician	
	Student Participate	Total Time: 1hr 50min

Figure 6. Course Structure: Wednesday Afternoon

Thursday   **Critical Composite Maintenance and Repair Issues**

Time	Mode	Content
Morning 8:00 to 9:50	**Primary Mode[s]:** Lecture Lab **Supplemental Mode[s]:** Support Technician Student Participate Testimonial from Practitioner	**Module 4.3.2**: Describe composite damage and repair inspection procedures •Describe critical steps for inspecting completed bonded repair, including NDI and interpretation of results •[LAB]: Prepare bonded repair for cure, including bagging and heating apparatus and cure **Safety Message 3.7** **Post-repair Inspections** Total Time: 1hr 50min
Morning 9:50 to 10:10	Break	Total Time: 20 min
Morning 10:10 to 12:00	**Primary Mode[s]:** Lecture Lab **Supplemental Mode[s]:** Support Technician Student Participate P. Pt Presentation Video	**Module 4.3.2**: Describe composite damage and repair inspection procedures •[LAB]: Prepare bonded repair for cure, including bagging and heating apparatus and cure **Module 4.4.3**: Describe composite laminate bolted assembly and repair methods •Describe basics of composite bolted structural assembly •Describe basics of composite bolted repair Total Time: 1hr 50min
Afternoon 12:00 to 1:00	Lunch	Total Time: 1 hr

Figure 7. Course Structure: Thursday Morning

Thursday   **Critical Composite Maintenance and Repair Issues**

Time	Mode	Content
Afternoon 1:00 to 2:50	**Primary Mode[s]:** Lecture Lab **Supplemental Mode[s]:** Support Technician Student Participate Demonstration	**Module 4.4.4**: Perform & inspect a simple bolted composite repair •Demonstrate composite drilling versus metal drilling •Describe process parameters which affect bolted composite repair quality & in-process controls necessary to avoid defects Total Time: 1hr 50min
Afternoon 2:50 to 3:10	Break	Total Time: 20 min
Afternoon 3:10 to 5:00	**Primary Mode[s]:** Lab **Supplemental Mode[s]:** Support Technician Student Participate Testimonial from Practitioner	**Module 4.4.4**: Perform & inspect a simple bolted composite repair •[LAB] : Demonstrate & apply common damage removal, surface preparation, drilling & fastening techniques used for bolted repair & how to inspect them for acceptability •[LAB] : Verify correct fastener selection, inspect drilled holes, & check if fasteners were properly installed during bolted repair lab trials **Safety Message 3.8** **Bolted Repairs** Total Time: 1hr 50min

Figure 8. Course Structure: Thursday Afternoon

Friday — Critical Composite Maintenance and Repair Issues

Morning 8:00 to 9:50	**Primary Mode[s]:** Lecture **Supplemental Mode[s]:** P. Pt Presentation Support Technician Student Participate	**Module 4.2.3 Demonstrate awareness of course principles applied to composite damage disposition and repair [Case Team Studies]** Laboratory divided into teams of 3 (inspector, engineer, technician). Various damaged panels from lab in 4.3.1 (both sandwich and laminate stiffened panels) are to provided to each group for them to make full repair dispositions for each damage scenario. The students will: •Understand configuration and materials used in fabrication •Assess damage and map damage **Total Time: 1hr 50min**
9:50 to 10:10	Break	**Total Time: 20 min**
Morning 10:10 to 12:00	**Primary Mode[s]:** Lecture **Supplemental Mode[s]:** P. Pt Presentation Testimonial from Practitioner Support Technician Student Participate	**Module 4.2.3 Demonstrate awareness of course principles applied to composite damage disposition and repair [Case Team Studies]** •Utilize source documentation •Write repair procedure and QC plan based on chosen repair option •Write post repair inspection and approval plan **Safety Message 3.9 Importance of Teamwork** **Total Time: 1hr 50min**

Figure 9. Course Structure: Friday Morning

Friday — Critical Composite Maintenance and Repair Issues

Afternoon 12:00 to 1:00	Lunch	**Total Time: 1 hr**
Afternoon 1:00 to 2:00	**Primary Mode[s]:** Discussion	**Course Summary & Review** Review of all course modules & Safety Messages **Total Time: 1hr 0min**
Afternoon 2:00 to 2:20	Break	**Total Time: 20 min**
Afternoon 2:20 to 3:20	**Primary Mode[s]:** Discussion	**Student Assessment**: Students to be tested against course information **Total Time: 1hr 0min**
Afternoon 3:20 to 4:00	**Primary Mode[s]:** Discussion	**Student Questionnaire** of course value, lecturer & technician competence using the following criteria: Knowledge, Presentation, Effectiveness How well content and assessments satisfied the requirements of the module terminal course objectives **Total Time: 40min**

Figure 10. Course Structure: Friday Afternoon

5.2 CURRICULUM FORMAT.

The format of the course is at the discretion of the training organization. It may be classroom-oriented with classroom experience and laboratory practice intermingled. It may also be a mix of distance, online learning of content, concluding with a 3-day laboratory. One possible scenario is represented in figures 1-10 with the course conducted over a 5-day period. Note that the modules, TCOs (4 digit items), laboratory experiences, and safety messages are blended in an order that best suits the training organization's classroom and laboratory needs. Note again that figures 1-10 do not include the basic course.

APPENDIX A—INSTRUCTOR'S GUIDE

Since the composites technology lacks standard practices and since these practices vary among manufacturers and repair organizations, instructors have considerable latitude in selecting content to fulfill the intent of the terminal course objective (TCO). The Instructor's Guide provides additional content to support the standard course implementation, but it is optional, since it contains details that may be helpful but are not needed to meet the standard. The Instructor's Guide is not intended to be the authoritative source for populating the TCOs, but should be considered an added resource to the individual teaching materials provided by the teaching organizations.

A.1 BASE KNOWLEDGE.

The high strength-to-weight ratio properties of composite materials are especially attractive for airframe manufacturers wishing to improve performance and fuel efficiency of both military and commercial aircraft. Use of composite materials in aircraft has increased dramatically, as shown in figure A-1 (courtesy of NASA-Langley Research Center).

Figure A-1. Military Aircraft use of Composites

Use of composites in commercial aircraft has increased, although at a slower rate than for military applications, as shown in figure A-2 (courtesy of NASA-Langley Research Center). Later aircraft developments (since the year 2000) have composite structural weight percentages from approximately 25% (Airbus A380) to 50% (Boeing 787).

Figure A-2. Commercial Aircraft use of Composites

A.1.1 UNDERSTAND THE BASICS OF COMPOSITE MATERIALS TECHNOLOGY.

The principles in this course provide an introduction to composite materials technology that is necessary for understanding issues involved with maintenance and repair of composite materials. Sources for terminologies include the SAE AIR 4844 Composite and Metal Bonding Glossary and the reduced glossary in section A.1.1.6 of this report.

A.1.1.1 Distinguish Among Resin, Fiber, and Core Applications and Uses.

Common terms include carbon, glass and aramid fibers, resins and adhesives, bagging films, release films, potting compounds, honeycomb core, and foams of various types. Examples of process understanding include how vacuum systems work to apply pressure, how thermocouples measure temperature, and how hot bonders are used to control temperature, vacuum pressure, and curing times.

A.1.1.1.1 Resins.

A resin is often called a matrix when used in conjunction with reinforcing fibers, i.e., a composite consisting of the fibers and the resin. In modern composite materials applications there are many resins available and numerous products within each type. The term resins is normally used to describe relatively low-viscosity liquid materials that form the matrix of a composite when cured. The word viscosity is used to describe the degree of fluidity and flow. For example, water is a thin (low-viscosity) fluid that flows easily; honey is much thicker and exhibits high viscosity.

Volume fraction is a term that describes the volume of resin and fiber in a composite as a fraction of its total weight. Sixty percent and fifty percent fiber are typical percentages in wet lay-up systems.

Epoxy resins are most commonly used in aircraft structural components. Phenolic resins are used for internal composite parts, such as floor panels, galleys, and overhead storage bins, because phenolic resins exhibit improved flammability. Polyimide and bismaleimide resins are used in higher-service temperatures applications. Polyester, vinyl ester, and epoxy resins are used extensively in boat building and for many other uses.

Wet lay-up resins are typically two-part systems, whereby a curing agent is mixed with a base resin and the mixture is brushed onto layers of fiberglass, carbon fiber, aramid fiber, or other fibers. These liquid resins may be cured to a solid product at room temperature or at elevated temperatures to accelerate the chemical reaction required to make them cure to a solid. Resin and curing agents are individually weighed out in the correct proportions, to the nearest tenth of a gram, using electronic scales and then mixed thoroughly for at least 3 minutes. This is important in order to achieve the required strength and temperature properties of the final composite part. The required curing time and pressure are also very important. Each resin has a required minimum curing time at a specified temperature to achieve full cure.

When the required amounts of base resin and curing agent have been taken from their containers, it is important to replace the lids immediately, because moisture absorption into the curing agent degrades the material and reduces the strength of the final product. These two-part systems must be weighed accurately to ensure the correct mix ratio. Data sheets for different systems identify significantly different ratios. Some two-part systems contain dyes to color the two parts, for example, red and white or blue and yellow. When a completely uniform color is achieved with no streaks, the resin can be considered mixed.

Alternatively, the resin can be supplied already applied to the fabric or tape in a form called prepreg. It is supplied as a roll between two plastic release sheets, which must be removed before application or the layers of prepreg will not stick to each other during the curing process. The resin at this point is at the B stage and must be kept in a freezer at -18°C (0°F) until use. In this condition it has a shelf life of 6 to 12 months, depending on the type.

Prepreg also has an open-time (or out-time) limit at room temperature of about 1 month. After this period, the material must be tested for acceptance before use or be scrapped. Records must be kept of shelf life and open time for each roll. If a repair is not cured within the open-time limit of the prepreg, it must be removed and scrapped and another patch must be made using fresh material. Before use, a roll must be allowed to warm to room temperature and must not be opened until all moisture has been completely evaporated. The total time out of the freezer must be recorded on each occasion with an added 15 minutes, and when the open-time limit has been reached, the material must be tested before further use. Prepreg materials used in the fabrication of aircraft structural components normally come in two types, those that cure at 120°C (250°F) and those that cure at 180°C (350°F). Some types can be cured at 150°C (300°F).

All resins and adhesives need their specified curing time to achieve their fully cured state and mechanical properties. The cure temperature must be measured by thermocouples and must be monitored by a recording system. Matrix resins come in a wide range of chemical formulations and some can work at much higher temperatures than others. Each resin or adhesive has a glass transition temperature (T_g). This is the temperature at which the resin changes from being hard and brittle to being soft and more ductile. Designers usually work to the wet T_g, when the resin is partly saturated with water, which is significantly lower than the dry T_g. A safety factor is often added because the composite compression strength depends upon the modulus, which is related to the hardness of the resin. The maximum temperature at which a composite part can be used depends on the choice of resin and fiber but is usually limited by the resin. Carbon and glass fibers can be used at very high temperatures, but aramid and polythene fibers can be limited in maximum temperature by the fibers rather than the resins. In repair work, a major requirement is that only the (1) original equipment manufacturer (OEM), (2) Structural Repair Manual (SRM)-specified resins and fibers or, (3) OEM or Designated Engineering Representative (DER)-approved alternatives may be used. This is necessary to maintain the original design strength of the part.

A.1.1.1.2 Adhesives.

Adhesives are used to bond composite parts together and are often used for bonding metal parts, such as parts made from aluminum alloys, titanium alloys, and occasionally corrosion-resistant steel. Adhesives have a T_g, just like resins.

Because metal reinforcements are sometimes used in composite parts, there is a need to understand the adhesive bonding of metal parts as well as composites. The surface preparation of metal parts to be bonded is even more critical than the surface preparation of composite parts. A clean surface by itself is not sufficient for strong, durable bonding. Surface preparation for bonding raises the surface energy of the metal or composite to the highest value possible. The reason for this is because a resin or adhesive will only wet the surface to be bonded if it has a lower, but preferably much lower, surface energy than the surface itself. If a liquid, such as a drop of mercury, has a higher surface energy than the metal or composite surface, it will form a ball and roll around on the metal and composite surface, but will not wet or stick to it. For this reason, adhesives and resins have a fairly low surface energy and specially treated metals have a high surface energy.

Adhesives are chemically similar to composite matrix resins but are of higher viscosity to prevent the adhesive from flowing out of the joints and leaving them resin-starved. Two types of adhesives are typically used in bonding composites: paste adhesives and film adhesives.

Two-part, room-temperature curing adhesive systems are at approximately toothpaste viscosity when completely mixed. They also need to be weighed out accurately and the two parts mixed thoroughly for at least 3 minutes. Many two-part paste adhesives can be purchased as kits containing the correct amounts of part A and part B to give the required mix ratio. This avoids the need to weigh the two parts but often means more wasted material. Epoxy adhesives are the most common type used for bonding aircraft structural composite parts. The maximum service temperature of paste adhesives varies considerably with their chemical composition; therefore, it

is essential to ensure that only the OEM-approved type, or an approved equivalent, is used for each specific purpose. One-part paste adhesives are also available, but require low-temperature storage. Curing paste adhesives takes place above a minimum temperature, which depends on the chemical formulation. The full curing time must always be allowed before the part can return to service. Data sheets provide requirements for curing times, pressures, and temperatures specific to a particular adhesive. For the two-part adhesives, lids must also be immediately replaced on containers to avoid moisture absorption. One-part resins must be returned to the freezer as soon as possible after the required amount has been taken.

Adhesives can also be supplied in the B stage, or partly cured condition, known as film adhesives. Film adhesives are a thin film of adhesive between two plastic release sheets. They come in various weights per square foot or square meter and can be cured at 120°C (250°F), 150°C (300°F), or 180°C (350°F), depending on the type. These cure temperatures must be accurately maintained over the entire bondline; too low a temperature means an incomplete cure and too high a temperature may make the bondline too brittle. Each adhesive data sheet will provide the temperature limits for that particular product. Although a lower temperature may be used for a longer time for certain film adhesives, all film adhesives have a lower temperature limit, below which a full cure will not take place, as described in the data sheet for the material being used. Shelf life and out-time limits for these materials must be observed. When bonding aluminum alloy skins to aluminum alloy or aramid honeycomb, it is essential to use a heavy-weight film adhesive to produce a good size of fillet at the intersection where the honeycomb joins the skin. When bonding composite prepreg to honeycomb, the surplus resin from the prepreg may not adequately ensure good fillets to the honeycomb core. A thin layer of film adhesive should be added.

A.1.1.1.3 Fibers.

The three fiber types most commonly used in composite aircraft structures are carbon, glass, and aramid. These fiber types are easily distinguished from one another: carbon is black, glass is water clear, and aramid is yellow opaque. Quartz fiber is sometimes used for radomes, and boron fiber patches are often used to repair metal parts that exhibit fatigue cracks. All these fibers, except boron, which is too thick to be woven, can be made into fabrics that have many different weave styles or can be made into unidirectional tapes.

Carbon fibers are the most commonly employed fiber for critical structural aircraft components. Aircraft applications include wing, fuselage, stabilizers (horizontal and vertical), elevators, ailerons, main wing flaps, and rudder structures. Carbon fibers are also used in undercarriage doors, engine cowlings, helicopter rotor blades, and in undercarriage components for some helicopters. Carbon fiber can be supplied in several grades of strength, modulus, and forms. Carbon fibers are often woven into fabric (or cloth) forms for original part and repair prepregs or dry fiber mats for wet lay-up repairs.

Glass fibers have good radio frequency transmission properties and are not electrically conductive, which is ideal for radomes. Aramid and quartz fibers are used less frequently in radomes applications. Glass fibers are also used for galleys, floor panels, and overhead stowage bins. Glass fibers are available in two basic types: E glass and S glass. E-glass fiber is less

costly and by far the most commonly used, whereas S-glass fiber has better mechanical properties and is used when the additional cost can be justified.

Aramid fibers are supplied in several types. Kevlar® 49 is the most common for aircraft use and Kevlar 149 can be supplied if required. Kevlar 149 exhibits less water absorption than Kevlar 49.

A.1.1.1.4 Dry Fabric/Tape or Prepreg.

To make a strong, durable repair, it is essential to select the correct fiber type, matrix, and weight of fabric (or tape) with the right surface finish. When fabric is cut from a roll, a label that identifies the details of the fabric must be attached or included with the cut piece. For example, (1) surface finish details, once fabric is removed from the roll, can cause difficulties without proper identification, as there is no simple means of identifying the finish used on a fabric, and (2) some composite lay-ups may use fabrics and tapes of different weights at certain points in the lay-up, or layers of aramid or glass are added at special positions. If not properly identified, this can result in some plies in a lay-up not being of the same materials or the same weight.

A fiber-reinforced composite is only strong in the direction of the fiber. The angle of each ply is important to the strength and stiffness of the laminate, and each layer or ply must be laid up in the direction given on the drawing or other approved documentation. Part drawings and other approved repair documentation will contain ply tables showing the material type and lay-up direction of each ply in a part. An orientation clock (sometimes called a warp clock) is also shown on each drawing to give the lay-up direction of each ply, relative to a key direction on the part. The warp direction of a fabric is the length of the roll, and its strength is greatest in this direction. Fibers in the width direction are known as weft, fill, or woof. Although composites are very strong in the fiber direction, their transverse strength may be only 1/30 of their longitudinal strength, which must be considered when designing the repair.

Fiber properties are affected by surface preparation during a repair. Clean, dry composite surfaces, which have been carefully abraded with aluminum oxide or silicon carbide abrasive paper, should be used in such a way that only the surface resin layer is abraded without damaging the first layer of fiber. Grit size is recommended in the approved repair documentation being used. Note that for repair work, the first repair ply should be oriented in the same direction as the surface ply to which it is to be bonded.

A.1.1.1.5 Core Materials.

A considerable range of core materials exists, each serving its own specific purposes, although many are used for a range of components. The factors that determine choice are strength, upper-temperature limits, moisture absorption, and cost. Each material used to make cores has its own good and bad properties. A core material must be carefully selected to match the environment it will be used in. As an example, balsa wood can be a good core material, but only if it is not exposed to moisture. When foam cores are used in boat hull construction, they must be of the closed cell type to avoid water absorption in the event the outer skin is damaged.

Typical core materials used in composite aircraft structural sandwich parts are:

- Aramid paper honeycomb
- Glass cloth honeycomb
- Aramid cloth honeycomb
- Aluminum alloy honeycomb

Other core materials currently available are:

- Polystyrene foam (Styrofoam™)
- Polyvinyl chloride foam (cross-linked and uncross-linked versions)
- Polyurethane foam
- Polyimide foam
- Polymethacrylimide foam (Rohacell®)
- Carbon fiber cloth honeycombs

All honeycombs can be supplied in a range of cell sizes, cell shapes, and core densities. The normal hexagonal cell core does not bend easily, except when heat-formed. Aramid paper honeycomb with hexagonal core is often heat-formed by the OEM, as this can be cheaper than buying over expanded or flex core. Overexpanded honeycomb is made to bend one way and to wrap around leading edges or to make tubes. Flex core, or double flex core, cell shapes are used for nose radomes and other double curvature requirements. Only electrically nonconductive core materials may be used to repair radomes. One problem that can occur with radomes and other parts during repair is that hexagonal honeycomb, which is supplied flat, will not always take up the shape required in a radome if the honeycomb cannot be heat-formed. It may be necessary to request OEM approval to use overexpanded or flex core of equivalent strength and stiffness in these locations for repairs.

Aluminum alloy honeycomb must be anodized to ensure good environmental resistance and bond strength to the adhesive that joins it to the skin. The bondline area, end-on to the cells, is so small that good fillets of adhesive are essential to a good bond (0.5-mm or 0.020-inch minimum fillet size is recommended). For this reason only, the correct type of film adhesive must be used to bond honeycomb. If in doubt, always use a slightly heavier than specified film adhesive when bonding to honeycomb to ensure a good fillet.

A.1.1.1.6 Sandwich Construction.

Aircraft currently contain many parts that use sandwich composite construction, such as flight control surfaces, undercarriage doors, engine cowlings, fairings, and wing trailing-edge panels. Sandwich parts consist of two thin facesheets with a core material between them to provide stiffness and strength at low weight, as shown in figure A-3. Similar to a steel I-beam used in civil applications, the bending moment is resisted by tensile and compressive axial loads in the facesheets (skins) and by shear in the core. Because sandwich structure have thin facesheets they are easily damaged, but are relatively easy to repair.

Figure A-3. Sandwich Construction (courtesy Heatcon Composite Systems)

These thin-faced structures are easily damaged, but are relatively easy to repair.

Sandwich panels are at optimum design when the weight of the two skins is equal to the weight of the core material. However, in applications such as passenger and cargo floor panels in aircraft, the indentation or damage resistance may be very important to the part service life. In these applications, damage resistance can be improved at the lowest weight by using a heavier, and hence stronger, core material rather than adding to the weight of the top skin. The size of the indenting object is also important. Small indenters tend to cause core compression failure, and larger indenters cause shear failure. Therefore, core materials for these types of applications need both good compression and shear strengths to prevent damage from wheels for food carts.

A.1.1.2 Describe Various Composite Processing Parameters.

An OEM, or an approved subcontractor, makes composite components to very high standards.

Items and procedures that are considered essential to promote high-quality composite processing during production and repair of components include:

- Production-quality tooling

- All film adhesives and prepregs are stored to the manufacturer's recommendations, and records are kept to ensure that materials are within their shelf life and in good condition at the time of use.

- Staff are properly trained in the composite fabrication process.

- The workshop temperature and relative humidity are maintained within specified limits.

- The part is laid up in a clean room with a positive pressure, where temperature and humidity are also controlled.

- Some repairs may have to be performed on the aircraft. Unless it is warm and dry outside, with very little wind to blow dust around, all repairs should be made inside a hangar. It may be necessary to erect a tent or similar device around the repair area. For hot-bonded repairs, it is very important to keep the hangar doors closed during the cure cycle to prevent drafts from cooling the repair area. It may be necessary to use heaters in this area to ensure that the correct temperature and humidity are maintained for the full cure time. It should also be noted that room temperature-curing resins will take longer to cure if the outside or hangar temperatures are below 20°C. The prepared repair surface should be covered with a clean plastic sheet to avoid contamination until the patch has been applied. Every effort must be made to maintain cleanliness, temperature, and humidity, within limits, during on-aircraft repairs. Similar requirements apply in workshops. No diesel or other exhaust fumes are permitted in a bond shop. No smoking or eating is permitted. No mold release materials may be applied in a workshop. They must be applied outside the workshop and allowed to dry before tools are brought into the shop. Dust must be kept to a minimum. If solvents are used to clean a surface, they must be pure laboratory-grade solvents with minimal residues. Wiping cloths are required to be free of silicones and any other oils or grease. Wiping must continue until the wiped surface remains clean. The last wipe should be made with a clean, dry, Type I cloth to remove the last of the residue. Initial wiping may use Type II cloths.

- Schedule activities so there are no long delays between laying up the part and the cure cycle.

- When practical, an autoclave should be used to ensure the required pressure and temperature with a computer-controlled program to control the heat-up rate, the cure cycle, and the cooldown rate.

- The pressure must be monitored throughout the cure.

- Vacuum-bonded repairs usually use a hot-bonding controller and heater blanket to control the temperature and monitor the vacuum pressure and cure time.

A.1.1.2.1 Repair Processing.

Making a repair to a composite part is a difficult process for the airline or repair station because there is a lack of information about the construction of the part to be repaired. Because of this, OEMs have concerns about the quality of repairs made at remote stations around the world. When composite repairs are performed by non-OEMs, it is essential that the repair materials, repair processes, and the repair personnel are fully approved and qualified.

Principal processing parameters include the following:

- The workshop must be clean, dry, and at the correct temperature and humidity. The recommended conditions are a clean room with positive air pressure to minimize the entry of dust and dirt, a room temperature of 18°C (64°F) to 30°C (86°F), and a maximum relative humidity of 60% (35% relative humidity is preferred). Tests have

shown that relative humidity below 35% does not lead to any significant improvement in results. Good lighting should be provided.

- Any required tooling should be clean and in good condition and must already be in the clean room.

- The component must be clean and dry and the surface must be correctly prepared for bonding the repair patch. Check the moisture content with a moisture meter (thermographic-type preferred) if the component is fabricated from fiberglass or aramid fibers. Dry the surface to approved repair documentation requirements.

- The required repair materials must be available and in the correct condition (i.e., dry and warmed to room temperature (no heating is permitted) and within their specified shelf lives and out-times.

- Each ply of composite must be the correct material type, weave and weight, and laid up in the right place and in the right direction. All release films must be removed and discarded in one pile, and then checked to be sure all the release films have been removed before proceeding with the bagging process. One way of doing this is to remove one release film during lay-up, lay the ply and then roll the ply down before removing the top release film for each layer.

- If new honeycomb is used in a repair, it must be of the right type, weight, and cell size; have the specified finish; and be undamaged. It must also be oriented as indicated in the drawing and in line with the core being repaired. It must be bonded to the skin with the specified adhesive and bonded to the existing core with potting compound or foaming adhesive. If other core materials such as foams are used, they must be of the right type and weight, and they must be joined to the existing core with the adhesive specified in the approved repair documentation.

- If vacuum pressure is used, the bagging film, sealant, and the breather cloth lay-up, together with all release films (perforated and nonperforated) must be correct and in the required positions. All the materials must be clean, dry, and in good condition.

- Vacuum pressure must be correct and the leakage rate check requirement must be met.

- To avoid degradation of film adhesives, prepregs, and wet lay-up resins at room temperature, bonding should commence as soon as possible after the materials have been laid up. Some large jobs may take 3 days to lay up, which is a significant amount of the permitted open time for a prepreg if the material is anywhere near the end if its shelf life. For such work, the remaining open time for the material should be checked before the lay-up starts. The specific prepreg material working life is provided in the supplier data sheet.

- The temperature readings from all the thermocouples must be within the required limits and maintained correctly for the specified length of time to achieve full cure. Localized heating of cool spots may be needed to maintain a uniform temperature.

- Vacuum pressure must be maintained during the entire cure time and until the component temperature has fallen to 50°C (122°F) or less.

- After curing, the release films and peel plies must be carefully removed to avoid damage.

Other important processing considerations include the following:

- Some prepregs need a perforated release sheet to allow volatiles to escape and to bleed off excess resin. For example, a sample did not cure using a nonperforated release sheet on both sides, but it did cure when one nonperforated release sheet was replaced with a perforated sheet.

- The breather cloth must be placed under the vacuum extract fitting to ensure good airflow and to prevent resin from going into the vacuum line. If resin gets into the exhaust fitting, it may block it and, when cured, solid resin is difficult to remove.

- Do not locate the vacuum extract fitting too close to the part, because it may draw resin into itself.

- Do not place the vacuum port fitting on the part because it will leave a permanent mark.

- Tools (molds) must be absolutely clean and smooth; otherwise, all defects will be impressed on the part.

- The release sheet should extend well beyond the end of the part to avoid resin sticking to the mold face.

- Composite laminates will not bond together properly at zero pressure. Just rolling the plies together, even with good roller pressure, will not be sufficient. This is especially true when time-expired material is being used for training purposes. A good vacuum is essential for a good bond and to extract volatiles.

A.1.1.2.2 Thermocouples.

AIR 4844 defines thermocouples as "A device which uses a circuit of two wires of dissimilar metals or alloys, the two junctions of which are at different temperatures." Thermocouples must be correctly located. A net electromotive force (emf) occurs as a result of this temperature difference. Using a number of thermocouples, six or more for large repairs, will assist in detecting differential temperatures throughout the part. As always, consult the approved repair documentation for the number required for each task. Thermocouples are often accurate to about ±5 degrees. To ensure a good cure, add 5 degrees to the standard cure temperature for the repair. Use the correct thermocouple, as several types are available and their calibration requirements are very different. In addition, a hot bonder will only work with the thermocouple for which it was designed. In special laboratory cases, where great accuracy was required, it was common practice many years ago to place one junction (the cold junction) in melting ice and the other at the point of temperature measurement. In portable equipment, such as hot bonders, a cold junction compensation circuit is used, and only the hot junction employed for temperature

measurement is visible to the user. A net electromotive force (emf), or current, occurs as a result of the temperature difference between the two junctions. The minute emf, or current voltage, is sufficient to activate a galvanometer or potentiometer, which are calibrated in terms of temperature. The most common thermocouple types, listed in order of increasing cost, are iron/constantan, chromel/alumel, and platinum/platinum rhodium alloy.

A.1.1.3 Describe Composite Material Properties and Effects of Processing.

Material properties include strength, stiffness, impact resistance, fatigue resistance, and creep resistance, as a function of temperature relative thermal expansion coefficients between different layers of fabric or tape and between the component and the tools used.

Metals have almost the same strength in all three directions although even metals have lower transverse properties after being rolled to thickness. Composites are strong only in the directions of the fibers. Their transverse strength is very low, about 1/30 of the strength in the fiber direction. This is a similar ratio to wood, which is also strong along the grain but not across the grain. Metals, except for very low-grade castings, do not have voids, but composites can contain voids, depending on the quality of the processing. It is important to be aware of these points when working with composites.

Another factor, especially important when using carbon fiber in conjunction with aluminum, is the need for corrosion protection at the assembly stage. A layer of fiberglass must be used between the carbon fiber and aluminum as well as applying any other corrosion protection specified in approved repair documentation.

Some composite properties that are unique to composites are listed below.

- Strength in the fiber direction depends almost entirely on the fibers.

- Stiffness in the fiber direction depends almost entirely on the fibers.

- Impact resistance depends on the toughness of the fibers and the strain-to-failure of the resin matrix, as well as the strength of the matrix to fiber bond.

- Composite compression strength depends on the fiber and matrix resin modulus and matrix void content.

- Interlaminar shear strength, or short-beam shear strength, depends on resin matrix properties and void content (ASTM D 2344). Void content is sometimes called porosity. While the dictionary defines porosity as "allowing liquid to soak through," void content is a better description, as voids in a resin are closed cell. Voids reduce composite compression strength and other properties, and must be kept to a minimum. Debulking before cure is important. Vacuum bonded repairs may need extra material to compensate for a probable higher void content.

- Fatigue resistance depends on the relative fatigue strain-to-failure of the fiber and resin. Some work has shown that if the fiber has a fatigue strain-to-failure higher than the resin,

then the resin will fail first and laminate failure will be progressive. This work has also shown that if the resin fatigue strain-to-failure is higher than the fiber, then the fatigue properties become fiber-dependent.

- Fiber-to-resin interface depends strongly on the fiber finish.

- The maximum operating temperature of a composite is defined by the maximum operating temperature of the resin matrix. This is related to the T_g, which often depends on the cure temperature. Therefore, the correct resin must be used, and it must be cured at the correct temperature for the required time.

- Creep resistance depends on the creep characteristics of the fiber.

- Temperature cycling can have a significant effect on a composite part. For example, if aramid and carbon fibers are mixed in a laminate and the component suffers a wide range of temperature cycling, e.g., in an engine cowling, disbonding may occur due to the stresses imposed by the differential thermal expansion and contraction, because aramid and carbon fibers have different characteristics. Delaminations have been found to occur over a period of time in engine cowlings of this design. Cracking, due to linear expansion differences, has been reported and must be considered in the design.

- The resin must be fully cured. If the resin matrix is not fully cured, the performance of the composite may be seriously affected. The resin matrix may have a lower modulus than required, and the compression strength could be reduced. The moisture absorption of the matrix could increase. The T_g could be lower than the specification requires.

- The plies must be debulked prior to cure. If the plies are not debulked sufficiently to remove air before heat is applied, and the vacuum is not sufficient to remove any solvent vapors that outgas from the resin during cure, then voids may be present in the cured product. These can result in reduced compression strength. The correct debulking procedure must be followed, and the required vacuum or autoclave pressure must be maintained for the full cure time. The correct perforated release films and breather cloths must also be used.

- For radomes, paint thickness can affect transmission properties; therefore, paint thickness must be kept within specified limits. If radomes are repainted, the original paint must be removed to ensure that this requirement is met.

A.1.1.4 Describe Various Composite Machining, Assembly, and Finishing Processes.

A.1.1.4.1 Composite Machining.

Drilling and machining of composites requires different techniques than metals. Different drills and cutters and different cutting speeds and feed rates are required, and cooling is very important. Approved source documentation must be consulted for the correct tools and procedures. Cutting tools must be sharp, and cutting speeds and feed rates must be correct.

Use of a backing plate of wood, plywood, or plastic can avoid breakout on the rear face. For example, placing a piece of wood or plywood in a vice and pushing a twist drill through it at a high feed rate will result in large splinters.

Position a composite so that the fibers are cut or sanded in tension. This is especially important with aramid fibers. Aramid fibers are softer and need a sharper cutting tool. Tools that were previously used to cut glass or carbon fibers will generally not be sharp enough to cut aramid fibers. Since carbon fiber wears out steel tools, frequent sharpening of all cutting tools is necessary.

A.1.1.4.2 Routers.

Routing is a very coarse cutting system. Only sharp routing tools should be used with the support of templates.

High-speed diamond grit-coated band saws are good for cutting composites, but require backing support. Dust extraction must be provided on all power tools, and all dust produced by hand tools should be removed with a vacuum cleaner as soon as possible. Never use a pressurized air line to blow dust away, as this only adds to the particle concentration in the air and is an unacceptable health hazard.

A.1.1.4.3 Cutting Honeycomb Cores.

Aluminum alloy, aramid, and other honeycomb materials used as honeycomb can be supplied in sheets cut to the required thickness. A sharp knife, e.g., Stanley® knife or a razor blade, can be used for trimming, but cutting to thickness from a block requires a high-speed band saw with a blade speed of 5,000 meters (16,000 ft) per minute. To avoid tearing the honeycomb, the blade teeth should be offset by 0.05 mm (0.002 inch).

For sandwich panel repairs, it is common practice to cut a piece of honeycomb to the required shape to fit a hole slightly thicker than the final thickness. Apply a generous amount of a suitable adhesive to the bottom skin to ensure good fillet, then "pot" the honeycomb in place around the edges with a suitable potting compound to join it to the existing honeycomb.

Once the adhesives have cured, the skin should be protected, before the honeycomb core, which projects above the skin, can be carefully sanded flush with the skin.

A.1.1.4.4 Cutting Foam Cores.

Some core materials can be shaped and cut with hot wire cutters; other materials, such as polyurethane foams, give off dangerous fumes and using hot wire cutters is not permitted. Check with the material data sheet for the correct and safe cutting method.

A.1.1.4.5 Water Jet Cutting.

This cutting method uses high-pressure water that contains an abrasive grit, and it is a very good production-line system when a large number of parts need to be cut. It can also be used for single cuts. The machinery is very expensive and needs good safety systems because the water jet can cut through a finger or an arm quickly. It has the advantage that accurate profiles can be cut quickly without increase in temperature so no damage is done to the matrix resin during the cutting process. It also washes away the machining dust. Other cutting and sawing processes, described below, generate considerable frictional heat. This causes unwanted fumes during cutting. Extraction systems should be used to remove the vapors and dust.

A.1.1.4.6 Oscillating Saws (Cast Cutters).

This method uses a circular saw blade that oscillates at a very small angle. If the blade touches the skin, the skin is flexible enough to allow it to move the same amount as the blade, therefore, no cutting action takes place. When the same saw is used on a rigid composite, it will cut the material because it is rigid compared to skin. This technique is often used to cut out the damaged sections of skin from a sandwich panel with honeycomb or other core prior to removing the damaged core. The following recommendations are made to ensure safety and quality:

- Use the correct blades.
- A dull (worn) blade may cause injury and should not be used.
- Tighten the blade-retaining nut before each use.
- Clamp the composite to minimize vibration and to improve the quality of cut.
- Too much pressure may cause the blade to break.
- Clean the blade before each cutting operation.
- Do not use near solvents or other flammable fluids.

A.1.1.4.7 Tank Cutters.

Tank cutters are similar to circular hacksaw blades used by plumbers to cut holes in water tanks and are used to cut large radii at the corners of holes in damaged sandwich panel skins. When cutting out sections of skin to remove impact or disbond damage, square corners should not be used as they can induce fatigue cracks at a later stage. A 25-mm (1-inch) or larger corner radii should always be used for skin repairs.

A.1.1.4.8 Grinding Burrs.

Grinders come in a wide range of shapes and sizes and are used to smooth the edges of cutouts in skins and to blend the sides with the corners produced by tank cutters.

A.1.1.4.9 Hand Power Drills and Other Tools.

A high cutting speed and a low feed rate is recommended. The temperature of the part that is drilled must not exceed 60°C (140°F) during machining of any kind.

Power tools should be air-powered and not electrically powered, since flammable solvents are used to clean off excess resin or adhesive from parts, tools, and molds. Carbon dust can also short out electrical equipment. Fuel or fuel vapor may also be a problem when working in proximity to an aircraft. Pneumatic tools with rear exhausts are recommended and are beneficial for two reasons: (1) air is directed away from the work and dust is not blown around the shop, and (2) venturi-type vacuum attachments are used for dust extraction. This applies to high-speed grinders and all other air-powered tools. Carbide or cobalt-tipped spade bits are good, but diamond polycrystalline-coated tools are better. For carbon fiber, one manufacturer recommends drilling dry, if possible.

When lubricants are necessary, filtered air, CO_2, non-oil-containing Freon® or BOELUBE®[1] are suitable media. It is important not to exceed the T_g of the resin during drilling.

A.1.1.4.10 Hole Drilling.

One of the problems of structural repairs with mechanical fasteners is ensuring the accurate alignment of holes and close-tolerance holes. When mechanical fasteners are used that require either a close-tolerance hole or a small degree of interference fit, a very accurate hole needs to be made with the proper drilling tool. A jig may be needed to ensure accurate hole alignment and hole size.

Hole tolerances for mechanically fastening composite parts are not the same as those for fastening metal parts. The appropriate authoritative documentation should be consulted. When drilling composites, using a backing block can avoid splitting the laminate on the back face.

A.1.1.4.11 High-Speed Grinders.

High-speed grinders are used for light sanding, feather edging, and cutting. They must be small and light enough to be handled easily. A high skill level is required to achieve good results. Use the correct disc with the right grit type and size. Generally, 120 grit should be coarse enough to remove paint and 240-320 grit is suitable for preparing surfaces for adhesive bonding. The objective is to lightly abrade the resin at the surface without cutting into any of the surface fibers.

A.1.1.4.12 Orbital Sanders.

Orbital sanders are normally air-powered and should be used carefully with the correct grit type and size. When the first piece of grit paper becomes worn-out or clogged, the grit paper should be replaced. While it is important not to remove too much material too quickly, using a grit size that is too coarse, after time, becomes a finer grade because grit on worn paper slowly breaks up into a smaller grit size.

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[1] BOELUBE is a registered trademark of The Boeing Company. These products represent a family of proprietary lubricants developed through Boeing manufacturing operations and lubricant experience. The Orelube Corporation holds an exclusive worldwide license from The Boeing Company to manufacture and market the BOELUBE series of lubricants.

A.1.1.4.13 Abrasive Papers and Grits.

Too fine a grit size will cause frictional heat and a slow rate of material removal. However, too large a grit size will cause deep scratches and may remove more material than intended, which could lead to a larger repair. Choosing the optimum grit size and type is essential.

Aluminum oxide and silicon carbide are the most commonly used grits. Suitable grades of 3M Scotch-Brite® abrasive pads may also be recommended in approved repair documentation.

A.1.1.4.14 Diamond-Tipped Wheel Trimmers.

Diamond-tipped wheel trimmers are used mainly for cutting panel edges to a specific shape. They are designed to cut quickly and to give a good edge finish.

A.1.1.4.15 Lubrication and Cooling During Machining Processes.

Machining composites requires avoiding heating the composite to the point where the T_g of the resin is exceeded. Drilling above the resin T_g may cause clogging. As a guideline, the composite should not be heated above 60°C (140°F) to minimize this risk.

Composite parts do not conduct heat away from drilling, and keeping the drill bit cool is a major concern. Air cooling, using clean, filtered air, CO_2 gas, or non-oil-containing Freon, may be used.

Approved lubricants are alcohol lubricants from the BOELUBE family. These lubricants will not ingress the fibers or resins, will not cause outgassing in honeycomb structures, will not contaminate the adhesive or bonding agents, and are removed easily with alcohol solvents or a mild-detergent rinse. The BOELUBE lubricants are based on cetyl alcohol, also known as Hexadecanol or n-Hexadecyl alcohol. This waxy solid has a melting point of 49°C (120°F), which is a good indicator of the temperature around the drill and helps to avoid exceeding 60°C (140°F).

The wrong cutting speed or feed rate can cause heat or mechanical damage or delamination of composite parts. As a general rule, a high-speed, slow-feed technique is preferred.

A.1.1.4.16 Assembly Processes.

Composite components can be

* bonded together using suitable film or paste adhesives.
* mechanically fastened using a range of special fasteners.

Quite often a combination of these is used, such as bonding in some areas and fastening in others, or joints can be both bolted and bonded to provide extra security and fatigue resistance.

Complex assembly jigs may be required to ensure the precise location of critical parts, such as hinges in undercarriage doors or attachment lugs for empennage components. Parts that may need frequent removal for inspection or maintenance may need very accurate jigging so that any replacement parts will fit.

A.1.1.4.17 Finishing Processes.

The quality of a required finish often depends on whether or not the part is visible by passengers. A smooth finish may also be needed to ensure smooth airflow over wing or empennage leading-edge surfaces. Special adhesive or resin finishing films are sometimes applied when a particularly good finish is required. The quality of the surface finish on a mold used to make a part is also critical, as the slightest mark on the mold will be impressed in the cured part. Sometimes fillers can be used and the surface rubbed down by hand to remove minor marks.

Primers and paints are used to protect composite parts from environmental effects such as moisture and ultraviolet rays.

Coatings are used for a range of purposes, and if part of a coating has to be removed for any purpose, it must be completely replaced.

Conductive coatings are added to give protection from lightning strikes. Some examples are metal foils, flame-sprayed aluminum, metal-coated fabrics, or expanded foil mesh. When repairing composites, it is important to remember that any such coatings must be replaced and their electrical conductivity checked as part of the repair process.

Erosion-resistant coatings are used on radomes, wing leading edges, empennage leading edges and the leading edges of helicopter rotor blades. These coatings come in the form of neoprene or polyurethane rubber boots for radomes, erosion-resistant paints, or titanium or corrosion-resistant steel sheets formed to the shape required.

Some interior panels may be coated with DuPont® Tedlar® or polyvinyl fluoride (PVF) to keep moisture out and provide an easily cleaned surface.

Speed tape or a thin coat of sealant may provide temporary protection for an allowable damage that will be repaired in the depot.

A.1.1.5 Describe Stiffened Laminate and Sandwich Applications and Structural Properties.

A.1.1.5.1 Stiffened Laminate Applications.

Most manufacturers are moving towards using stiffened laminate composites; therefore, repair technicians must be familiar with this construction. Examples of stiffened laminate applications are wing and empennage main torque box skins and spars, and fuselage skins, stringers, and frames. These components have skins that are thicker than the skins in sandwich panels. Fuselage skin laminates are fairly thick in places where bending or torsion loads are high. The wing skins themselves are stiffened with composite stringers, which may be bonded, bolted, or

attached by both methods. Wing skin profile is maintained and air loads are transferred to the spars by ribs, which are attached to the skins and to the spars at each end. The wing skins use a ply lay-up designed to give good torsion stiffness to the wings as well as good bending strength. The lower wing skin takes tension loads in flight and some compression load on the ground, whereas the upper skin is in compression during flight but carries some tension loading on the ground and more in the case of a heavy landing. Twisting (torsion) loads are applied to the wing by engine thrust, reverse thrust, and by the use of ailerons and spoilers. Similarly, strenuous use of rudder control by the pilot, especially when compensating for engine failure on one side, will apply severe torsion loads to the fuselage.

A.1.1.5.2 Sandwich Structure Applications.

Sandwich structures are much lighter, frequently using skins of only two or three layers of a fairly thin carbon, glass, or aramid fabric. They may use aramid honeycomb, aluminum alloy honeycomb, or various types of foam or balsa wood as core materials.

Usually, but not always, aluminum honeycomb is used in panels with skins of aluminum alloy, whereas composite-skinned panels usually have aramid or glass honeycomb or foam cores.

There are many applications of sandwich structures; some of which are ailerons, elevators, rudders, floor panels, wing trailing-edge panels, wing-to-body fairings, flap screw jack fairings, engine cowlings, undercarriage doors, radomes, galleys, overhead stowage bins, etc. Sandwich structures are used because they result in light and stiff structures. In many applications, great strength is not required, but stiffness is needed to maintain shape. Most of these items are readily removable and can be replaced with a spare part during the repair, reducing delays in returning aircraft to service. For bonded repairs, there are no instruments that can measure bond strength. Nondestructive inspection (NDI) methods can often detect disbonds but not the strength of a bond. Strict adherence to all repair procedures is the only way to ensure high bond strength.

A.1.1.6 Glossary of Terms.

For a complete and updated list of terms used in composite and metal bonding, see the latest issue of SAE AIR 4844.

Adhesive: A substance capable of holding two materials together by surface attachment. Adhesive can be in film, liquid, or paste form. In this context, the term is used to denote structural adhesives, i.e., those which create attachments capable of transmitting significant structural loads.

Adhesion: The state in which two surfaces are held together at an interface by mechanical or chemical forces or interlocking action or both.

Aramid fiber: A fiber made from a type of highly oriented organic material derived from polyamide (nylon) but incorporating an aromatic ring structure.

"A" stage: This is the initial state of the resin, as produced by the manufacturer. It is an early stage in the polymerization reaction of certain thermosetting resins (especially phenolics) in which the material is linear in structure, soluble in some liquids, and fusible. The A stage is usually considered a point at which little or no reaction has occurred. Prepreg in an A-stage condition would be extremely sticky, lumpy, and have little integrity. This is also called resole.

"B" stage: The B stage is an intermediate stage in the reaction of certain thermosetting resins in which the material softens when heated and becomes plastic and fusible, but may not entirely dissolve or fuse. This helps to facilitate handling and processing. This is also called resitol. The resin in an uncured prepreg is in this stage. These resins require refrigerated storage at -18°C (0°F) to prevent gradual cure to a point at which they become unusable. They have a shelf life that varies from one product to another, but is usually 1 year. They also have a limited out-time at room temperature, which varies from 1 week to 1 month.

Bagging film: An impervious plastic film, such as nylon or PVF, that covers the repaired area or completely envelops the entire assembly and is sealed at the edges so a vacuum can be drawn.

Bismaleimide: A type of polyimide that cures by an addition, rather than by a condensation reaction, thus avoiding problems with volatiles formation, which is produced by a vinyl-type polymerization of a prepolymer terminated with two maleimide groups. Intermediate in temperature capability between epoxy and polyimide.

"C" stage: This is the final stage in the reaction of certain thermosetting resins, in which the material is practically insoluble and infusible. It is sometimes referred to as resite. The resin in a fully cured thermoset molding is in this stage.

Carbon fiber: Fiber produced by the pyrolysis of organic precursor fibers, such as rayon, polyacrylonitrile, and pitch in an inert environment. The term is often used interchangeably with the term graphite; however, carbon fibers and graphite fibers differ. The basic differences are in the temperature at which the fibers are made and heat-treated and in the amount of elemental carbon produced. Carbon fibers are typically carbonized in the region of 1315°C (2400°F) and assay at 93%-95% carbon, while graphite fibers are graphitized at 1900°-2480°C (3450°-4500°F) and assay at more than 99% elemental carbon. Also see pyrolysis (of fibers).

Cell: (1) In honeycomb core, a cell is a single honeycomb unit, usually hexagonal in shape. (2) In cellular plastic, a single small cavity surrounded partially or completely by its walls.

Composite material: A combination of two or more materials (reinforcing elements, fillers, and composite matrix binder) differing in form or composition on a macroscale. The constituents retain their identities; i.e., they do not dissolve or merge completely into one another, although they act in concert. Normally the components can be physically identified and exhibit an interface between one another.

Cross-linking: Applied to polymer molecules, the setting up of chemical links between the molecular chains. When extensive, as in most thermosetting resins, cross-linking makes one super molecule of all the chains.

Cure: To irreversibly change the properties of a thermosetting resin by chemical reaction; i.e., condensation, ring opening, or addition. Cure may be accomplished by adding curing agents (cross-linking), with or without heat or pressure. The state of an epoxy material is permanently changed from the B stage to the C stage by a controlled action of heat and pressure, see ASTM D 907.

Debulking: Compacting of a thick laminate under moderate heat and pressure, i.e., noncuring conditions or vacuum to remove most of the air, to ensure seating on the tool and to prevent wrinkles.

Epoxy resin: A polymerizable thermoset polymer containing one or more epoxide groups and curable by reaction with amines, alcohols, phenols, carboxylic acids, acid anhydrides, and mercaptans. An important matrix resin in composites and structural adhesives.

Fabric: A material made of woven fibers or filaments.

Fiber: A general term used to describe filamentary materials. Often, fiber is used synonymously with filament. It is a general term for a filament with a finite length that is at least 100 times its diameter, which is typically 0.10 to 0.13 mm (0.004 to 0.005 in.). In most cases, it is prepared by drawing from a molten bath, spinning, or deposition on a substrate. Fibers can be continuous or specific short lengths (discontinuous), normally no less than 3.2 mm (0.125 in.).

Filament: The smallest unit of a fibrous material. The basic units formed during drawing and spinning, which are gathered into strands of fiber for use in composites. Filaments are usually of large length and very small diameter, usually less than 25 micro meters (1 mil) or (0.001 in.). Normally, filaments are not used individually. Some textile filaments can function as a yarn when they are of sufficient strength and flexibility.

Film adhesive: A synthetic resin adhesive, usually of the thermosetting type, in the form of a dry film of resin with or without a paper, glass, or other carrier.

Foaming film adhesive: An adhesive film used to join honeycomb core in bonded assemblies. Contains a foaming agent that produces an expansion ratio, usually between 2 and 3, during the cure.

Finish: Chemical finish (coating) applied to glass fibers to facilitate resin wetting, resin bonding, and good environmental performance of a cured laminate. Also improves fabric handling. A diluted epoxy finish is sometimes used on carbon fiber before prepregging. A number of finishes are used on glass-fiber fabrics and tapes, depending on the resin system to be used. To ensure a good, durable bond, it is absolutely essential that the glass to be used has been treated with a finish compatible with the chosen resin system, i.e., polyester, epoxy, phenolic, or other resin. Tables of resins and suitable glass finishes are supplied by glass-fiber manufacturers.

Foam core: Core made from expanded or foamed plastic. A plastic core in which the density is reduced by the presence of numerous small cavities (cells), interconnecting or not, dispersed throughout the mass.

Gel point: The stage at which a liquid begins to exhibit pseudoelastic properties. This stage may be conveniently observed from the inflection point on a viscosity time plot. The point in a cure beyond which the material will no longer flow without breaking down the matrix network formed to that point. The point at which the matrix transitions from a fluid to a solid occurs.

Glass fiber: A fiber spun from an inorganic product of fusion that has cooled to a rigid condition without crystallizing. A glass filament that has been cut to a measurable length. Staple fibers of relatively short length are suitable for spinning into yarn.

Glass transition temperature, T_g: The approximate midpoint of the temperature range over which the glass transition takes place. It is the point at which rigid behavior changes to rubbery behavior. It is also the inflection point on a plot of modulus versus temperature. The measured value of T_g depends, to some extent, on the method of test.

Hot bonder: An instrument that supplies power to heater blankets and provides a vacuum source and thermocouples for controlling the cure cycle of adhesives, resins, and prepregs for repair work.

Honeycomb: Manufactured product of resin-impregnated sheet material (paper, glass fabric, carbon fabric, or aramid fabric) or metal foil, formed into hexagonal-shaped cells. Other cell shapes are also produced. Used as a core material in sandwich constructions.

Interweave: Weave together, interlace.

Modulus of elasticity: The ratio of the stress applied to the strain or deformation produced in a material that is elastically deformed. Also called Young's Modulus.

Mold release agent: A lubricant, liquid, or powder (often silicone oils or waxes) used to prevent the part from sticking to the mold or tool.

Phenolic resin: A thermosetting resin produced by the condensation of an aromatic alcohol with an aldehyde, particularly of phenol with formaldehyde. Used in high-temperature applications with various fillers and reinforcements. Used for aircraft interior components because, in a fire, it gives off less smoke and toxic fumes than epoxy or polyester resins.

Polyester resin: A family of resins produced by the reaction of dibasic acids with dihydric alcohols. Polyethylene terephthalate is a thermoplastic that can be extruded, injection molded, or blow molded. Unsaturated polyesters are thermosets and are used in the reinforced plastics industry for applications such as boats and auto parts. Modifications with multifunctional acids and bases and some unsaturated reactants permit cross-linking to thermosetting resins. Polyesters modified with fatty acids are called alkyds.

Polyimide (PI): A polymer produced by reacting an aromatic dianhydride with an aromatic diamine. It is a highly heat-resistant resin, 315°C (600°F). Similar to a polyamide, differing only in the number of hydrogen molecules contained in the groupings. Suitable for use as a binder or an adhesive. May be either thermoplastic or thermoset.

Polymerization: A chemical reaction in which the molecules of a monomer are linked together to form large molecules whose molecular weight is a multiple of the original substance. When two or more monomers are involved, the process is called copolymerization. See ASTM D 907.

Potting compound: A resin that has been thickened by using a filler (e.g., milled fibers or fine metal shavings). Also, a resin reduced in density by the addition of hollow glass or phenolic micro spheres. Used for jointing honeycomb and edge-filling sandwich panels.

Prepreg: Abbreviation for a pre-impregnated cloth that contains resin in the B stage for the manufacture or repair of composite components.

PVF: Polyvinyl fluoride.

Ream: To use a reamer in a hole, already near to size, to produce a close-tolerance hole with a smooth finish.

Reamer: A splined cutting tool, like a drill, used to finish a hole to a close tolerance.

Relative humidity (RH): The ratio of the actual pressure of existing water vapor to the maximum possible (saturation) pressure of water vapor in the atmosphere at the same temperature, expressed as a percentage.

Release film: (1) An impermeable layer of film that does not bond to the resin being cured. (2) Release film is a permeable or impermeable layer of film that does not bond to the resin being cured. The most common films available are modified halohydrocarbons, PVF, coated fiberglass cloth (porous and nonporous), polyester, and fluorinated ethylene propylene (FEP). FEP films are used as release films for all types of epoxy resin and are light weight and easily conformable. Release films are interleaved between any adhesive film, resin, potting compound, or sealant and a surface not intended to be bonded.

Resin: A solid or pseudo-solid organic material, usually of high molecular weight, that exhibits a tendency to flow when subjected to stress. It usually has a softening or melting range and fractures conchoidally. Most resins are polymers. In reinforced plastics, the material used to bond together the reinforcement material, the matrix. Also see International Organization for Standards (ISO) 472 and ASTM D 907.

Solvent: A substance (usually a liquid) used for dissolving or cleaning materials during reinforced plastics operations. Often flammable or toxic. Should be handled in accordance with safety instructions. Methyl isobutyl ketone (MIBK) is a ketone that is widely used as a solvent. Unlike the other common ketone solvents, such as acetone, it has a quite low solubility in water.

Stress: Load per unit area, e.g., N/mm^2, lb/in^2

Strain: Deformation due to stress. Measured as the change in length per unit length in a given direction and expressed as a percentage or mm/mm or in./in.

Surface tension: The contractive force in the surface film of a liquid that tends to make the liquid occupy the least possible volume. It is due to the tendency of the body of liquid to attract the unbalanced surface molecules towards the interior. It is expressed in milli-Joules per meter squared (mJ/m^2) and varies for different liquids, being very high for mercury and very low for ether. It decreases with increasing temperature.

Surface energy: Solid materials have a surface energy similar to surface tension, which depends on the activity of the surface. It is also measured in mJ/m^2. Contamination or oxidation can reduce it considerably. Some freshly produced oxides, such as aluminum oxide, provide a good bonding surface. The purpose of surface treatment of metals and other solids before bonding is to raise this surface energy as high as possible so the adhesive will wet and bond well to the surface. There are two groups of solids:

- Low-energy surfaces with values below about 50 mJ/m^2, usually organic low-melting point materials.

- High-energy surfaces with values greater than 200 mJ/m^2, usually inorganic high-melting point materials.

Tape: Unidirectional prepreg fabricated in widths up to 305 mm (12 inches) for carbon and 75 mm (3 inches) for boron. Dry-cloth tapes are available in various fabrics and are usually less than 100 mm (4 inches) in width. Tapes also have woven selvages, which are usually left in the work. Unidirectional material is called tape regardless of width.

Thermocouple: A device that uses a circuit of two wires of dissimilar metals or alloys, the two junctions of which are at different temperatures. A net emf occurs as a result of this temperature difference. The minute emf or current is sufficient to drive a galvanometer or potentiometer.

Thermoplastic: A substance capable of being repeatedly softened by an increase in temperature and hardened by a decrease in temperature. The change in this material due to heating is physical rather than chemical. This material in its softened stage can be shaped by flow into molds or extrusions to form parts. See ISO 472 and ASTM D 907.

Thermoset: A plastic that when cured by application of heat or chemical means, changes into a substantially infusible and insoluble material. Once cured, a thermoset cannot be returned to the uncured state.

Ultrasonic (UT): Pertaining to mechanical vibrations having a frequency greater than approximately 20,000 Hz.

Vacuum system: The system consists of a pump to remove air, which reduces the air pressure inside a vacuum bag, some piping, and a vacuum gauge. A vacuum gauge is needed to measure the degree of vacuum achieved. Removal of the air inside the bag means that a large proportion of the atmospheric pressure can be used to provide pressure to hold together the parts being bonded during the cure cycle.

Vinyl ester: These resins are becoming more important in boat building. Although they are more expensive, their high quality has resulted in their consumption being almost equal to polyesters. They offer good impact and fatigue resistance, and they make a good permeation barrier to resist blistering in marine laminates. Vinyl esters fall between polyesters and epoxies in both cost and performance.

Viscosity: A viscous liquid is one that does not flow easily compared to water. Viscosity is measured in Pascal seconds in a standard test. (See ASTM D 907.)

Voids: Air, water, vapor, or gas that has been trapped and cured into a laminate or adhesive bondline. Voids are often caused by air entrapment during the laying down of the adhesive or prepreg, the presence of moisture, or the use of insufficient bonding pressure. Voids are essentially incapable of transmitting structural stresses or nonradiative energy fields. They are just small empty spaces that cannot support the fibers or transmit loads. The term porosity is sometimes used to describe voids, but porosity means the ability to allow a liquid to flow through. Voids cannot do this unless they go from one side of a material to the other.

Warp direction: This is the direction of the length of a roll of cloth or prepreg and is the direction of greatest strength in a fabric.

Weave: The particular manner in which a fabric is formed by interlacing yarns. Usually assigned a style number.

Weft direction: This is the transverse direction in a cloth, which does not always have as many fibers per unit length as the warp direction. Sometimes called the fill or woof direction.

Yarn: An assemblage of twisted filaments, fibers, or stands, either natural or manufactured, to form a continuous length that is suitable for use in weaving or interweaving into textile materials. A group of fibers contained in one bundle for weaving.

A.1.1.7 Related Documentation.

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- Keith B. Armstrong, L. Graham Bevan, and William F. Cole II, "Care and Repair of Advanced Composites," second edition, *SAE International*, 400, Commonwealth Drive, Warrendale, Pennsylvania 15096-0001, USA, 2005, ISBN 0-7680-1062-4.

- "Cutting, Machining and Repairing Composites of Kevlar," DuPont External Affairs, 1111 Tatnall Street, Wilmington, Delaware 19898, USA.

- Zeke Smith, "Understanding Composite Aircraft Construction," *Pathway Book Service*, Lower Village, Gilsum, New Hampshire 03448, USA.

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- T.S Srivatsan and D.M. Bowden, eds., "Machining of Composite Materials," *ASM International*, 1992, ISBN 0-87170-459-5.

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- O. Konur and F.L. Matthews, "The Effect of the Properties of the Constituents on the Fatigue Performance of Composites a Review," *Composites*, Vol. 20, No. 4, July 1989, pp 317-328.

- Fatigue in Composites, H. Rees, "Understand the Basics of Composite Materials Maintenance and Repair," Woodhead, 2003, ISBN 185573608.

- IATA, "Guidance Material for the Design, Maintenance, Inspection and Repair of Thermosetting, Epoxy Matrix, Composite Aircraft Structures," IATA Doc:GEN: 3043, 1st Edition, May 1991, published by the International Air Transport Association, 2000, Peel Street, Montreal, Quebec, Canada, H3A-2R4.

- *Composite Structures*, edited I.H. Marshall, Special Issue, "Supportability of Composite Airframes," Guest ed. E. Demuts, published Elsevier Applied Science, Vol. 10, No. 1, 1988, ISSN: 0263-8223, See Foreword.

- Physical Methods for Materials Characterisation, P.E.J. Flewett, et al., Published by The Institute of Physics, London, United Kingdom, 2003, ISBN - 0750308087.

- A Review of the Influence of Absorbed Moisture on the Properties of Composite Materials Based on Epoxy Resins, W.W Wright, 1979, Royal Aircraft Establishment (now DSTL), Farnborough, United Kingdom, Tech Memo MAT324.

A.1.2 UNDERSTAND THE BASICS OF COMPOSITE MATERIALS MAINTENANCE AND REPAIR.

A.1.2.1 Maintenance Procedures From Damage Detection Through Repair Completion.

A.1.2.1.1 Damage Assessment.

Inspect the damage carefully, which may require careful cleaning of the part before a detailed inspection can be made. If a small puncture exists in a honeycomb panel that is within the specific allowable damage limit (ADL) for that component, it should be dried to specified repair documentation requirements, and then filled with potting compound and taped over. This prevents the damage from deteriorating, and the part must be scheduled for permanent repair within the time limits given in the source documentation. If the damage requires repair before the next flight, the part must be replaced either while the original part is removed for repair or the original part must be repaired before the next flight per the instructions in the authorized repair documentation. Note that the term "allowable damage" allows the aircraft to return to service without being permanently repaired, but does not exclude the requirement for a permanent repair. However, if the temporary repair is inspected and no further damage is

detected, the temporary repair (i.e., speed tape) may be reinstalled without permanent repair. If the damage exceeds tolerance limits, the OEM should be consulted. If the above procedures are not satisfactory, the part must be removed and replaced, if appropriate; rebuilt by the OEM or a qualified maintenance and repair organization (MRO); or be scrapped. In the event that the damaged part is not removable (e.g., a wing skin), the aircraft is grounded, commonly referred to as aircraft on ground (AOG) and an AOG team is sent out to complete a permanent repair. Typically, the AOG team is sent out from the OEM.

The full extent of the damage to a composite part must be mapped using visual and NDI techniques. If the damaged part is a honeycomb sandwich panel, a coin tap test may be used to map the damage. If the damage is to a solid laminate area of a sandwich panel (e.g., the edge band) or a stiffened laminate part, the coin tap test will only detect disbond in the first few layers and an ultrasonic method will be required to establish the boundary of the damage. If a part is determined to be damaged beyond the specific ADL for that part, it must be repaired before the next flight.

A.1.2.1.2 Repair Options.

Authorized documentation must be consulted for permitted sizes of repairable damage. The repair size that is allowed depends on the type of repair. For example, repairs using room-temperature-curing adhesives or resins are usually limited in size. If a hot 65°-93°C (150°-200°F) curing adhesive or resin is used, the permitted repair size becomes larger. If repairs are made at the original cure temperature, then large repairs and sometimes unlimited size repairs are allowed. However, hot-cured repairs may require tooling to maintain the shape of the part, and this tooling may not be available.

Repairs at the original cure temperature also require pressures higher than can be attained using a vacuum bag, thus requiring an autoclave or press. If a large repair is necessary, the part may be sent to an approved repair station that has the required equipment. If the damage is to a stiffened laminate part, the skin should be carefully trimmed to avoid damaging the substructure (e.g., a stringer or rib). Special fasteners and drilling equipment may be needed. Damage to flight-critical structure will usually involve consultation with the OEM.

The part drawing or authorized repair documentation, commonly known as an SRM, is typically available at large airlines and MROs that provide the exact lay-up details, the type of fiber used, and the weight and orientation of each tape and fabric layer. The SRM will list the type of sandwich core, if used, as well as the resins and adhesives that may be used for the part in question. Repairs using the original part materials can be more difficult than manufacturing since availability of the materials in small batches may be difficult. The correct part and revision number must be used to verify the required repair materials and lay-ups.

It is essential to select the correct fiber type and weight of fabric or tape with the correct surface finish to make a strong, durable repair. It is very easy to use a fabric with the wrong weight, and great care must be taken to ensure that this does not happen. When fabric, prepreg, or film adhesive is cut from a roll, a label with full identification details must be attached or included in the plastic bag used to keep the material clean. For dry cloths, the label must give surface finish

details and fabric type and weight. For prepregs and film adhesives, the type and weight is needed, because various weights are available. Composite lay-ups may consist of fabrics and tapes of different weights and types. Aramid is sometimes used with carbon or glass lay-ups to add toughness. As a result, all the plies in a lay-up may not be of the same material or the same weight. Correct identification, location, and orientation of each ply within the lay-up is important. Since the transverse strength of a specific composite layer (ply) is low compared to the fiber direction of the ply, the orientation of each ply is critical to ensure adequate repair strength and stiffness.

When preparing the repair surface, it is important to ensure that it is clean, it has been dried to SRM requirements, and the repair fabric has the required finish. Careful abrasion with aluminum oxide or silicon carbide abrasive paper should be used. Only the surface resin layer should be abraded without damaging the first layer of fiber, using the grit size recommended in the SRM. Note that for repair work, the first ply should be oriented in the same direction as the ply to which it is to be bonded.

A.1.2.1.3 Repair Materials.

- If the required materials are not in stock, the manufacturer may have to be contacted for alternatives.

- If the materials are in stock, it is necessary to check that they are within their permitted shelf life.

- If all the materials are within their shelf-life limits, the materials can be ordered from stock and the work can be planned.

A.1.2.1.4 Damage Removal.

The damage must be cut away completely and if more damage is found, then the repair must be re-assessed for permitted size and repair method.

A.1.2.1.5 Paint and Surface Protection Systems Removal.

The original paint and primer, and any other surface protection system (such as aluminum flame spray for lightning protection), must be removed very carefully to avoid damaging the first layer of fiber. For repairs, careful sanding is probably the best method.

A.1.2.1.6 Core Damage Removal.

Assuming that a honeycomb panel is being repaired, any damaged honeycomb core must be removed and the undamaged bottom skin sanded lightly.

A.1.2.1.7 Core Replacement.

Depending on the type of repair, a two-part paste adhesive may be applied to the bottom skin or a layer of film adhesive may be put in place. This may require a fairly heavy layer of film adhesive to bond the honeycomb core. The new piece of honeycomb must be cut to size so the ribbon direction of the honeycomb matches the original, and the adhesive must be spread on the bottom cells (if a paste is used). If a hot cure is to be performed, the edges of the core must be joined either with an approved potting compound or a layer of foaming film adhesive must be placed around the edge.

Heat and pressure will need to be applied to both the bottom and top skin if the honeycomb on the bottom skin is to be cured at the right temperature and at the same time as the top skin. For this process, the honeycomb must be exactly flush with the top skin. For this reason, it is often better to cure the honeycomb to the bottom skin joint and the edge potting compound in one operation, and then sand the core flush with the top skin and bond the new top skin as a second operation. Room temperature repairs are much easier than hot-cured repairs because the honeycomb can be potted in place without any pressure and left to cure while the top skin layers are cut to shape. Often, room temperature repairs can be made without tooling, which is another advantage over hot cure. However, the SRM usually permits only small repairs of at-room-temperature cure.

A.1.2.1.8 Preparation for Final Cure.

Before final room temperature or hot curing starts, the repair area for the skin patch, the honeycomb core, and the new honeycomb insert must be dried to SRM requirements. If the skins are glass or aramid fiber-reinforced, a moisture meter can be used to check if drying is required. Next, and ideally within 1 hour of drying, the top skin layers must be put in place, after the replacement honeycomb core has been sanded down to the correct height. An extra layer of film adhesive over the honeycomb area is helpful for a good bond, if a hot cure is used. The top skin layers must be of the correct fabric with each ply laid in the correct sequence and aligned in the SRM-specified directions. The overlaps at the edges of the damaged area must be those given in the SRM. When bonding composites or metal repairs at the original cure temperature, additional pressure above the vacuum must be provided, or if any moisture is present, the skins around the repair area will be blown apart due to steam pressure in the cells. The limitations of vacuum pressure need to be understood, although it is a very useful and convenient method of pressure application. Tooling must be used if needed to maintain the part profile.

A.1.2.1.9 Repair Processing.

- Apply a vacuum bag to the repair. A vacuum bag is also used with an autoclave repair to ensure that the autoclave pressure will hold the plies together. If the autoclave pressure leaks into the vacuum bag, the actual pressure to clamp the parts together will be reduced. The lay-up of the vacuum bag and all the release films, both perforated and nonperforated in their correct positions is given in the SRM. If hot curing is used, then thermocouples must also be located as specified in the SRM. The specified temperature and vacuum or autoclave pressure must be maintained throughout the cure cycle, and the pressure must be maintained until the temperature has fallen below 50°C (122°F).

- Postrepair vacuum bag removal. Care must be taken when removing the vacuum bag and release films to ensure that no damage is done to the repair area or the rest of the part.

- Postrepair inspection. Visual inspections and NDIs should be carried out at this stage to confirm that there are no disbonded areas in the repair. The in-process quality control (QC) records (e.g., strip charts printed from the hot bonder or autoclave) must be inspected to make sure that the correct vacuum, autoclave pressure (if used), and temperature were used for the specified period of time.

- Restoration of protection coatings. If the repair is considered satisfactory, any protective coatings, such as erosion-resistant coatings or lightning protection systems, need to be restored before painting. Lightning protection systems must be tested and meet the SRM electrical conductivity requirements.

- Paint restoration. The part should be painted in accordance with the company logo using SRM-approved materials. Some paints, such as polyurethanes, require special masks be worn and safety precautions be taken when being applied.

A.1.2.2 List Key Composite and Expendable Materials Needed for Simple Laminate Structure Repair Including Appropriate Storage Requirements.

- Structural Materials

  Dry (unimpregnated fabric): Carbon, glass, and aramid fibers need to be stored in clean, dry conditions at room temperature, and aramids need to be stored in black plastic bags and out of contact with sunlight to avoid degradation by ultraviolet radiation. Fabrics and tapes made from these materials need to be those specified in the SRM.

  Prepreg materials and film adhesives need to be kept in a freezer at 0°F (-18°C). Shelf life and open time must be recorded.

  Foaming adhesives must be kept in a freezer at 0°F (-18°C), and shelf lives and open times must be recorded.

  Prepregs, film adhesives, and foaming adhesives must be retested at the expiration of their shelf lives and open times or be scrapped. This is because time-expired materials may be partially cured and may not melt and flow well enough during the repair cure cycle to achieve strong and durable bonds.

  Honeycomb materials must be kept in their plastic bags and boxes to avoid damage and then again with what is left over after use. Aramid honeycombs should be dried before use because these materials absorb moisture from the atmosphere. Drying should be conducted per the SRM and Form 8110. They should be used within 1 hour of drying to ensure strong, durable bonds.

- Expendable materials

  There is quite a range of expendable materials that may be used. All the plastic film materials need to be kept clean and dry.

- Vacuum bagging films

  The price of vacuum bag sealant tapes depends on the maximum usage temperature; therefore, the cheaper type can be used at lower temperatures.

  Release films may be perforated or nonperforated. The perforated types are used when surplus resin needs to be bled from a lay-up. The size and number of perforations is important, as too much resin may be lost and the composite surface can become resin-starved if the perforations are too large or too numerous. The choice of perforation size and density can only be found by experience for a given resin system.

  Peel plies are fabrics made from nonstick plastic materials that have no chemical release agents on them. This is vital because their purpose is to provide a rough and uncontaminated surface when two composite parts are joined by adhesive bonding. The use of peel ply, usually for large areas, is to avoid the need to abrade large areas of composite prior to bonding.

  Release fabrics are very similar in appearance to peel plies and must not be confused with them. Release fabrics do have a chemical release agent on the surface to ensure that a part will release from the mold. If this surface later needs to be painted then the chemical release agent must be removed before painting is attempted.

  Breather cloths are used over the lay-up and inside the vacuum bag to allow easy air extraction when the vacuum pump is switched on. They provide an essential pathway for the air to escape and allow atmospheric pressure to be applied to hold the parts together.

  Bleeder cloths are made from the same material as breather cloth and are used on top of the perforated release film to absorb the surplus resin that flows through the perforations.

  Liquid mold release agents may be used when the affected surface does not need to be bonded afterwards. Mold release agents must be applied outside the workshop so other parts will not be contaminated, and can be returned to the workshop only after they have dried on the mold surface.

  Sealant materials, i.e., thiokol, polythioether, and silicone have limited storage lives and must be stored at the temperatures specified on their data sheets. Silicones should be stored separately from other sealants.

A.1.2.3 Tools and Equipment Needed for a Simple Laminate Structural Repair.

A.1.2.3.1 Tooling or Profile Jigs.

The repair of composite or bonded metal parts may require tooling, varying from a flat bench for vacuum bonding flat panels to extremely expensive tools approaching those used in original manufacture for very large repairs and autoclave bonding for components having complex shapes. If both skins of a sandwich panel have to be repaired, then some form of tooling will be needed to maintain the shape, even when room temperature curing is used. If a repair has to be made at the original cure temperature and the part is curved, then tooling capable of withstanding the cure temperature without distortion should be used.   (See SAE Commercial Aircraft Composite Repair Committee (CACRC) publication, SAE AIR 5431—Repair Tooling.)

A.1.2.3.2 Important Factors to Consider.

The cure temperature must not distort the tooling, and ideally, the tooling should be made from a material having the same thermal expansion coefficient as the part. This is particularly important for profiled parts that need to be made to an accurate shape.

The tooling should not be too heavy, which would make movement and general handling difficult.

The tooling must be designed for surface bagging, not for envelope bagging, so the tooling itself is not under great stress during cure.

A.1.2.3.3 Small Repair Tooling.

Splash moldings can be made from another part of the same part number by making a gypsum intermediate tooling on the part. This type of tooling can be used to make a profile from low-temperature curing prepreg. The tooling can be used to make a patch that can be bonded to the part for repair. Alternatively, the original part can be used if a layer of release film is laid over it, provided that the spare part can be made available for the length of time required to make the patch.

Simple, solid laminate tooling can be made from a number of suitable fabric layers and epoxy resins to achieve the stiffness required.

Two-Part Fiberglass Tooling: Another simple tooling technique, if a light-weight stiff tooling is required, is to make a sandwich panel of fiberglass, or other fiber, and a two-part, room-temperature-curing epoxy resin and honeycomb core. A layer of release film must be taped to another part of the same shape that is being repaired, then a few fiberglass skins can be laid up, a honeycomb core added, and then a few more layers of fiberglass. The assembly can then be vacuum bagged to the part and allowed to cure.

Simple aluminum alloy tooling can sometimes be made by rolling a plate to the required radius for the parts. In this case, envelope bagging works well.

Several companies make room-temperature and low-temperature-curing prepregs from most fabrics, and after the initial cure, these can be step cured at intervals of about 20°C up to 200°C, and become very good, durable mold tooling for repairs or manufacture.

A.1.2.3.4 Equipment.

The following equipment is needed for a simple laminate structural repair.

- A flat workbench of adequate size.

- If the part is contoured and not flat, then a mold or suitable tooling may be needed to maintain the part shape.

- Hand tools will be needed, such as pneumatic drills and diamond-cutting discs.

- Tank cutters are needed to cut corner radii.

- Grinding burrs are required to smooth corners and blend them into straight cuts.

- Stanley knives or large, old hacksaw blades ground to a sharp edge are useful for cutting honeycomb.

- A high-speed band saw will be needed if sheets of expanded honeycomb need to be cut to thickness. The band saw requires special blades with a 0.05-mm (0.002-inch) offset on the teeth to avoid tearing the honeycomb. These blades need to be sharp.

- Sheets of abrasive paper of the required type and grit size must be readily available.

- A can of acetone is required, with suitable tissues or rags to wipe off any surplus resin from the part and for cleaning tools.

- Medium files are useful for trimming edges.

- A vacuum pump and gauge are needed with associated piping.

- A moisture meter is used to check when fiberglass and aramid parts are dry enough to be bonded. As there are no economical moisture meters at present, carbon fiber parts must be dried to the SRM procedures as a precaution.

- Thermocouples will be required if a hot cure is needed.

- Heater blankets will be required for hot curing. It should be noted that these need to be at least 2 inches (50 mm) larger than the repair area to avoid cool spots at the edge of a repair.

- A hot-bonding console is required to control the heater blankets via the thermocouples that are laid around the part.

- Heat lamps may be needed to add heat to any local cold spots to maintain an even temperature across the entire part. If honeycomb panels are located over ribs or any other heavy structure, they may require additional local heating in these areas, because heavy parts act as heat sinks.

- If liquid resins are used, suitable paint brushes will be needed to apply the resin to the fabric layers.

- Electronic scales accurate to one-tenth of a gram will be needed to weigh out resins and two-part adhesives.

- Shallow aluminum trays will be needed for resin and adhesive mixing.

- Packs of medical tongue depressors can be used for stirring liquid resins.

- Sharp scissors will be required to cut the cloth plies to shape.

- A supply of clean rags is needed to clean up any spilled resin.

- A roll of clean tissue is needed when cleaning paint brushes and other items.

- If a solid laminate has to be repaired using fasteners, then suitable drills and reamers will be required to produce accurate holes. These may require support jigs to ensure precise hole alignment.

A.1.2.4 Understand Personal and Equipment Safety Requirements.

Some people working with composites and adhesives have, after a period of exposure, become so sensitive to epoxy resins that they cannot stay in the same room with a film adhesive without developing a skin rash.

Dermatitis, or skin rash, can occur if workers do not wear gloves at all times. Resins must not be allowed to contact the skin. If contact with resins occurs, it should be washed off with soap and water immediately.

- Wear rubber or latex gloves and overalls, and lap the gloves over the sleeves. If solvents are used to remove resin from the skin, the natural skin oils will be dissolved and cause the skin to crack and can become infected. Ensure that the gloves are compatible with the solvent cleaner. For example, acetone will dissolve cheap latex gloves very quickly. Gloves need to be worn when handling prepregs and film adhesives as well as wet lay-up resins. Allergic skin reactions to resins, adhesives, and prepregs are also consequences of not wearing personal protective equipment such as overalls.

- Use sealed containers of cleaning solvent, such as acetone. Do not leave it open to evaporate into the workshop air. Most solvents are flammable and have vapors that are a health hazard. Electrical equipment used in a bonding shop must be spark proof, but only specially designed equipment meet this standard.

- Compressed air must not be used to blow dust away from the part. It contaminates the atmosphere and the bonding surfaces. Vacuums with composite dust-collection systems should be used instead.

- Lubricants and water-repellent compounds act as release agents and must not be located in any bond shop.

- All tools must be handled with care, and any supplied safety guards must be used.

- Masks must be worn if the material being sprayed requires them.

- Safety glasses or goggles should always be worn. This is especially important if low-viscosity resins are being mixed. Approved face shields can also be used.

- Helmets that cover the ears should be worn during the paint-spraying process.

- Approved hearing protection should be worn during noisy processes, such as riveting or grinding.

- Ventilation booths with extractor fans should be used when mixing resins that emit hazardous or unpleasant vapors.

A.1.2.5 Describe the Differences Between Repairing Composite and Metal Structures.

Metal surfaces need special treatments to raise their surface energy to ensure good adhesion. Aluminum alloys need phosphoric or chromic acid anodizing to produce a high-energy oxide layer with a porous structure that aids good adhesion. Other methods are available but are not as effective. Titanium and corrosion-resistant steel can be bonded, but always check the SRM recommendation for treatment. Good surface preparation is more important than anything else for good, durable bonds in repairing metal parts.

Primers are often used when bonding metals. The primers must be compatible with the adhesive used. Primers do not improve bond strength significantly, but they do improve corrosion resistance and, hence, durability.

Metal surfaces are not porous, and therefore, when honeycomb-cored panels are being repaired, there is no escape path for volatiles from the resin or air that is in the core. To allow vacuum pressure to work, a thin positioning cloth from 3M, called AF 3306-2, can be used between the honeycomb and the film adhesive that is laid up against the skin. This allows air and volatiles to be extracted from the core to produce a good bonding pressure. Once a good vacuum has been drawn, heat is applied, melting the adhesive, which absorbs the positioning cloth. Thus, a good bond with good fillets to the honeycomb is achieved. A fairly heavy layer of film adhesive is needed to absorb the fabric and ensure good filleting to the honeycomb cell ends. Film adhesive weighing 425 g/m^2 (085 lb/ft^2) is recommended or two layers of 300 g/m^2 (0.06 lb/ft^2), if the heavier grade is not available.

Metals have the same properties in all directions. The metal repair patch should be the same thickness as the original skin or one gauge thicker, but no more; otherwise, too much load will be attracted to the patch. The patch should be chamfered to half its thickness and tapered over about a 1/2 inch (12.7 mm) at the edges to reduce edge stresses and improve airflow. All shaping and trimming must be done before anodizing to avoid contaminating the bonding surfaces. The overlap all around the skin cutout should be in accordance with the SRM, and it is usually 50 times the skin thickness.

In-plane properties of composites depend on ply orientation. The composite skin plies must be oriented correctly, per the SRM or drawing, and laid up in the correct sequence so the patch is oriented in the direction that reflects the strength requirements. In contrast, a sheet of metal has similar properties in all directions.

The existing skin must be cleaned, dried, and lightly abraded, but for small repairs, this is easier than the chemical treatment of metal skins. Always use the SRM-recommended drying time as a minimum. The reason for this is that composite resins, and in the case of aramids, the fibers will absorb moisture. The parts will not feel wet or show any visible signs of moisture, but drying is needed to obtain stronger and more durable bonds to composite surfaces. The longer a part has been in service, the more important drying becomes. Moisture meters can be used on fiberglass and aramids but not on carbon fibers because they are electrically conductive. For high bond strength and long-term durability, metals require complex chemical surface treatments.

Composite skins are fairly porous. If a wet lay-up repair is done at room temperature, or up to about 95°C, or if a prepreg is used, some gas and air can be drawn out through the resin to give a bonding pressure to hold the skin to the honeycomb and the repair area of the skin. The area surrounding the repair may be blown apart if there is water in the honeycomb and a cure at the original temperature is attempted. To avoid this problem, all repairs to honeycomb panels should be made at a temperature significantly lower than the original manufacturing temperature. If this is not permitted, then a higher pressure must be provided by a press or autoclave.

Primers are seldom used on composites for adhesive bonding. They are almost always used on metal parts. Modern paint systems, such as Desothane® HS (CA 8000) from PRC De Soto®, use a strong primer that will remain on the aircraft for its whole life. An intercoat is then applied and followed by a topcoat. The use of a solvent stripper, based on benzyl alcohol, allows the topcoat and intercoat to be removed several times without damaging the primer. Great care must be taken with this process as many aircraft are repainted ten or more times during a 30-year service life. Repair areas will need to use the manufacturer's primer to restore this capability. Metals almost always use primers to minimize corrosion and increase durability of the bond. Similarly, composites use a primer to protect the first layer of fiber. This paint system is also used for metals when a brightly colored, high-strength, durable primer is used to protect an anodized aluminum surface.

Similarities between metal and composite bond repairs are described below.

- Adequate pressure is needed for bonding.

- Correct cure temperatures and times must be used.

- Composite repair patches cured at high temperature often use a film adhesive to help achieve a good bonding of the repair patch to the honeycomb and to the skin. This is similar to a metal repair patch.

- Only anodized aluminum alloy honeycomb should be used for repair or manufacture. This applies to composite and metal panels if aluminum honeycomb is used in the original panel. Note that most composite-skinned honeycomb panels use aramid honeycomb or foam cores.

- The original primer must be restored in repaired areas.

A.1.2.6 Describe the Process of Metal Bonding.

Good surface preparation of metals and most other surfaces is the single-most important factor in achieving good, durable adhesive bonds.

Sheet metal, for adhesively bonded construction or repair, should be kept clean and free of scratches until the anodizing or other treatment processes. All cutting, trimming, and edge chamfering should be completed before the part is anodized or given any other surface treatment. Holes and countersinks should also be drilled before anodizing, if possible. One OEM process specification for phosphoric acid anodizing is as follows:

- Step 1. Vapor degreasing.

- Step 2. Alkaline cleaning in a tank.

- Step 3. Alkaline solution rinsed in clean water.

- Step 4. Deoxidize per OEM specification.

- Step 5. Rinse again.

- Step 6. Anodize in phosphoric acid solution at the specified voltage for the required period of time.

- Step 7. Remove details from the anodizing solution and start rinsing within 2 minutes after the current is switched off. Cold water rinse for 10-15 minutes.

- Step 8. Dry thoroughly at 60°C (maximum).

- Step 9. Examine for the presence of an anodic coating.

After thorough drying and within 4 hours, the part should be sprayed with a primer and allowed to dry. It is then ready for the bonding process. If the bonding process does not take place immediately, the part should be placed in a clean plastic bag. Priming within 4 hours of drying is essential, as the highly reactive surface produced will attract dust and dirt that will progressively reduce the high surface energy required for good bonding.

A.1.2.7 Discuss the Importance of Approved Data, Methods, and Procedures Used in Product-Specific, Composite Maintenance and Repair.

The essential theme throughout this course is to stress the importance of technicians, inspectors, and engineers working as a team. In remote regions that require repairs, these functions must be fulfilled even if individuals are not assigned to these functions. Therefore, personnel must have the proper skills to perform an acceptable repair, depending on their unique environments. Each teammate involved in damage disposition, inspection, and repair should have the training needed to complete their tasks.

Teamwork is essential to composite maintenance, particularly as associated with the steps involved in aircraft structural inspection, disposition, and repair. Team members should have some awareness of the different skills needed to successfully perform each step. This awareness serves to better understand personal skill limits and where to get help. The information contained in approved documentation [A-1] for a particular aircraft structure should have the necessary supporting databases and sufficient details to guide a team in the field through the steps of inspection, disposition, and repair.

This topic is discussed in more detail in sections A.1.3.6 and A.1.3.7.

A.1.2.8 Related Documentation.

Armstrong, Keith B., Bevan, L. Graham, and Cole, William F., II, "Care and Repair of Advanced Composites," second edition, *SAE International*, 400, Commonwealth Drive, Warrendale, Pennsylvania 15096-0001, USA, 2005, ISBN 0-7680-1062-4.

A.1.3 UNDERSTAND OTHER CRITICAL ELEMENTS OF COMPOSITE MAINTENANCE AND REPAIR.

A.1.3.1 Discuss Issues Affecting the Selection of Bonded or Bolted Repairs.

If a stiffened laminate structure has to be repaired, it is generally accepted that bonded joints are more effective for thin laminates. As an example, the required bolt pattern for a repair to a small through penetration of a current-generation commercial aircraft composite horizontal stabilizer is larger in area than a bonded repair for the same damage, and there is the additional weight of the repair plate and the metal fasteners. For thin laminates, there is also the concern of short grip lengths and low thickness-to-diameter ratio. As thicker stiffened laminate primary structure (e.g., wing main toque box skins and spars) comes into use, bolted repairs will become more efficient due to the lengths of the required scarf or ply steps for bonded repairs. Bolted repairs are currently more easily inspected for structural integrity than bonded repairs.

Repairs are intended to restore the load path removed by damage, which in the ideal case, restores the original load distribution of the component.

There are four main approaches to patch repairs:

- External bonded patches

- Flush (scarf-bonded) patches
- Bolted patches
- Bolted and bonded patches

Adhesively bonded patches provide effective load transfer; external patches minimize further damage to the structure since fastener holes are not required and less material is removed compared to flush-bonded patches. In general, bonded repairs are capable of restoring the original strength of the composite, but require increased technical skills due to the greater degree of complexity. Repair time is often longer, the need for dry parts to avoid porosity associated with the formation of water vapor during cure, and the limited storage life of adhesives and other materials are other disadvantages of bonded repairs.

External bonded patches are generally restricted to thin-skin applications (up to 2-3 millimeters in thickness for carbon/epoxy structures), whereas flush (scarf-bonded) and bolted repairs are applicable to thicker sections.

Scarf repairs are also used when either aerodynamic smoothness or other operational characteristics are required. Scarf repairs to thick laminates are difficult to apply, requiring highly skilled technicians. In addition, scarf repairs to thick laminates require removal of a large amount of material because of the shallow scarf angles required. The obvious disadvantage is that there is less of the original material to carry the load. Scarfed or stepped joints used for thick materials require very long overlaps or scarfed areas to meet the recommended overlap or scarf length (50 to 80 times the thickness) of the material being repaired. Some scarf joints are specified with 30 times the thickness of the material as the taper. To achieve the full benefit of scarf joints, the scarf on the structure and the repair section must be accurately matched. One way to reduce the removal of good material for a scarfed joint is to make the joint double-sided, as shown in figure A-4. Stepped lap joints must be cut carefully to avoid cutting part way through any ply. This may require the use of special tooling. The advantage of stepped lap joints over scarf joints is the adhesive has less of a tendency to run out of the joint.

Scarf distance

Double-sided
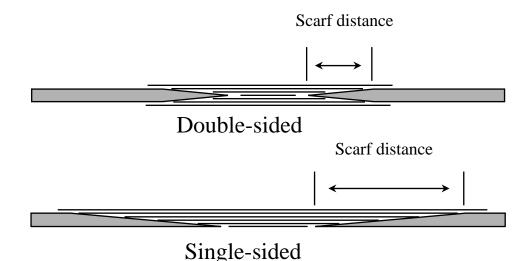
Scarf distance

Single-sided

Figure A-4. Single- and Double-Sided Scarfing
(Courtesy Abaris Training Resources Incorporated)

If a quick, temporary repair has to be made to a sandwich panel or fairing, a bolted repair, using a thin sheet of metal on the outside and large washers on the inside, can allow the aircraft to return to base. This may be economical if it allows the aircraft to be in position for the next day's operations. Such a temporary repair will ultimately require a large permanent repair and may result in the part being scrapped.

Titanium alloy is generally used for a metal patch due to its resistance to corrosion when in contact with carbon/epoxy. Aluminum alloys do corrode when used with carbon/epoxy; however, it can be used if precautions are taken to avoid corrosion by insulating the repair from the carbon/epoxy structure. Aluminum alloy is significantly easier to drill than titanium. Another alternative is to use a carbon/epoxy patch that has been formed by wet lay-up from carbon cloth, although the load-bearing capacity is less than titanium alloy.

Types of repair are related to damage assessment and location on the aircraft. For instance:

- If a low-observability repair is required, a flush bonded repair will have to be made.

- In areas where the airflow is critical, such as the leading edge of a wing or an engine nacelle, a flush bonded repair is required.

- If damage occurs to a heavily loaded, stiffened laminate structure, bolted repairs are most likely to be used, sometimes in conjunction with adhesive bonding. For example, wing skins, stiffened by stringers, are likely to be repaired using bolts or blind fasteners.

- When these scenarios are considered, the number of occasions when there is a choice, bolting is less likely to be chosen.

- In general, if the panels are of sandwich construction, bonded repairs are usually chosen.

- If the parts are heavily loaded, stiffened laminate structures, bolted repairs are quite often used.

A.1.3.2   Describe Various Electrical Requirements and Effects, Including Prevention of Corrosion, Hazards of Electromagnetic Interference, and Electrostatic Discharge (lightning protection systems), and how They Need to be Considered During the Repair Processes.

A.1.3.2.1 Corrosion Prevention.

- When aluminum parts are in contact with composite parts, the possibility of corrosion to the aluminum needs to be considered.

- All bolts and fasteners used in carbon fiber composites must be made from bare titanium alloy, titanium alloy coated with aluminum, or corrosion-resistant steel.

- Aluminum or alloy steel fasteners may not be used in carbon fiber (graphite) structures.

- Glass fiber and aramid fiber composites are not a problem, because they are not electrically conductive.

- Metal parts used as reinforcements in composites must be properly surface-treated and primed.

- Aluminum alloys adjacent to carbon-reinforced composite parts require a layer of glass fiber to be applied to the composite parts to minimize electrical contact and prevent corrosion to the aluminum parts. The adhesive should also be nonconductive and have low moisture uptake.

- Trials using carbon fiber composites as reinforcements for cracked aluminum alloy parts have consistently shown that if moisture enters the crack, severe aluminum corrosion takes place.

A.1.3.2.2 Hazards of Electromagnetic Interference.

For many years, aircraft made from aluminum alloys have had very few problems with electromagnetic interference, because the aircraft structure could conduct the signals away and prevent any unwanted effects.

With the increased use of composites, this problem has become more significant. If radio frequency/electromagnetic interference shielding is included in any part of the design, repair work must ensure that this protection is replaced and tested to ensure that the original level of protection is replaced and meets the specification requirements as well as the latest Aeronautical Radio Incorporated standards.

A.1.3.2.3 Lightning Protection Systems.

Lightning strikes are powerful and can reach up to 2 million volts and up to 300 amps. Lightning protection systems are applied to radomes, wing tips, rudder and elevator tips, and any aircraft extremities, including the leading edges of engine nacelles. These protection systems are vital to safe aircraft operation. If parts that are coated with lightning protection systems have to be repaired, the repair must include full restoration of the lightning protection system.

In one case of a lightning-damaged elevator, the inductive forces had fractured the grounding lead between the horizontal stabilizer and the elevator, although it was about 3 cm wide and about 3 mm thick. The electrical current also ran along the carbon fiber front spar, up through a bolt, and out the elevator tip. The bolt was undercut at the head and the bolt hole was enlarged. The temperature was so high at the bolt position (approximately 1000°C) that the resin had completely vanished from the fibers. In addition, the current had traveled from the fuselage and through the elevator-operating ball race bearing. The bearing itself felt gritty, wherein each ball had been almost welded to the race bearing itself. One elevator hinge bearing felt much the same.

On one other recorded occasion, a light aircraft was lost because the elevator control bearing had been welded in similar circumstances.

Lightning strike protection (LSP) systems are bonded to the exposed surfaces of the component in the form of metallic mesh, flame-sprayed coatings, or metallic grids to protect the aircraft from lightning damage. The grids, often referred to as picture frames, are comprised of solid metallic strips, which are bonded in a grid fashion over lightning-susceptible locations that contain fasteners. Many of these susceptible fasteners are located in fuelled areas or in areas covering electrical-sensitive equipment. The fasteners are drilled and countersunk through the bonded grid such that the fastener heads are nearly touching the metallic strip.

With this configuration, a lightning strike to the fastener head will be conducted from the fastener head to the grid for dispersal along the part surface, rather than having the strike penetrate into the part. The dispersal by the grid, rather than into the laminate, prevents a concentrated charge from damaging the fastener and the fastener hole laminate structure, because copper is closer to graphite in the galvanic series.

LSPs are not always in the form of coatings and may consist of aluminum strips, usually on the outside of radomes, but sometimes on the inside. Picture frames made of aluminum are usually used at rudder, aileron, and elevator tips.

Copper mesh and foil are also used for LSP when applied to graphite structures to eliminate galvanic-corrosion effects.

The conductive coating forms of lightning protection are:

- Flame-sprayed aluminum—Metal-coated fabrics, such as Thorstrand, Alumesh, and Alutiss. These fibers are coated with aluminum and embedded in the outer layer of composite components. Wire mesh has also been used, i.e., woven aluminum wire fabric.

- Expanded foil mesh is a relatively recent development and has good conductivity. Copper or aluminum foil can be used to make the mesh. In some cases, an aluminum foil was laid into a mold and the composite prepreg laid up onto it. On some occasions, a layer of SpeedTape may be used as a temporary replacement for an area of flame-sprayed aluminum on the surface of a composite part after a repair had removed the original layer.

A.1.3.2.4 Electrostatic Discharge.

The friction of high-velocity air passing over a large-surface electrical insulator, such as a radome or cockpit window, results in the build-up of static electricity that periodically discharges by arcing to a metallic part of the aircraft.

These discharges can punch very small holes in a radome's protective coating and through the outer skin, which can lead to moisture penetration and a loss of radar signal strength. Using an anti-static coating, which allows the charge to slowly bleed away, can solve this problem.

A.1.3.2.5 Precipitation Static.

Static charge and precipitation static have a common origin but differ in degree. Normal air produces a static charge by the friction of air molecules, but precipitation static comes from larger masses, such as raindrops, hailstones, dry ice, and snow. This is also known as p-static.

P-static can lead to radio communication problems, often at low altitudes when communications for landing are most important. For these reasons, conductive coatings are applied to windows and radomes and must be restored when radomes are repaired.

Composite structures are very susceptible to p-static effects. Care must be taken to maintain configurations that bleed off p-static charge when designing and implementing composite repairs.

A.1.3.2.6 Radar Signals and Radomes.

Most aircraft radomes are made from fiberglass or aramid skins with honeycomb cores. Some use quartz fiber fabrics.

The thickness of the sandwich is critical. It must be one quarter of the radar signal wavelength for maximum signal transmission. The correct thickness must be maintained and pass the wave test, which checks the signal transmission. For an X-Band Radar, the correct thickness is 0.31 inch (8 mm).

Resins used to repair radomes must not contain any conductive fillers. These too can drain away some of the signal and reduce transmission strength.

A.1.3.2.7 Safety Issues Related to Electrical Requirements.

- Work strictly to approved source documentation.

- Do not use more than the specified amount of paint on a radome, as this can reduce signal transmission.

- Always perform a wave test and ensure that this meets the standard required.

- Restore any lightning protection system fitted to the radome and check that it meets the electrical resistance requirement.

A.1.3.3 Understand the Need for Protective Coatings and Surface Finishing Steps Used for Composite Aircraft Structure.

Protective coatings may be used for static bleed-off, as mentioned above, or they may be needed for lightning or erosion protection.

A.1.3.3.1 Erosion-Resistant Coatings.

Erosion-resistant coatings have a number of applications ranging from polyurethane paint coatings or polyurethane or neoprene boots on radomes to metal erosion shields on the leading edges of helicopter blades and on gas turbine engine bypass fan blades. If boots are used, they should be 0.5 mm (0.02 inch) thick, or thicker, to give longer service life.

These coatings are used to reduce erosion caused by rain, hail, dust, or sand. The higher the speed of an aircraft or the airspeed of a helicopter blade or engine fan blade, the more erosion is likely to take place. Clearly, if helicopters are operated in desert areas, or aircraft fly to dusty airfields, then more erosion will take place.

Metal erosion shields are often made from titanium alloy, corrosion-resistant steel, or aluminum sheet formed to the required contour.

A.1.3.4 Describe Typical Paint and Surface-Layer Removal Techniques for Finished Composite Parts.

A.1.3.4.1 Paint Removal for Repair.

The paint system must be removed to perform any bonded repair. For small- to medium-sized repairs, aluminum oxide, silicon carbide abrasive papers, or Scotch-Brite pads of suitable grit size are commonly used. Care must be taken not to go through the primer coat or to damage the first layer of fibers. When the primer coat is visible, a finer abrasive paper should be used to avoid damage to the first layer of fiber. Abrasive papers become finer as the grit breaks up with use, but this usually generates more heat. Hand abrasion is preferred since mechanical abrasion may cause further damage to the component.

A.1.3.4.2 Large Area Paint Removal.

If a large part or the entire aircraft needs to be repainted, alternative systems need to be considered. The paint scheme on an aircraft can be changed ten or more times during the life of an aircraft. Aircraft made today are expected to be in service for about 30 years. Paint is heavy, so it must be removed before the new paint is applied. If an aircraft is rented to a number of airlines over the years, then the number of paint scheme changes could be even larger. This is a difficult area to deal with because new methods are being tried at frequent intervals. This has also been studied by an SAE committee, which produced the documents SAE MA 4872, "Paint Stripping of Commercial Aircraft—Evaluation of Materials and Processes," and "IATA Guidelines for Evaluation of Aircraft Paint Stripping Materials and Processes," latest issue. SAE MA 4872 was last amended in March 1998, and copies can be obtained from SAE. SAE MA 4872 describes the methods to evaluate paint-stripping systems; it does not give recommendations or procedures for using them during maintenance.

A.1.3.4.3 Paint-Stripping Methods.

The objective for paint-stripping methods is to remove existing paint quickly, easily, and cheaply without damaging the first ply of composite, or in the case of aluminum alloy skins, the anodized oxide layer. Certain paints are more easily removed, such as Desothane CA 8000 polyurethane paint. If this system is used, the topcoat and inner coat can be removed with a solvent to avoid mechanical damage. Only the primer will need to be removed with an abrasive paper. SAE MA 4872 includes:

- High-pressure water blasting, with or without the use of chemical paint softeners

- Dry-media blasting (e.g., wheat starch, plastic media blast (PMB), etc.)

- Wet-media blasting (e.g., sodium bicarbonate)

- Ice pellet blasting (e.g., CO_2 or water, with or without the use of chemical paint softeners)

- Chemical paint stripping (e.g., with or without a dedicated strippable layer in the paint system)

- Thermal paint stripping (with or without the use of ice pellet stripping)

These techniques must be used with care and in accordance with any procedures that have been developed. It is essential that no damage is done either to an anodized metallic surface or to the outer ply of a composite component. Although inclusion in this report does not authoritatively endorse any of these processes, the following alternatives have been tried:

- The use of paint-stripping chemicals should be avoided, as they can damage the resin matrix. For example, a methylene chloride paint stripper was used on a fiberglass-skinned honeycomb panel. (The operators said they thought it was aluminum.) As a result, two layers of fiberglass skin were partially peeled away from the honeycomb.

- One technique to avoid this problem was to coat the composite with a relatively thick layer of bright-orange primer so it would be obvious to the person preparing the surface that it was time to stop before damage was done. This was not helpful if the operator lost concentration for more than a few seconds.

- Another technique is to use a strong primer coat, brightly colored as mentioned above, and then apply an inner coat, followed by a topcoat of polyurethane paint. These two layers can be removed using a benzyl alcohol-based solvent, which does not damage the primer coat. This paint scheme Desothane HS (CA 8000), from PPG Aerospace /PRC De Soto, has been successfully used for 5 years and is very durable. The topcoat and inner coat can be removed using a benzyl alcohol-based solvent without damaging the primer. This can be done many times if care is taken with this process. The neutral version of benzyl alcohol should be used because it does not damage aluminum structures, unlike the acidic versions, which must not be used on aluminum.

- Another potential method is from Sponge-Jet®, Inc. This method uses abrasive grit of various grit sizes and material types in a softer material (polyurethane foam) reducing the rate of surface removal to a more controllable level, and it absorbs some of the dust produced. The media can be recycled a few times, which reduces cost. This system is more controllable than dry PMB and can remove paint layer-by-layer without damaging the primer, provided that care is taken and the correct grade of grit is used at the correct pressure and jet angle settings. The Aero-Alox™ 320 grit system has been used on metallic and composite aircraft, aerospace components, and radomes. The Sponge-Jet method looks promising. This is a relatively new process not listed in SAE MA 4872, but seems to be an improvement on those listed.

These techniques need to be used carefully and in accordance with the manufacturer's instructions, as they are sensitive to operator technique. They are suggestions only, and any system used must be OEM-approved and must be used as specified by the supplier. The above list does not constitute approval of any particular system.

A.1.3.5 Discuss Proper Disposal of Wastes From the Composite Repair Process.

Environmental Health and Safety Acts in the USA, and similar legislation worldwide, provide the required information to properly dispose the waste. Legislation varies across the world, and all known hazards should be removed as much as possible whether legislation exists or not. Local regulations must be observed at the minimum.

- Almost all the materials used need to be disposed of as hazardous waste.

- All resins, film, paste adhesives, and prepregs are safer in landfill sites if they are fully cured before disposal. These materials are less toxic in the cured state, as the base resin and curing agent have reacted together to produce a solid resin. This applies not only to cutoffs of prepreg when cloths are trimmed to shape, but also to time-expired materials that need to be cured even if they are about to be scrapped. With a little forethought and planning, they can be cured in an oven or autoclave along with production items. However, exothermic heat can be a problem and is certain to occur if large amounts are cured at the same time. For example, it is not sensible to cure a complete roll of film adhesive or prepreg at one time, even when it is time-expired. Similar problems are likely to arise if paste adhesives or liquid resins that are time-expired are cured in amounts exceeding the data sheet limits. Shallow aluminum trays should be used for mixing so that heat can escape, and maximum mixing quantities should not be exceeded. Check the data sheet. Disposal must be in hazardous-waste containers.

- Dust from cutting composites or abrading their surfaces should be removed with a vacuum cleaner that uses a disposable bag that can be sealed. Disposal must be in hazardous-waste containers.

- Waste solvents also need to be disposed of in accordance with local regulations and recycled where facilities exist.

- Other waste, such as paint brushes solidified by resin, and tissues and wipes that have wiped resin from parts or benches, also needs to be treated as hazardous waste. Scraps of dry fabric too small for use at another time and mixing sticks coated with resin also need to be included. Disposal must be made in hazardous-waste containers.

- All adhesives, resins, sealants, solvents, and other chemicals must be labelled as to its use and disposition.

- Because waste is produced in significant quantities, workshop benches and floors should be vacuumed at least once a day and preferably twice a day. Use a vacuum cleaner with a disposable bag that can be sealed. Disposal must be in hazardous-waste containers.

- Technical data sheets and material safety data sheets must be made available to those using composite materials, resins, adhesives, sealants, solvents, paints, etc., so that personal safety precautions can be taken and waste disposal instructions observed. The company must set up collection points for hazardous materials with suitable instructions for their use. Follow local health and safety regulations and local regulations for the disposal of hazardous waste.

A.1.3.6 Understand Personnel Skill Limits and Where to Receive Assistance During Maintenance.

Performing a repair action beyond a person's capability can be a flight safety issue. The repair of aircraft composite structure is an important task on which the lives of others depend.

Always check the data sheet, SRM, and Form 8110 for details. For example, there are many different two-part resin systems, and their mix ratios vary by quite large amounts. To obtain the required strength and the correct maximum service temperature, adhesives and resins must be weighed correctly and mixed thoroughly for at least 3 minutes using a timer or watch. Always check the mix ratio for the two-part system being used. Almost certainly it will not be the same as a previously mixed resin.

A foreman or supervisor can help, but in some companies, these posts have only administrative responsibilities.

One area where it is essential to use the skills, and equipment, of others is in NDI. Ultrasonic methods, in particular, need experienced personnel to correctly interpret NDI device readouts. Only trained personnel, with radiation-film badges, are allowed to interpret x-ray film. Thermography and holography also require trained personnel.

A.1.3.7 Discuss the Importance of Documenting and Sharing Information About Damage Scenarios Discovered in Service Between OEM, MROs, and Regulators.

Whenever in-service damage is found that is not included in approved documents, inspection and repair will generally require special instructions. Such cases require guidance from other team members with the skills needed to determine the full extent of damage and develop a repair that meets the airworthiness requirements that were originally posed on the base structure. For

primary structures, the data and analyses needed to substantiate that the repair meets such requirements may be extensive (e.g., stiffness, strength, fatigue, and damage tolerance for critical load cases). Team members not versed in making such engineering judgments for a specific structure should never assume that the inspection and repair is similar to previous experiences with other structures.

In addition to training for the general skills used in composite damage disposition, inspection, and repair, the personnel (engineers, inspectors, and technicians) must have detailed knowledge of the particular part. This includes an understanding of approved source documentation, such as OEM specifications, drawings, inspection procedures, and repair processes. Some of this information may be referenced or exist in the OEM SRM, or equivalent. In addition to an understanding of the source documentation, the team must often have numerous technical skills for the damage disposition, inspection, and repair of a particular structure. Sometimes such skills may only be attained in training provided directly by the OEM.

A complete disposition process is needed after composite damage is first discovered. Operations personnel become the first line of defense in reporting known service events that may have damaged the aircraft. They may also discover clearly visible damage while servicing the aircraft or during frequent walk-around inspections. Operations personnel must report possible damage to maintenance personnel before the appropriate inspection and disposition is initiated. Maintenance personnel that get involved must understand the limits of the inspection methods used to disposition the composite damage. This understanding usually exists in an organization performing heavy composite maintenance inspections while aircraft are grounded for a significant period of time. However, possible damage discovered in the field would likely first be reported to field maintenance personnel without the special skills needed for a more thorough composite inspection. Since damage to composite structure often includes characteristics that are not visibly obvious (e.g., delamination and debonded elements), operations and maintenance personnel in the field must be informed that special skills are needed for inspection and disposition. Properly trained people must be contacted to avoid putting an aircraft with severely damaged composite structure back in service.

Operations personnel are critical to protecting the safety of aircraft exposed to anomalous service events that may lead to damage not considered in design. Their important role is limited to the realization that such an event may have damaged the aircraft and, therefore, needs to be reported for maintenance action. Some anomalous service events that need to be reported include (1) high-energy impacts due to service vehicle collisions; (2) flight excursions outside the design envelope; (3) severe landing loads; and (4) other abnormal flight, landing, or ground events outside the scope of that substantiated during type design certification. Hail damage can sometimes be severe. Damage caused in such anomalous events is a valid safety threat for all types of aircraft structures. A particular concern for composite structures comes from the nature of damage, which may not include obvious evidence of severe loading or material distress (e.g., dents or other permanent distortions that indicate metal yielding). Safety management principles for communication between the type certificate holder, owner, maintenance organization, and operations personnel is paramount to mitigating risks. Any damage suspected due to an anomalous ground, flight, or landing event must be reported to maintenance engineers

and inspectors to help determine the severity of damage. This should be the case whether or not exterior damage is visibly detectable.

Composite damage that can be reliably detected using visual inspection methods will usually require additional NDI methods to determine the full extent of damage. Composite inspectors must be trained to realize that small visual dents and other indications of surface failures often come with additional hidden damage that requires NDI for disposition. Source documentation should be reviewed to determine the recommended NDI method for given structural details. The different composite damage types and NDI methods that can be used to fully characterize the extent of damage will be reviewed later in this section.

Once the damage is completely characterized, engineers will need to consult source documentation to determine if it is within ADL. Component records should also be reviewed for previous repairs and existing ADL in the proximity of damage. Many approved documents restrict the distance between composite repairs and ADL. When damage is within ADL for a given location, some maintenance action may still be needed to replace protective surface layers and seal the damage. Damage beyond the ADL must be repaired. Source documentation also provides limits on what can be repaired in the field without further consultation with the OEM. Use of repair limits highlights the importance of an accurate damage disposition process.

The design details, material types, tooling, and process instructions used for previously approved repairs should be clearly outlined in source documentation. These steps must be closely followed by maintenance (repair) technicians. Any deviations require approval. Options for both permanent and temporary repairs often exist for composite structures. The lack of environmental resistance and long-term durability for most temporary composite repairs dictates that permanent repairs be made before exceeding time limits. A need for temporary composite repairs in the field relates to the long aircraft downtime for some permanent repairs, such as those involving laminate patch curing and bonding.

Technicians, also known as mechanics, make actual composite repairs. They must work to approved repair documentation and must not exceed any limitations provided by the documentation, OEM, or Federal Aviation Administration (FAA) DER. Technicians must always use approved materials, equipment, parts and procedures, and provide proper documentation in various forms, such as the component master worksheet and materials record sheet.

Mechanics must be trained in composite repair processes, including associated tooling and equipment. In addition, technicians generally need good hand-eye coordination.

With a primary responsibility for inspecting finished parts, inspectors need to understand the composite materials being used and the construction of each part, which requires the ability to read and interpret engineering drawings and approved repair documentation. Inspectors must be familiar with the operation and use of NDI equipment to evaluate the existence of defects and verify the structural integrity of composite parts. Inspectors are responsible for verifying the traceability of materials and personnel is described in routine work documents, such as inspection reports and warranty investigation reports.

Inspectors must be trained in the use of a variety of inspection techniques for composites. In addition, inspectors must have good eyesight and hearing.

Engineers must have sound training in composite materials to design repairs and to liaise with OEMs as necessary. They must be able to understand aircraft drawings, authorized repair manuals, and all types of approved repair documentation. Engineers also need to be able to suggest alternative materials when necessary and be able to explain product specification requirements to technicians, inspectors, and managers. Specific topics that they must understand are the critical aspects of structural design, radome thickness related to radar signal wavelength, and the importance of lightning protection, erosion protection, weight and balance, mechanical clearance, and drainage systems. Finally, engineers should be able to write workshop procedures to ensure quality work and understand the defect types that can occur, the importance of drying procedures, and the NDI methods required to detect and evaluate defects.

Engineers require a minimum of a Bachelor of Science engineering degree, or equivalent, at an accredited academic institution, including some training in aircraft composite structural design and analysis. The latter training is best received while gaining industry experience. Engineers also need a good understanding of the regulatory documents and procedures to follow if damage or repairs are beyond those given in source documentation.

Routine and nonroutine operations must be documented. The inspection or repair process is predetermined, and instructions often are formally printed and include sign-off sections for technicians, inspectors and engineers. Important records include:

- Component Master Worksheet
- Materials Record Sheet
- Inspection Report
- Technical Instruction

A.1.3.8 Discuss the Importance of Documenting and Sharing Information About Damage Scenarios Discovered in Service Between OEM, MROs, and Regulators.

Incident reports have historically been required for any defect that occurs in flight, e.g., an engine failure, a cracked window ply, or any similar defect.

Defects found on the ground during maintenance have to be reported in Maintenance Occurrence Reports (MOR).

It is important that these reports are sent to the OEM and the regulatory body without delay.

As an example, if a specific composite part is consistently or easily damaged, these reports can provide the impetus for the OEM to modify the part to be more durable.

More serious problems and their corrective procedures can result in mandatory Service Bulletins (SB) with time limits within which inspection or repair must be carried out.

Minor faults can be corrected by SBs where the adoption is optional. A fault during inspection must be submitted via MOR.

A.1.4 BE AWARE OF COMPOSITE MAINTENANCE RESEARCH AND INDUSTRY DEVELOPMENTS.

A.1.4.1 Explain the General Lack of Standards Available for Composite Engineering Practices and Realize That Composite Maintenance and Repair Requires Specially Acquired Skills.

As a relatively new technology, composite material development and practice continues to evolve. This section will provide an overview of some of the key advances and stimulate further research into these advances.

The innovative use of composite materials is enabling commercial and military aircraft to perform at higher levels. Some of the advantages compared to metal structures include higher strength-to-weight ratios, improved corrosion resistance, and custom design based on directional strength characteristics of composites.

Properties and use of composites involve anisotropic strength characteristics, the interaction among different composites joined together by bonding or bolting, and the development of proprietary materials. In commercial aerospace, the intense competition among airframe manufacturers, notably Airbus and Boeing, has restricted information flow.

Next-generation design and manufacturing technologies for large composite structures is being researched and developed. For example, reduction of part count is a central theme in lean manufacturing. Composites promote lean-manufacturing concepts through the significant reduction in part count by cocuring components into assemblies through reduced numbers of processes employed.

A primary aim of composites' research is to develop processes that reduce production costs. New processes to automate fiber placement and new adhesive systems are being investigated, among others.

As the percentage of composite materials dramatically increases in new aircraft development, key primary structures are being designed using composites, such as wing spars in a large aircraft wing for the Airbus A400M. Improved toughened resin systems, usually proprietary to the airframe manufacturers, continue to be developed to improve strength characteristics of composite matrices.

Evolving composite technologies are frequently considered proprietary and not available to the public domain. As a result, composite property standards are in the early stages of development and in flux, requiring special skills and awareness of safety implications of composite maintenance and repair.

Methods of inspection and property materials, if considered proprietary, may make detection, disposition, and repair more difficult due to nonstandardized practices and documentation. Underlying damage to composites, unlike aluminum, may not be visible, because the composite

bounces back after impact or the visible surface of the composite may exhibit minor damage while the back side, nonvisible surface, may have extensive damage. Further, other supporting structures, having absorbed much of the impact energy, may show significant damage compared to the visible composite surface.

A.1.4.2 Discuss Emerging Advances in Repair Process Technologies may Appear for Bonded and Bolted Repair and Quality Control.

A.1.4.2.1 Process Technologies.

Advances are primarily focused on bonded repair. For existing materials, the upper service temperature is related to the T_g, which in turn, is usually related to the cure temperature. New process technologies that result in lower-temperature cure while retaining the same upper usage temperature would be desirable, especially in field repair situations. Some film adhesives and paste adhesives can be cured for longer times at lower temperatures, while still achieving a full cure. Others have to be cured in a narrow band of temperature.

The following are some newer or improved repair materials:

- Cytec FM 300-2 film adhesive is a 120°C (250°F) curing version with a service temperature of 150°C (300°F).

- Huntsman Epocast® 52 can be cured at 65°C (150°F), but has a service temperature of 177°C (350°F).

- Hysol EA 9390 has been very successful for many years and can be cured at 93°C (200°F) to give a service temperature of 177°C (350°F).

- Hysol EA 9394 is a paste adhesive that can be cured at 25°C (77°F) for 5-7 days or at 66°C (150°F) for 1 hour to give a service temperature of 177°C (350°F).

- Hysol EA9396, a low-viscosity laminating resin, can be cured at 25°C (77°F) for 5-7 days or at 66°C (150°F) for 1 hour to give a service temperature of 177°C (350°F).

- Hysol EA 9396/C-2 can be cured at 93°C (200°F) for 1 hour to give a service temperature of 204°C (400°F).

- These materials meet the desired attributes of a low cure temperature with a high T_g. Some of them, however, do not achieve the strength and stiffness properties of the original part materials that they are intended to repair.

- 3M has produced a water-based primer (EW 5000 AS), a composite bonding film (AF555), and a new two-part paste adhesive (EC3333B/A), but no upper service temperature is given on the data sheet. A number of new potting compounds with low smoke and toxicity have also been released.

A.1.4.2.2 Nondestructive Inspection.

In general, NDI testing can detect only delamination, fluid ingress, porosity, and disbonds, not the strength of adhesive bonds.

Some newer NDI techniques are listed below.

- Visual and tap tests used for secondary structures may have to be enhanced as composite applications expand into primary (load-bearing) structures.

- Increasing automation in a number of NDI technologies, including ultrasonics, thermography, shearography, microwaves, and digital radiography.

- Mobile automated scanners (MAUS) automate ultrasonic and eddy-current scans.

- The computer-assisted tap test (CATT) produces two-dimensional color images of possible flaws.

- Mechanical impedance analysis (MIA) probes excite local surfaces and test them for stiffness.

- New NDI techniques showing promise for assessing adhesive bond strength include nonlinear ultrasonics, shear-wave ultrasonics, vibrothermography, stress-wave interrogation, and laser ultrasonics.

- Vacuum pulling against composite surface to assess bond strength of skin laminate to honeycomb, such as sandwich structure.

A.1.4.3 Discuss Emerging Damage and Repair Inspection Technologies, Such as Bond Testing, Moisture Meters, Interferometer (Three-Dimensional Characterization).

Repair work usually requires equipment that is inexpensive, portable, easy to use, and locally available. More sophisticated equipment tends to be too expensive to maintain at a large number of locations.

The tap coin, the washer, and the small tap hammer are inexpensive and simple to use. Unfortunately, these simple methods cannot provide a permanent record of the damage. More sophisticated and automated methods can print a copy of the damaged area, which is useful and provides a permanent record of the damage found.

A.1.4.3.1 Bond Testing.

- Staveley makes the Bondmaster™ 1000 bond tester, which uses probes for pitch/catch, resonance, and MIA techniques. This portable instrument weighs only 3-6 pounds, depending on the batteries used. It will detect flaws in metal/metal bonds, graphite/Nomex®, graphite/graphite, and metal/honeycomb bonds. With three probes, it has all the methods previously included in one instrument.

- The Fokker Bond Tester Model 90 is still in use and can be found on the website of Stork-Fokker Aerospace and The Technical University of Delft. The Fokker Bond Tester uses the principle of ultrasonic resonance testing to determine the through-thickness acoustic resonance frequency and amplitude of a ply or laminate with just one transducer. The technique is especially suitable for thin laminates in which it can detect air bubbles, delamination, and porosity (better described as voids). It can be used on relatively thick laminates but can give no depth information. Small defects, approximately 6 mm or larger, can be detected.

- Future bondline characterization may be conducted by the advent of a Laser Bond Inspection system that is currently under development. This system uses a laser pulse to excite the composite, while a scanner measures the composite's response to the pulse. Current methods can only measure the void content of bondlines, but it cannot determine how strong the bond is between the adherences. Laser Bond Inspection systems are meant to verify that the bondline meets the strength requirements rather than just measuring bondline regularity. When applied to hot-bonded repairs, weak hot-bonded patches give a distinct response to the laser pulse that can be used to detect unacceptable repair processing.

A.1.4.3.2 Bond Strength.

Although bond strength cannot yet be measured, other very useful information can be obtained from these methods.

The older, nonautomated methods relied on human memory of one sound compared to another. It actually works remarkably well, but it cannot give numerical values to the results.

The new methods include:

- Ultrasonic and x-ray NDI equipment. These methods complement each other. X-rays are better at finding cracks normal to the plane of the skin and finding water in honeycomb structures, whereas ultrasonics are better at finding disbonds and delamination between the plies of composites and between the skin and core of sandwich panels.

- Panametrics offers a portable ultrasonic system. The Epoch LT is an affordable hand-held flaw detector.

- RD Tech, now part of Olympus NDT, offers their Portable Phased Array Ultrasonic system known as Omniscan.

- NDT Solutions offer a similar system called Rapidscan.

- Of considerable current interest, for the Airbus A380, is the endoscope method that can be used with GLARE, which is a lamination of plies of aluminum alloy and glass fiber. This endoscope method is a small instrument that uses visible light to create an optically magnified image on a monitor. It is used to detect small surface defects that are invisible

to the naked eye. Endoscopes can access places that are difficult to reach and inspect. This method can only be used for surface-breaking cracks. Contrast is important, therefore black materials, such as carbon fiber composites, are far more difficult to inspect for cracks.

- Tap test methods work well for thin laminates, honeycomb, and other sandwich panels but are less effective on thicker parts. The tap test is often performed with a medium-sized coin, a steel washer approximately 25 mm in diameter, a small tap hammer (shown in figure A-5), or the Mitsui Woodpecker WP-632DS. The Woodpecker is connected to a computer that records the sound of a good section of panel and then compares it to subsequent readings in other areas that may contain disbonds. The latest version can print a C-scan of the damaged area. In the future, the Woodpecker will be designed to work with thick laminates. The Woodpecker is lightweight and easy to use. Three similar tap test methods are the Boeing tap hammer; the Wichitech RD3, developed by Boeing and licensed to Wichitech; and the CATT, developed by Iowa State University and licensed to Advanced Structural Imaging, Inc., which has an automatic tapping carriage that eliminates any effects due to a human operator.

Figure A-5. Picture of Military Specification Tap Hammer
(Courtesy of Abaris Training Resources Incorporated)

A.1.4.3.3 Moisture Meters.

Most moisture meters rely on radio frequency dielectric power loss to detect moisture, which is attributed to an increase in the conductivity of the composite due to moisture absorption. However, carbon fiber, now the most commonly used fiber in aircraft structures, is electrically

conductive itself. Therefore, contact meters can only be used on structural components that do not contain carbon fibers.

Moisture detectors emit electrical energy and cannot be used in the presence of metals or other conductive materials, including antistatic coatings containing carbon. Panels having such coatings must be tested from the back side, which typically is not coated with antistatic paint. Caution must be exercised with panels that contain buried metallic doublers. These doublers will give a false indication of moisture and may cause a panel to be removed needlessly. Conversely, if moisture is genuinely present in the area of the metal insert, this type of meter cannot positively identify it. Typical moisture meters are the A8-AF from the Moisture Register Products Division of the Aqua Measure Instrument Company and the M.49/P from J.R. Technology. J.R. Technology also supplies the MW 105 moisture meter, which is the only digital (microwave) moisture meter in the world. It corrects the readings for the density of the material. This is the most accurate system known, as the measurement enables the density to be calculated and accounted for in the readings.

Methods for detecting moisture in carbon fiber composites work on different principles, but they have proved to be very expensive and are no longer available. It is often easier to use a drying process that has been proven to be effective. This is usually a specified drying temperature for a specified period of time and is given in the SRM or other authoritative source.

A.1.4.3.4 Interferometric and Shearographic Methods.

Holographic interferometry techniques have been extensively investigated in recent years. They can detect loose rivets, cracks under rivet heads, and weak adhesive bonds in metal structures. In composites they can detect heat damage, impact damage, weak bonds, and delamination in sandwich structures.

The holography method provides an object image from the properties of reflected light using their intensity, wavelength, and phase. Phase provides the three-dimensional effect. The light source must be coherent, in that it is monochromatic and has simultaneous emission, thus lasers are used. Interference occurs when an object changes its relative position and, after double exposure to laser light, the light's phase has changed. The double exposure occurs before and after object movement. This can be produced by lightly loading the component mechanically or by applying localized heating. The method works well on both metal and composite structures. The method can detect loose rivets, cracks under rivet heads, and weak adhesive bonds in metal structures. In composites, it can detect impact damage, heat damage, weak bonds, and delamination in sandwich structure.

Holographic interferometry measures purely out-of-plane displacement. Another type of interferometry is called shearography. This measures the first derivative of the out-of-plane displacement or the slope of the deformation.

A.1.4.4 Discuss Advances in Composite Repair Analysis and Design, Which may be Used by Engineers to Develop a Repair for Aircraft Structure.

Development of techniques for improving repairs and the assessment of the repair effectiveness is ongoing. This overview is not comprehensive, but does provide a summary of issues being investigated.

A.1.4.4.1 Major Sources of Damage to an Aircraft.

For elevated components, lightning strike and maintenance scaffolding impacts pose a danger to composite material components. Honeycomb composites comprise nearly 90% of damage, by some estimates, and can include moisture ingress, heat, hail, and impact. Heat can damage engine nacelles, and trailing edges of ailerons are vulnerable to damage. For aircraft control surfaces, such as flaps, most composite damage is by impact, rather than environmental, such as stone damage from wheels and jet pipes. Delaminations can be caused by water ingress into the composite structure.

A.1.4.4.2 The CACRC Involvement.

The CACRC is a joint project of manufacturers, airlines, repair stations, and regulators. The CACRC, a committee under the auspices of the SAE, is actively developing standard composite repair practices and training that will be available to the industry, which can be regarded as standard practice. Documentation can be obtained through the SAE website, www.sae.org.

Recognizing the importance of composite maintenance and repair, since 1992 the FAA has been actively involved in facilitating and funding research that is directly related to the primary causes of aircraft damage. The following list is not comprehensive, but provides an overview for understanding the principal areas of concern in the area of composites' analysis and design.

- 1996—Published AC-145-6 "Repair Stations for Composite and Bonded Structure"

- 1996—Stanford University funded a small study to address repair material and substrate interchangeability

- 2000—Study "Bonded Repair of Aircraft Composite Sandwich Structures," FAA report DOT/FAA/AR-03/74

The results from one study, published in report DOT/FAA/AR-03/74 by the FAA, drew the following conclusions:

- Repairs by repair depots to picture frame shear elements restored at least 90% of the average pristine strength, with one exception

- With the occurrence of a heat-blanket failure, the resulting repair also failed consistently

- Field repairs can be considered to be equivalent to autoclave prepreg repairs, under the correct conditions

- Successful repairs require trained personnel; training is critical

- All the studied NDI field methods underestimated the damage size, with the tap hammer being the least conservative. The limitations of field inspections are a critical element of good repair in practice and must be understood.

A.1.4.4.3 Ongoing Research Activities: Solid Laminate Repairs and Criticality of Various Steps in the Repair Process.

Research is concentrating on several aspects of the repair process. Key elements include:

- Existing CACRC standards are being evaluated, as related to technician skill level, using different repair geometries

- Evaluation of the degree to which a repair is linked to the repair technician's skills and knowledge

- Assessment of the quality and reliability of composite repair via testing to provide indications as to the level of criticality and degree of training

- Quantitatively determine the effect of deviations from repair process on the adequacy of repairs

- Development of a check list to interrogate critical points in the repair process for FAA inspectors

- Establishment of the value of existing CACRC standards for composite repair technician qualification

- Development of a sensor schematic to detect and guide inspection actions (Northwestern University)

- Sandwich structure repairs and criticality of various steps in the repair process

- OEM and CACRC repair techniques

- Investigation of the effects of surface contamination and improper curing

- Effect of scarf ratio and repair material

- Damage tolerance of impacted repairs

- Investigating analytical methods for repair

A.1.4.5   Outline the Process for Repair Structural Substantiation and Approval to Meet Regulatory Requirements.

A.1.4.5.1 Regulatory Requirements.

Repair of composite structural parts must be in compliance with the airworthiness regulations associated with various categories of aviation products. These regulations prescribe the structural flight and ground loads and the operating environment, including design, strength, and durability requirements for those loads and environment. The regulations for the approval of airplanes and helicopters for the U.S., Europe, and Canada are defined by various regulatory agencies worldwide.

All composite inspection and repair design, materials, and processes must be substantiated and approved by meeting the requirements of the airworthiness regulations. Composite inspection and repair designs, materials, and processes that have not been previously substantiated and approved must follow the regulatory processes for structural substantiation and approval according to a set of specific criteria, such as structural loading on, and strength of, a composite component.

In defining a repair to a damaged composite structure, the primary task in the approval process is to establish those parts of the regulations that will be impacted by the damage and repair, and then demonstrate that the repaired structure still complies with the regulations.

For the safe use of composite materials in the manufacture and repair of aircraft structural parts, the FAA, European Aviation Safety Agency (EASA), and other regulatory bodies require compliance with the airworthiness standards associated with various categories of aviation products. These requirements are contained in the regulations. When a composite structure is damaged and repaired, the repair materials, process, and design must comply with these regulations. Normally, the structure repair should return the repaired airplane to the same condition prior to damage. However, sometimes this is not possible to do, and the airplane may have some limitations imposed because of the repair, i.e., a new weight limit.

A.1.4.5.2 Regulatory Authority Guidance.

The FAA and other civil aviation authorities issue guidance that provides information, requirements, and methods for showing compliance with the airworthiness requirements for composite structures. An Advisory Circular (AC) is one example of guidance documentation issued by the FAA.

A.1.4.5.3 Approved OEM Documentation.

Proprietary repair is available through the aircraft manufacturer's SRM. The SRM is the primary source of repair practice. If a specific event is not covered in the SRM, then either the manufacturer or a DER should be contacted for more information.

Repairs for primary load-bearing and secondary structures must be described in the manufacturer's SRM or be confirmed by a DER.

Repair instructions defined in an SRM are based on material and processing qualification tests that ensure repeatable material properties and processes. These instructions, which are approved by the civil aviation authority, such as the FAA or DER, must be followed. If they are changed, the changes must be evaluated to establish the impact of those changes and adjust the repair accordingly. Structural repair materials are subject to material specifications that detail storage and handling instructions to ensure that, if processed per the SRM or other repair instructions, a repair will meet the specified design strength, stiffness, and durability.

A.1.4.5.4 Approval of a Repair to an Aircraft Structural Component.

Repair approval requires the following:

- The repair was designed by a qualified engineer who used approved data and materials to restore the part's original strength and stiffness. Repair design criteria for permanent repairs are fundamentally the same as those used to design the original part. Repair design criteria for temporary repairs can be less demanding, but may approach permanent repair status if the temporary repair is on the aircraft for a considerable time (e.g., an interim repair).

- The repair materials were stored and handled within the bounds of the approved documentation. The use of improperly stored or handled adhesives, sealants, prepregs, or wet lay-up ingredients may result in structurally unsafe aircraft components. AC 145-6 defines repair station requirements for approved repair material purchasing and QC (e.g., testing, storage, and handling).

- The repair was processed per the approved process instructions, and the required in-process controls were carried out. It is essential that the approved in-process controls be strictly carried out. All equipment used during repair processing, including damage and repair inspection equipment, must be certified and maintained to the required specifications.

- The material and in-process records and the postrepair inspection have been judged acceptable. Once a repair is completed, it must be inspected and approved by an authorized maintenance organization inspector before the part is returned to service.

A.2 TEAMWORK AND DISPOSITION.

A.2.1 UNDERSTAND ROLES AND RESPONSIBILITIES.

A.2.1.1 Describe Critical Safety Aspects of Repair Design, Process Planning, and Approval.

A.2.1.1.1 Repair Design.

Qualified engineering personnel must use approved information and data to design repairs. The intent of structural component repairs is to restore the original strength and stiffness, regardless of whether the repair is to be bonded or bolted, as the operator is not privy to the design loads. This requires the knowledge of the strength and stiffness data of the original material and the strength and stiffness of the repair materials. If the bolted repair is chosen, bearing data of the repair material and fastener characteristics must be known. All original strength and stiffness data used for repair designs must be derived from the database used for the original type certificate. All data used for repair materials must be approved by the appropriate authority or a DER. Repairs to moveable control surfaces must consider the effects on the overall part stiffness, weight, balance and flutter characteristics. The repair designs in approved documents, such as an SRM, are not to be extended to components other than those specified.

Within most major OEM structural repair manuals, repairs require an evaluation for damage tolerance capability and are classified as Category A, B, or C.

- A Category A repair is a permanent repair for which normal planned inspections are sufficient, and no other actions are necessary.

- A Category B repair is a permanent repair for which supplemental inspections are necessary at specified thresholds and repeat intervals.

- A Category C repair is a time-limited repair for which supplemental inspections are necessary, followed by a replaced part or reworked repair within a specified time limit.

Principal Structural Elements (PSE) are primary structural parts that are considered critical to flight safety. Examples of PSEs are wing main torque box spars, skins, and stringers; fuselage skins, stringers, and frames; horizontal and vertical main torque box spars, skins, and stringers; and wing trailing-edge flaps. Repairs to PSEs must comply with the same damage tolerance requirements as the original part. For bolted metal patch repairs to composite laminate-stiffened structural PSEs, a fatigue analysis is required for full damage tolerance compliance. Repairs outside OEM structural repair manuals applied to primary structure parts are also required to be evaluated for damage tolerance capability.

Repairs that are not critical for the damage tolerance capability of the aircraft are classified as permanent, interim, or time limited, based on the expected durability of the repair.

In the event that an approved repair design is not available, the maintenance engineer has several options.

- Contact the OEM for an approved repair. In this case, the damage evaluation will be transmitted to the OEM, and a specific repair will be designed.

- Design a specific repair for the damage. In some instances, such as damage to a PSE not covered by the SRM, an adequate damage disposition or repair design will require evaluation by the OEM.

- Replace the damaged part

A.2.1.1.2 Repair Process and Inspection Planning.

The following documents are part of the maintenance and repair process, and must be either consulted for damage inspection or repair instructions, or filled out to maintain records of repaired components and repair materials.

- Maintenance planning data (MPD) document, SRM, and component records. For planned maintenance events, such as medium or heavy maintenance checks (e.g., C or D checks), the MPD document must be reviewed for directed inspections. For components that have temporary, interim, or time-limited repairs, the records must be reviewed for flight cycles experienced in order to know if the repairs need to be replaced with permanent repairs. Many approved documents, such as the SRM, have restrictions for proximity of repairs and damages. Therefore, when damages are found, the component records must be reviewed for previous repairs or previously allowed damages. The SRM will contain inspection, allowable damage limits, and repair instructions that are based on approved databases. If the damage and repair designs are available from the approved data, there is no need to communicate with the OEM or other DERs for instructions.

- Routine work documents. Routine work documents are used for planned maintenance tasks. The inspection or repair processes are predetermined, and often instructions are formally printed, along with sign-offs for technicians, QC, and inspection personnel. The following are examples of documentation required by Title 14 Code of Federal Regulations (CFR) 145 in the U.S. and EASA Certification Specifications (CS)-145 in Europe:

  - Component Master Worksheet: The part number, defect or damage description, and the repair action taken are recorded on the worksheet.

  - Materials Record Sheet: The material record sheet provides details of all materials used, including part, serial, and batch numbers. The material batch numbers allow traceability of repair parts or materials.

  - Component Record Card: This card shows the part number, serial number, and details of each component. This will enable future work to be related to previous repairs on the same part.

  - Inspection Record: This form provides information for a report after a component has been inspected. The part may be new, damaged, or repaired.

- Technical Instruction, AB 110: This form provides specific instructions for the performance of technical processes to ensure that adequate information is available to those performing the work.

- There are other documents required such as warranty and investigation reports, reject notes, stock record cards, and unserviceable tags.

A more comprehensible list of required documentation can be found in "Care and Repair of Advanced Composites," second edition, by Keith Armstrong, L. Graham Bevan, and William F. Cole.

When damages are detected during routine inspections in the maintenance depot, the instructions within the SRM must be strictly followed, and component and material records must be maintained.

When damage is found on the aircraft while on the ramp during normal operations, the SRM may not be available, but the damage must be reported to a maintenance engineer for a disposition. Also, the above work documents may not be available on the ramp, but records of damages, inspection results, repair dispositions, and materials used must be kept for entry into the above documents.

Because approved repair documentation is unlikely to be available on the ramp, it is essential that operations personnel have access to qualified maintenance engineers so proper damage or repair dispositions can be made. In the event the damage is beyond SRM repair limits, it is also essential, for expeditious repair dispositions, that the maintenance engineers have access to a DER or qualified OEM personnel. Some OEMs have service engineering personnel available on a 24-hour basis to be available for operator queries. For small maintenance and repair organizations dealing with general aviation (GA) aircraft, the specific roles of engineer, technician, and inspector may all be performed by one person. In these cases, the qualifications are still the same; maintenance personnel must be qualified for the roles they assume.

There may be situations where the damage is not discovered by ground personnel, but an incident occurs with the knowledge of flight or other operations personnel that may involve damage to aircraft structural components. Such incidents, as bird impacts during flight, hard landings, engine or tire bursts, large runway debris thrown by the engines, or ground vehicle collisions, can cause severe damage to aircraft components. It is essential in the event of such incidents that the aircraft be inspected for any resulting damage as soon as possible before the next flight.

A.2.1.1.3 Related Documents.

SAE Reference (Commercial Aircraft Composites Repair Committee—CACRC)
AIR5946: Design and Application of Composite Repairs for Thermosetting Composites

A.2.1.2 Describe the Steps in Composite Damage Detection, Inspection, and Repair Processes.

- Detect the damage either by an operations technician on the ramp, by a technician during a heavy maintenance event, or by an inspector during a directed inspection per the Aircraft Maintenance Manual (AMM) or MPD. Note that some composite parts, particularly primary structural components, are designed for visual detection to initially discover damage.

- Assess and map the extent of the damage. Because visual inspection cannot determine the extent of any internal damage, use an instrumented NDI procedure, if available. A tap test method may be used if a defect or damage that is less than or equal to the maximum damage size allowed by the approved repair documentation can be found. If damage is found on the outside surface, inspect the inside surface, if accessible.

- Review available documentation, e.g., the OEM's SRM or other approved repair documentation. If none is available, consult with a qualified operator or MRO maintenance engineer, or consult the OEM for instructions.

- Compare damage before cleanup to the SRM allowable damage limits (ADL) for the appropriate zone of the specific component.

- If the damage is within the ADL, the damage is to be sealed per the instructions in the SRM, and the component (or aircraft) can be returned to service. Before sealing the damage, all contaminates and water must be removed from the component per the specific approved repair instructions.

- If the damage is beyond the ADL, then the damage is to be cleaned up, i.e., any damaged materials are to be removed, including damaged honeycomb core, if present, and any loose or broken fibers.

- Compare the cleaned up damage to the SRM repair damage limits for the appropriate zone of the specific component.

- Choose a repair method from those listed in the SRM repair section for the specific component.

- For a vacuum bag cure bonded repair of the outside facesheet and core of a sandwich component using prepreg material, the following steps are condensed from typical detailed instructions in an OEM structural repair manual.

  - All contaminates and water must be removed from the component using vacuum and heat.

  - Remove the protective coating (e.g., conductive coating, if present, paint enamel, and primer) using a prescribed method such as abrading or sanding.

-   Make the necessary taper to the damaged plies for the bonded, scarfed repair, using a prescribed taper ratio (e.g., 50 to 1).

-   Clean the abraded surface and tapered area with a soft cloth moistened with an approved solvent.

-   Prepare and clean a core plug the same size as the cutout in the facesheet.

-   Install core plug with foaming adhesive and cure using a vacuum bag, thermocouples, and heat blanket.

-   Sand the core plug to an acceptable height so the repair adhesive and plies will lay down smoothly.

-   Prepare the repair ply material and film adhesive (i.e., remove from freezer and allow to warm up). Cut out the required number of replacement and additional repair plies and film adhesive.

-   Place film adhesive down first, then each replacement repair ply, and finish with additional ply (or plies) over the tapered repair. Sweep each ply to remove any wrinkles.

-   If required by SRM or approved repair documentation, compact the repair plies using a temporary vacuum bag.

-   Place a parting film over the entire repair and install the vacuum system, a minimum of three thermocouples, the required surface bleeder cloths, and the heat blanket. Apply the vacuum seal around the repair.

-   Apply a vacuum of 22 inches of mercury, and cure the repair per the specified cure cycle. Ensure that the vacuum remains within allowable limits, then monitor the heat-up and cool-down rates, dwell temperature, and vacuum pressure throughout the cure cycle.

-   After the repair has been cooled down to a prescribed temperature, remove the vacuum pressure.

-   After cooldown is completed, remove the heat blanket, breather cloths, thermocouples, vacuum seal and bag, and parting film.

-   Inspect the repair for voids and anomalies using approved inspection methods. Make sure to inspect the area around the bonded repair, up to 6 inches away from the edge of the repair.

-   If inspection proves the repair to be satisfactory, restore protective coatings over the repaired area per the approved documentation and return the component to service. Some composite components are protected from excessive lightning strike damage by systems such as flame spray coatings, aluminum mesh, or

picture framing. If damaged or removed during a repair, these protection systems need to be restored per the SRM or other approved documentation. Graphite composite components that contact aluminum parts are isolated by corrosion prevention systems such as a layer of glass epoxy. If these isolation systems have been damaged or removed during the repair, they must be restored per approved documentation.

For a bolted repair of a laminate-stiffened carbon fiber component using metal repair plates and fasteners, the following steps are condensed from typical detailed instructions in a typical OEM SRM.

- Ensure that the damaged area and adjacent surface of the part are smooth and flat for the repair doubler.

- Clean the area with an approved solvent.

- Seal the damage as applicable.

- Select a repair doubler of the required thickness with the specified surface finish.

- Mark the fastener pattern on the repair doubler, place the doubler in a fixture, if available, and pilot drill all the fastener holes in the doubler.

- Place the doubler on the component to be repaired, ensure that it does not move and pilot drill the fasteners holes in the composite part. Move to opposite sides of the fastener pattern for each hole to be drilled. Install a temporary fastener in each hole after drilling to ensure that the doubler and part do not move.

- Remove the doubler and place in a fixture. Drill all piloted holes to a diameter that is 1/16 inch smaller than the final hole diameter, and remove all burrs.

- Drill all pilot holes in the composite part to a diameter that is 1/16 inch smaller than the final fastener hole diameter, and remove all burrs.

- Place the doubler on the composite part, aligning the fastener holes, and install a temporary fastener in every other hole to ensure that opposite holes are clamped on each side of the symmetry line.

- Ream all holes to full size.

- Remove the doubler and deburr all the holes on both the doubler and composite part. Chamfer edge of all holes on the fastener entrance side of the doubler to the same diameter as the radius on the underside of the fastener heads.

- Apply one coat of sealant to the mating surfaces of the doubler and composite part.

- Place the doubler over the composite part, aligning the holes in each part. Install temporary fasteners in each corner of the fastener pattern, and then install temporary fasteners in every other hole so that opposite holes are clamped on each side of the symmetry line.

- Select appropriate fasteners of the correct diameter and grip lengths.

- Install fasteners in the open holes through the squeezed out sealant.

- Remove all the temporary fasteners, and install the permanent fasteners in the open holes through the squeezed out sealant.

- Inspect the fasteners to see that they are correctly installed. Inspect the back side of the repair to see if the fastener sleeves are satisfactorily installed.

- Remove and replace any fasteners found to be incorrectly installed.

- Apply a fillet seal around the repair doubler.

- Restore protective coatings over the repaired area per the approved documentation and return the component to service.

A.2.1.2.1 Related Documents.

SAE References (Commercial Aircraft Composites Repair Committee—CACRC)
ARP4977: Drying of Thermosetting Composite Materials
ARP4991: Core Restoration of Thermosetting Composite Components
ARP5143: Vacuum Bagging of Thermosetting Composite Repairs
ARP5144: Heat Application for Thermosetting Resin Curing
ARP5319: Impregnation of Dry Fabric and Ply Lay-Up
ARP5256: Mixing Resins, Adhesives, and Potting Compounds
ARP5701: Lay-Up of Pre-Preg Composite Materials

A.2.1.3 List Basic NDI Methods Used for Damage Assessment and Postrepair Inspection.

Damage and repair assessments are important aspects of repair work. Several methods of damage and repair assessments are typically available in both the composite-manufacturing and repair arenas. A number of these techniques are discussed in sections A.2.1.3.1 through A.2.1.3.10.

A.2.1.3.1 Visual Inspection.

Visual inspection is the first and most obvious method used for damage detection. A composite aircraft's primary structural components are typically designed so that damage too small for visual detection is considered noncatastrophic, and therefore, component static and fatigue strengths and stiffnesses are assumed to be sustainable for the life of the aircraft. However, if surface damage is detected by visual means, there is the potential for hidden damage, and other

NDI methods may be required for a more complete damage assessment. Apart from directed NDI of specific components or specific areas of components, visual inspection is the current cornerstone of airline maintenance of composite structural components. If any damage is discovered, however small, it must be investigated, and the SRM (or equivalent documentation) must be consulted for appropriate action.

Visual inspection is also the first and most obvious method for postrepair inspection. Repairs using some materials, such as glass fiber-reinforced plastic (GFRP), are more easily visually inspected for repair defects, such as voids and delaminations, due to the translucent nature of the finished repair patch. However, carbon fiber-reinforced plastic (CFRP) repairs do not allow reliable visual detection of anomalies, except for excessive adhesive bleed-out and fillets. If no defects or anomalies are discovered in a postrepair visual inspection, the repair must still be inspected by either the tap method or by using pulse-echo (P/E) equipment. For most repairs, visual inspection is the precursor to the more reliable NDI.

The Advantages of visual inspection are as follows:

- No expensive equipment is needed.

- Airworthiness design philosophy is such that most damage that is of concern can be found visually.

The disadvantages of visual inspection are as follows:

- Inspectors have difficulty in maintaining concentration over large areas.

- Composite parts with small visual damage may have extensive nonvisible damage. This is particularly important in the case of nonvisible delaminations. A critical failure mode of structures fabricated from composites is compression, and any delaminations can reduce the compression strength and stiffness of a component. In addition, high-energy blunt impacts may cause damage to substructure, which may not be externally visible.

A.2.1.3.2 Tap Test.

Most commercial and military aircraft operators use the tap test for damage detection in composite components. Tap testers range from a simple coin, to a tap hammer, to more complicated, automated tap hammers.

The basic tap coin or hammer relies (figure A-6) on the human ear to detect changes in frequency. For example, a good bond or nondamaged part will emit a clear, high-frequency sound when gently tapped, while a disbonded or damaged part will emit a dull, lower-frequency sound.
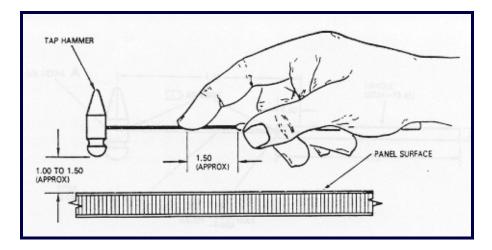
Figure A-6. Illustration of Military Specification Tap Hammer
(Courtesy Abaris Training Resources Incorporated)

The automated tap hammer does not rely on the efficiency of the human ear, which deteriorates with age and varies from person to person. The method may be described as audio sonic, because it operates in the human-hearing range. The advantage of the automated tap tester is that a change in frequency at which a defect is considered to exist can be set and the area mapped accordingly. The automated tap tester is considered more accurate because it is less reliant on the human ear.

In general, the tap test works well for detection of damages in thin skins of any type. The method is especially useful on sandwich structure with thin composite facesheets and honeycomb core. It can work on thicker facesheets or solid composite laminate structure if the first few plies are delaminated, but it cannot detect defects deeper in the laminate. Similarly, metal-skinned honeycomb parts or bonded metal doublers above a thickness of approximately 1 mm (0.04 in.) do not respond with a change in frequency if disbonded.

As for damage detection, most aircraft operators use the tap test for postrepair inspection of sandwich parts. It is simple, inexpensive, and quite reliable when used by experienced inspectors. The method has the same limitations for postrepair inspection as for damage detection. Changes of frequency are not obvious for deep delaminations within the repair or for anomalies in the bondline if the repair patch is more than a few plies thick.

The advantages of the tap test are as follows:

- It is simple and inexpensive.

- A tap test provides a quick initial method of investigating the extent or existence of a defect.

- A tap test can be used to reliably detect delaminations and disbonds in thin composite skins, and severe moisture within sandwich parts.

The disadvantages of the tap test are as follows:

- It is impractical to cover large areas, because it is difficult for an inspector to maintain concentration.

- The tap test can be highly subjective, although the automated tap testers significantly reduce this subjectivity.

- A tap test cannot locate small defects such as voids or minor moisture ingression.

- It is not effective on thick skins, and its effectiveness may be reduced when inspecting parts covered by protective coatings (e.g., lightning protection systems).

A.2.1.3.3 Ultrasonic Inspection.

Ultrasonic inspection uses ultrasonic or stress waves to study materials or structures. This type of inspection uses an ultrasonic signal that measures the attenuation of the signal. There are two types of ultrasonic inspection techniques typically used:

- Through transmission mode, using two transducers
- P/E, using a single transducer

In either mode, the transducer(s) must be coupled to the structure via a liquid or solid medium because of the severe impedance mismatch between air and solid materials (see figure A-7).
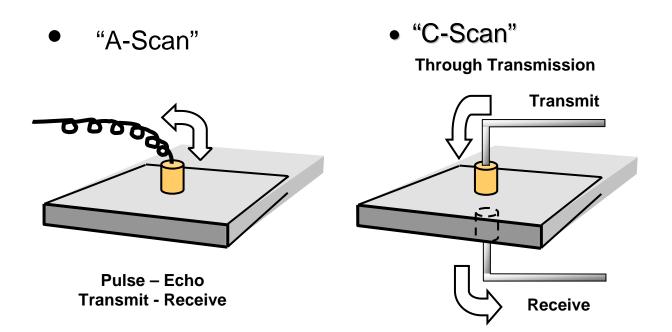

Figure A-7. Ultrasonic Inspection: A-Scan and C-Scan
(Courtesy of Abaris Training Resources Incorporated)

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A.2.1.3.4 Through Transmission.

Through transmission ultrasonic (TTU) is typically used in the factory for postfabrication inspection, and the coupling medium is with water jets for large components or immersion for smaller parts.

This method cannot only detect small (down to 0.5 in. (1.27 cm) diameter) defects, but the depth of the defect can also be determined. Large components can be inspected at reasonable speeds (e.g., 0.5 ft^2 (0.045 m^2) per minute), depending on a number of factors, including part complexity and required resolution. A C scan can be produced as a permanent record.

TTU can detect most delaminations in laminates as well as disbonds in adhesive joints and between the facesheets and core of sandwich structure core. Foreign inclusions that have significantly different acoustic impedance from that of the composite can also be found.

Ultrasonic testing requires calibration on known standards. This type of testing is basically comparing the trace of a good standard with the part being inspected, and interpreting the meaning of any differences found. This means that the TTU inspector must have a thorough knowledge of the structure being inspected. Based on the complexity of this method and the need for accurate interpretation, ultrasonic inspection requires well-trained, experienced personnel.

A.2.1.3.5 Pulse-Echo.

P/E uses a liquid gel or water as the couplant and is particularly suitable for fieldwork, i.e., damage detection and postrepair inspections.

Similar to TTU, P/E inspection can detect small defects through the thickness of a laminate as well as disbonds between facesheets and honeycomb core. Compared with factory TTU inspection, P/E is substantially slower and not as useful for large components.

It is more useful for inspecting areas that have yielded visual-damage indications. Most commercial aircraft operators will have P/E equipment available in their maintenance bases. To obtain accurate readings and correctly interpret the results, the inspectors must be carefully trained in the use of the equipment.

The advantages of ultrasonic inspection are as follows:

- This method can detect many types of anomalies: defects within the plane of laminates, delaminations, voids, foreign objects, moisture, disbonding, and some cracks.

- This technique can detect the depth of defects in thick laminates.

- For P/E, the equipment is very portable and flexible and only requires access to one side of a component.

- For TTU, large areas can be inspected at reasonable speeds.

- Also, for TTU, a three-dimensional (3D) image can be generated if necessary.

The disadvantages of ultrasonic inspection are as follows:

- A couplant is necessary between the transceiver and the component. This is usually in the form of a gel for P/E or water for TTU. However, new technology transducers, such as air-coupled transducers, are becoming available, and may eliminate these requirements.

- TTU typically requires removal of the component from the aircraft.

- P/E is a relatively slow inspection technique compared to TTU.

- TTU requires access to both surfaces of the part.

- Calibration standards are required for each material and thickness.

A.2.1.3.6 X-Ray.

Conventional x-ray of CFRP is difficult because the absorption characteristics of the fibers and resin are similar and the overall absorption is low.

The properties of glass and boron fibers are more suited to the use of x-ray as an inspection method for composites. Penetrants are often needed in conjunction with x-ray to identify defects such as very tiny cracks.

The advantages of x-ray are as follows:

- X-ray inspection may be used to detect transverse cracks, inclusions, honeycomb core damage, moisture ingression, voids, and porosity.

The disadvantages of x-ray are as follows:

- A considerable amount of safety measures are necessary with this technique.

- The equipment is not easily portable.

- Usefulness is limited by accessibility.

- The use of penetrants can result in contamination of the composite. Organic penetrants are affected by moisture, which may alter the recorded results, and halogen-based penetrants may result in stress corrosion.

A.2.1.3.7 Eddy-Current Inspection.

Eddy-current inspection is typically used to detect cracks emanating from fastener holes in metal structures without removing the fasteners.

Eddy current is of very limited use for detecting damage within composite structures and for inspecting repairs for integrity. It is limited to composites with a conducting phase, and the measurements obtained are sensitive to the volume fraction and integrity of that phase.

The advantages of eddy-current inspection are as follows:

- Eddy-current inspection can be used for detecting fractures in the substructure beneath a laminated skin.

- The equipment is easily portable.

- Eddy current can be used for checking volume fraction in CFRP composites.

The disadvantages of eddy-current inspection are as follows:

- Eddy-current inspection cannot be used to detect defects in GFRP composites due to a lack of a conducting phase in glass fibers.

- Eddy current is relatively insensitive to porosity, nonconducting inclusions, and delamination.

A.2.1.3.8 Thermography.

Two forms of thermographic inspection methods are currently available:

- Passive: The response of the structure being inspected to an applied heating transient is monitored.

- Active: Heating is produced by applying cyclic stress to the structure either in a fatigue test machine or in a resonant vibration system.

In both forms, the surface temperature of the structure is monitored, usually with an infrared camera, and anomalies in the temperature distribution can reveal the presence of defects. The passive method is more widely used than the active method, and its performance depends strongly on the heat source used. The conductivity and anisotropy of the composite are also important parameters. For example, in CFRP laminates, the conductivity in the laminate plane is approximately 9 times that in the through-thickness direction. This tends to obscure defects that are not close to the surface.

The advantages of thermography are as follows:

- Thermography is a single-sided scanning technology in which access to the back side of the component is unnecessary.

- Thermography is a full-field technology in which one image shows the entire area of interest on the component.

- Real-time technology in which the results are immediately visible to the inspector.

- Thermography is a quick method for inspecting large areas.

- Thermography is more convenient than x-ray because other personnel do not have to leave the area while the process occurs.

- Thermography can be used to find disbonds in adhesive joints, delaminations, and inclusions in which conductivity differs significantly from the base material.

- Thermography is often used by airline operators for detecting moisture in the form of ice in the honeycomb core of sandwich structures.

The disadvantages of thermography are as follows:

- Equipment costs are high.

- The method is not as sensitive as ultrasonic inspection for detecting delaminations and disbonds.

- The aircraft must be accessed soon after landing to detect moisture in the form of ice in the core of sandwich structures.

- Thermography is difficult to use with highly reflective materials such as metals.

- Skilled interpretation and knowledge of the aircraft structure and systems are required to determine which heat sources and sinks are real defects.

A.2.1.3.9 Bond Testers.

Bond testers are instruments that use the mechanical impedance method. They measure the change in local impedance produced by a defect when the structure is excited in the frequency range of 1 to 10 kHz. Defects, such as delaminations and adhesive disbands, can be detected.

These vibration techniques work at low frequencies, so coupling mediums are unnecessary.

Bond testers are readily portable; thus, they work well in the field. They are well suited for inspecting sandwich structures for facesheet separation from the core. Gross defects, such as widespread environmental degradation and facesheet disbonds in sandwich structure, produce readily measurable changes in resonant frequencies. Large areas can be inspected for gross defects in a very short time. This makes the technique attractive if small, localized defects are unimportant.

The advantages of bond testers are as follows:

- Bond testers are simple to use and can produce quick results.

- Bond testers are very portable and relatively inexpensive.

- No coupling fluids are required.

The disadvantages of bond testers are as follows:

- These instruments measure changes in resonant frequencies of whole components, and can only detect large degradation or disbonds.

- Bond testers cannot detect small, localized defects such as porosity or minor delamination.

- Bond testers cannot detect "kissing disbonds" in bonded joints, where the adhesive has failed, but the parts still contact each other.

A.2.1.3.10 Moisture Meters.

Moisture meters can be used to detect the presence of moisture when making repairs to GFRP or aramid materials. They can detect moisture within aramid honeycomb core. The moisture meters rely on radio frequency dielectric power loss. This power loss is attributed to an increase in the composite's conductivity is due to moisture absorption. Therefore, the techniques cannot be used with carbon or any other conductive material, such as metal, or with antistatic coatings that contain carbon.

The advantage of moisture meters is as follows:

- Moisture meters are very useful for checking for the presence of moisture when drying GFRP and aramid sandwich structures prior to bonded repairs. Moisture meters can readily detect moisture within GFRP and aramid laminates.

The disadvantages of moisture meters are as follows:

- Current models cannot be used on carbon structure. GFRP or aramid parts having an antistatic coating must be inspected for moisture from the back side.

- The presence of any metal inserts or doublers can give false indications and may cause panels to be removed needlessly.

A.2.1.3.11 Shearography.

Shearography is an emerging form of inspection that detects flaws based on a component's response to excitation (usually an imparted strain). With shearography, 3D images (out-of-plane surface displacement) of a component are compared under two different strain conditions to detect anomalies. In most cases, damaged areas will react to the strain differently than undamaged areas, thus indicating a flaw in the component. Both images are analyzed with a computer, where the two images are compared or sheared, and the flaws are highlighted in the resulting image.

The advantages of shearography are as follows:

- Shearography is a noncontact optical inspection method.

- With single-sided scanning, access to the back side of the component is not necessary.

- Full-field technology allows the entire component to be surveyed with a single image.

- The results are immediately visible to the inspector.

- Shearography is the best candidate for discovering kissing disbonds in bonded joints. (Note: Most inspection techniques cannot detect this type of flaw.)

- It provides a visual interpretation image of the entire component, and shows where the damage is located on the component.

The disadvantages of shearography are as follows:

- Shearography works best on structures less than 0.50 inch thick.

- It requires an expert to interpret the shearography results and may require additional inspection techniques to fully characterize the damage.

A.2.1.3.12 Related Documents.

SAE References (Commercial Aircraft Composites Repair Committee—CACRC)
AIR5279: Composite and Bonded Structure Inspector: Training Document
AMS2630: Inspection, Ultrasonic Product Over 0.5 inch (12.7 mm) Thick
AMS2632: Inspection, Ultrasonic, of Thin Materials 0.50 inch (12.7 mm) and Under in Cross-Sectional Thickness
ARP5089: Composite Repair NDT/NDI Handbook

A.2.1.4 Distinguish Between Skills Needed for Structures Engineers, Inspectors, and Technicians Dealing With Composite Maintenance and Repair.

Structures engineers, inspectors, and technicians all have vital roles in the proper maintenance and repair of composite structures. The skills needed by each discipline may differ, but each member of the repair team must be well versed in the requirements of the specific role, otherwise a wrongly assessed damage, a poor repair design, or an improperly processed repair may result.

A.2.1.4.1 Structures Engineer.

The structures engineer (at the OEM), operator, or MRO must have a Bachelor of Science degree, or equivalent, in engineering from an accredited academic institution, and some formal training in aircraft structural analysis.

To design an appropriate repair for a damaged component, the structures engineer must have detailed knowledge of the aircraft composite structural components and have access to, and understand, the loads and material design values used to certify the aircraft structure. Access to the design values for the repair materials is needed to be able to design a repair that restores the component's original strength and stiffness. For repairs to moveable control surfaces, the structures engineer must also consider the effects of a repair on the overall part stiffness, weight, balance, and flutter characteristics.

The structures engineer is responsible for providing repair designs and repair size limits and must know the capabilities and limitations of the repair technicians and their ability to process the repairs. Knowledge of the workability and limitations of the approved repair materials is essential when designing a repair. At the OEM, materials and process engineers usually provide details of material workability and limitations and repair-processing requirements. These details are provided in the OEM-authorized repair documentation, typically the SRM for specific repair materials and processing methods.

For most composite components, a typical SRM will contain ADL and repair designs for damages that exceed the ADL. The OEM structures engineering group is responsible for providing this information, and a structures DER will be needed to approve all data used for the ADL and repair designs. Regardless of where a specific repair is designed (at the OEM, at an airline operator's maintenance base, or at an MRO), all data used for ADL and repair designs must be approved by a structure DER.

Structures engineers responsible for providing ADL must have the ability to analyze the component for residual strength and stiffness with various forms and degrees of damages. To calculate accurate ADL, the engineer must have access to the certification loads for each specific component. In addition to the strength and stiffness databases used for the original component design, the structural analyst will need the database of residual strength and stiffness for that component with various types and degrees of damage. These damages will typically range from a dent or a crack (and any associated delaminations) to a through-penetration or hole. The ADL that the engineer will calculate typically require knowledge of damage parameters, such as dent depth and diameter and any associated cracks and known delaminations. The presence and location of other damages or repairs will affect the ADL for a specific component.

Structures engineers must have detailed knowledge of in-service inspection methods and their limitations to provide ADLs that qualified maintenance personnel can reliably detect.

A.2.1.4.2 Inspector.

The inspector is responsible for assessing and mapping any damage that may be discovered by maintenance personnel during operations or during a scheduled maintenance event, such as an aircraft maintenance C or D check. The inspector is also responsible for inspecting and approving any repair that may have been performed as a result of the damage assessment.

For planned maintenance events such as C or D checks, the MPD must be reviewed for directed inspections. In the maintenance depot, a variety of inspection techniques will be available to

composite component inspectors. Inspectors must have good eyesight and hearing, and be trained and qualified in these inspection techniques. An MPD may require specific inspection techniques to adequately inspect critical component zones (e.g., areas around the fittings). Inspection equipment commonly available in the maintenance depot is the tap hammer or coin; ultrasonic equipment, such as P/E, bond testers, eddy current, and x-ray equipment; and moisture meters. The more sophisticated MROs and operator maintenance depots may have thermographic equipment on hand, and in the future, may invest in interferometry to more accurately detect hidden flaws, such as loose fasteners and weak bondlines.

All the inspection techniques require extensive training and retraining so the inspector is consistently competent in the techniques, including application to specific structural details.

For damages discovered on the aircraft while on the ramp during routine walk-around inspections, the damage must be mapped as accurately as possible with the available inspection equipment. In these cases, to make an adequate damage disposition, the damage must be mapped by a person qualified in basic inspection techniques such as the tap hammer and P/E equipment. In many cases, a qualified inspector or technician may not be on hand; therefore, qualified personnel must be flown to the aircraft to make a proper disposition. In some cases, repair dispositions performed while on the ramp may be temporary (i.e., a temporary repair that covers the assessed damage and allows the operator to fly the aircraft, without passengers, to a maintenance base for a permanent repair).

Many materials, such as prepregs, may be qualified for use beyond the shelf life expiration dates. A number of simple tests must be performed, and if the test results are acceptable, the material is recertified for an additional time period. The inspector, in many facilities, is responsible for material retesting and recertification.

In most situations, the repair technician is responsible for monitoring his own in-process controls during a bonded repair, and the inspector examines and either approves or rejects the in-process control records. In some cases, however, an inspector monitors the in-process controls during the bonded repair. This is also usually the case at the OEM during any composite bonding process.

A.2.1.4.3 Repair Technician.

It is essential that the repair technician be familiar with the specific aircraft structure drawing system and the approved maintenance methods for the particular component in question. A technician must be able to follow the repair procedure explicitly and understand the ramifications of taking short cuts or guessing. Repair technicians need to be trained and qualified in all types of repair for which they are responsible (i.e., bonded and bolted repairs). In some large MROs and operator maintenance depots, there may be specialists for bonded composite repairs, bolted repairs to composite components, and metal bond repairs. In most small maintenance organizations, technicians will be responsible for processing all kinds of repairs.

The technician's duties are varied. In addition to performing the actual repairs, their responsibilities may include the following:

- Initial inspection of composite components during C and D checks. While not specifically qualified in the use of specific inspection techniques (see section A.2.1.4.2), a technician will need to be sufficiently trained to be able to perform initial damage assessments using visual and basic tap inspection techniques.

- Monitor repair materials, such as resins, adhesives, prepregs, potting compounds, and sealants, which are all perishable. Their shelf lives must be carefully monitored and correct storage conditions must be maintained.

- For components with temporary, interim, or time-limited repairs, the technician must review the component records for flight cycles in order to know if the repairs need to be replaced with permanent repairs. Many authorized documents, such as the SRM, have restrictions for proximity of repairs and damages. Therefore, when damages are found, the technician must review the component records for previous repairs or previously allowed damages.

The technician is also responsible for maintaining the following worksheets:

- Component master worksheet
- Materials record sheet
- Component record card

For good aircraft maintenance, these records must be kept up to date.

A.2.1.4.4 Related Documents.

SAE References (Commercial Aircraft Composites Repair Committee—CACRC)
AIR5278: Composite and Bonded Structure Engineers: Training Document
AIR5279: Composite and Bonded Structure Inspector: Training Document

A.2.1.5 Recognize Skill Limits in Practice and Where to Find Assistance.

While it is important that each member of the repair team understands the roles of the others, it is not recommended that any one team member be responsible for carrying out all roles. This may not be feasible in some small maintenance organizations, especially those repairing light or GA aircraft. In some organizations, the same person may be responsible for both the actual repair work and inspection. In a few cases, engineering, repairing, and inspecting may all be performed by the same person.

The following scenarios are presented to map the process for a damage/repair disposition:

- If damage is detected during a routine maintenance event, qualified personnel should be on hand to perform their part of the damage/repair dispositions. After the damage has been mapped by a qualified inspector, the repair technician will consult the approved

documentation for the specific ADL. If the damage is within the ADL, the repair technician will prepare the part for sealing and restoration to service. If the damage is larger than the ADL, and the documentation contains an approved repair, the repair technician will perform the repair using approved repair materials, adhering strictly to the approved repair process. The repair process steps will be checked and the completed repair will be inspected by a qualified inspector, and if the repair is found to be satisfactory, the component will be refinished, as necessary, for restoration to service.

- If the damage is discovered during operations, the ramp technician may need to consult with the appropriate maintenance depot engineer for help with the disposition. If the ramp technician is not qualified to make an inspection, a qualified inspector will have to be dispatched to the site. After the damage has been mapped, the engineer will compare the mapped damage to the ADL for the specific component in the approved documentation. If the damage is within limits, the damage/repair disposition will be relayed to the ramp technician, and the ramp technician will seal the part and restore it to service. If the damage is larger than the ADL, an approved repair is contained in the SRM, and the ramp technician is qualified to perform the repair, the ramp technician will perform the repair per the approved process with approved materials. The repair will be inspected, and if found satisfactory, the component will be refinished for restoration to service. If the ramp technician is not qualified to perform the repair or a qualified inspector is not available, a repair technician or inspector will need to be dispatched to the site.

- If a repair process is not available in the approved documentation, the engineer will need to either communicate with the OEM for an approved repair or request a repair from a DER that will restore the strength and stiffness of the component in question.

In any damage scenario, it is essential that only qualified personnel perform each task. If this protocol is not followed, any or all of the following may result:

- A repair design that is not approved may not properly restore component strength and stiffness.

- An incorrectly mapped damage may result in a component either being restored to operations with a critical damage or with an inadequate repair.

- A repair incorrectly processed may result in either the repair being found unsatisfactory (removed or a new repair made) or the part may be returned to service with an inadequate repair.

- An incorrectly inspected repair may result in a component being returned for service with an inadequate repair.

Safety Message

All aspects of composite maintenance and repair are interlinked: The damage to a composite aircraft component must be first discovered (if discovered by operations personnel, it must be

reported to maintenance). The damage must be assessed by qualified maintenance personnel. Damage disposition must include an interface with engineering personnel familiar with approved data for the structure in question. If repair is necessary, the repair design must be approved by the appropriate authority (previously approved data such as an SRM or other approved repair method).

The chosen repair method must also be approved (previously approved data such as an SRM or provided by a DER). The approved repair process must be performed by a qualified technician, using qualified materials, and strictly adhering to the appropriate in-process controls. The completed repair must be inspected for approval by a qualified inspector.

If any of the above maintenance actions contain errors or flaws, a deficient repair may result. It is essential for safety that qualified inspectors, technicians, and engineers are involved in composite maintenance and repair.

A.2.2 IDENTIFY AND DESCRIBE INFORMATION CONTAINED IN DOCUMENTATION AND REGULATORY RULES.

A.2.2.1 Describe Requirements in Material and Process Specifications and Approved Repair Information.

A.2.2.1.1 Material and Process Specifications.

To ensure that required standards are met, all materials and processes used in aircraft composite structural fabrication and repair must meet accepted specifications. For repair materials and processes, this means that sufficient development and testing was done to establish repeatable material characteristics, such as workability, shelf life, out-time, bonding compatibility, and guaranteed strength and stiffness performance. The material characteristics are then converted by OEMs or standards-developing organizations into material specifications that must be met by material suppliers, and process specifications or instructions that must be followed during fabrication or repair. Some standards-developing organizations include the Aerospace Standards branch of SAE, ASTM, ISO, and others.

Each material specification specifies strength and stiffness requirements, material handling and workability characteristics, out-time requirements, and a set of acceptance tests and expected values. The acceptance tests are part of the specification to ensure that the materials received from suppliers meet the specification values and, hence, can be correctly processed in a component or repair to yield the specified strength and stiffness performance.

OEM-approved repair documentation (SRMs) and component drawings often call out materials and processes by the OEM company specifications or other specifications accepted by the OEM. Sometimes, the actual material supplier designations (e.g., EA 9390) may be used. Repair work must use materials and processes to these specifications or approved equivalents. As an example, in Boeing SRMs, repair materials are typically defined by the Boeing specifications (e.g., Boeing Material Specification 8-256). For structural repairs to meet the specified strength

and stiffness performance, the end-user must store, handle, and process the repair materials per the process instructions.

A.2.2.1.2 Approved Repair Information.

There are several types of approved repair information. Most OEMs of 14 CFR Part 25 commercial aircraft provide approved repair information in the form of SRMs. Other smaller aircraft manufacturers may provide repair information in the form of data sheets and references to sources of information. Repair technicians may be directed to MIL-HDBK-17, also known as CMH-17, or other sources for material strength values, repair design recommendations, inspection, and processing information. Repairs designed using this information will require approval from the appropriate regulatory agency (e.g., by an FAA-approved DER). Some airlines may employ their own internal repair documentation (e.g., repair schemes for repetitive damages that their own DERs have approved).

A.2.2.1.3 Structural Repair Manuals.

Structural repair manuals are often sources of approved OEM repair information. SRMs that apply to 14 CFR Part 25 commercial aircraft are typically arranged per the indexing basis of the Air Transportation Association of America (ATA) 100 system. This system allocates a number (e.g., Chapter 57 for wings) to all the major parts of the aircraft. SRMs only use Chapters 51 through 57, which cover structures. Chapter 51 deals with general aircraft and repair topics, while specific repair information is provided in Chapters 52 through 57.

Examples of Boeing SRM chapters are:

- Chapter 51: General information about the aircraft structure and repair information

- Chapter 52: Doors—passenger and crew entry doors; cargo doors; service doors; fixed interior doors (flight compartment door); landing gear doors (nose, main, and main gear shock strut doors)

- Chapter 53: Fuselage

- Chapter 54: Engine nacelles and pylons

- Chapter 55: Vertical and horizontal stabilizers, elevators, and rudder

- Chapter 56: Windows—passenger, flight deck, and door windows

- Chapter 57: Wings—Wing center box; outer wing main torque box; wing tip/winglet; leading edge and leading-edge slats; trailing edge and trailing-edge devices—ailerons, flaps, flaperons, and spoilers

Each section within Chapters 52-57 provides detailed information about (1) the structural details of each component, (2) ADL, and (3) repair options, such as temporary, interim, and permanent repairs, and repair process instructions. The typical Boeing SRM presents diagrams of each

component, associated drawing numbers, overall dimensions, and lists of materials. Typically shown are sections cut-through components or component panels, showing ply lay-ups, orientations, and core details, if components are of sandwich construction.

Allowable damage covers minor damages that can be dispositioned without adding repair material, but with environmental exposure protection (e.g., SpeedTape). The damaged component can be returned to service until it can be permanently repaired. Damages may be classified into several types, such as dents, cracks, holes and punctures, delamination, nicks, scratches, and gouges, with allowable damage sizes presented for each type.

There are size and depth limits associated with each type of allowable damage, and limits of proximity to details such as edges, fasteners, other damages, and previous repairs. ADLs are based on the type and extent of the damages, and the component loads (e.g., stress or strain fields). For flight control panels, which have several actuator and hinge attachments, there will be significant variation in the stress fields and associated zones that have different ADLS. In the event of damage in some highly critical areas (e.g., adjacent to a hinge or actuator fitting), the SRM may direct the operator or MRO to contact the OEM for a disposition.

There are often three repair options referenced in approved data:

- Temporary repairs are allowed in order to permit the aircraft to meet flight schedules. These repairs are typically time-limited, and must be removed before or at the time limit, and replaced with a permanent repair. The limits for temporary repairs are usually quite short, due to the lack of confidence in the durability of the materials used for these quick repairs. The cure temperatures for these repairs range from room temperature to 150°F, thus, they do not exhibit high strength and stiffness at high operating temperatures.

- Interim repairs allow the aircraft to continue in service indefinitely with scheduled inspections of the interim repair. These types of repairs are usually cured at 150°F or higher and consequently exhibit better durability and high-temperature performance. The interim repairs are usually larger than temporary repairs.

- Permanent repairs are considered terminating actions and, in many cases, have no requirement for subsequent inspections outside those scheduled for the base structure.

Bonded permanent repairs are usually cured at 200°F or higher, and the associated size limits are larger, due to confidence in durability and superior performance, particularly at higher operating temperatures.

In some cases, such as for more highly loaded laminate-stiffened components, a bolted permanent repair will be offered. Bolted titanium patch repairs are offered for more highly loaded composite components, such as the B-777 and Airbus A320/330/340 horizontal and vertical stabilizer main torque boxes. These components have a laminate-stiffened configuration and employ many laminate plies in the skins, stiffeners, and spars. Similar to permanent repairs offered for metal components, these repairs are often limited in size and location. In the case of critical-to-flight components, the SRM will present zones that range from less critical to most critical, and repair size limits will shrink accordingly. As for ADLs, in the event of damage in

some highly critical areas, the SRM may direct the airline operator or MRO to contact the OEM for a repair disposition.

A.2.2.1.4 Repair Process Instructions.

Step-by-step instructions for temporary, interim, and permanent bonded repair options are presented in SRMs. Usually, the temporary and interim repair options are wet lay-up methods. For the permanent repair option, both prepreg and wet lay-up repair methods are offered, as well as bolted repairs for the thicker parts mentioned above. Exact step-by-step process instructions are presented for each type of repair. These steps include damage assessment and mapping, damage cleanup, surface preparation, material preparation, the actual repair process steps, including in-process QC instructions, postrepair inspection, and surface restoration.

Typically, ADL and repair instructions provided in an SRM are approved by the responsible regulatory authority and, therefore, are approved data. As an example, all Boeing SRM information is approved by DERs who work at the company.

The data in some OEM SRMs (e.g., some GA OEMs) do not have adequate research or testing backup for it to be considered approved data. For the unapproved SRM repair schemes, each specific repair disposition must be approved by a DER.

Many airlines have their own internal repair documents. These may include repair schemes for single, large damages or repetitive damages of a minor nature. Approval for the single, large repairs will usually require the OEM's approval, while the repetitive minor repairs may be approved by either the OEM or a local DER. In time, repair dispositions for repetitive minor damages are often added to the OEM's SRM. Repairs for larger damages that are outside the scope of the SRM are often not added to the SRM because they may involve several elements (e.g., a ground collision that damages a portion of a wing skin, a rib, and a spar) and their interfaces (e.g., fastening). Approved repairs for these types of large, complex damages are typically unique, and are not considered sufficiently repetitive for adding complex instructions and diagrams to the SRM.

In-house modifications to specific aircraft components may be made by airline operators to ease maintenance or strengthen a part to increase durability or service life. In many cases, these modifications may be adopted later by the OEM as a SB or, if deemed necessary, adopted by the regulatory agency as an Airworthiness Directive (AD).

GA OEMs may provide SRMs, and some may provide approved repairs for critical components or components that are consistently damaged. As previously mentioned, some GA OEMs may refer repair technicians to available repair information such as the MIL HDBK-17. In every case, the repair information must be approved by a representative of the appropriate regulatory agency (e.g., an FAA-approved DER).

A.2.2.2 Demonstrate the use of Source Documents.

There are a number of source documents issued by aircraft OEMs containing maintenance, modification, rework, and repair information. Transport aircraft, commuter and GA, and rotorcraft are operated to the requirements of their specific regulatory agencies. The regulatory requirements dealing with the various categories of aircraft and rotorcraft may differ based on the specific aircraft type. The OEMs of these aircraft all issue some form of source documentation that provides maintenance and repair information to help the operators conform to the specific requirements. Some of the more common types of source documents are discussed below.

SBs are issued by OEMs and are the means by which modifications, inspections, or rework instructions are automatically passed on to owners/operators and repair technicians (MROs) of OEM aircraft. SBs have several levels of priority. These range from mandatory and alert, which indicates that a high level of urgency exists and flight safety is a concern, to a lower level of SBs, generally called routine. Routine SBs can deal with life or reliability extension recommendations for parts that have been troublesome in service, but for which safety has not been the issue.

SBs may be issued as the result of an Airworthiness Directive (AD) issued by the FAA as a requirement to modify a specific component due to previously detected damage or partial failures. At other times, an AD is issued to direct operators to perform a specific OEM SB within a certain time frame.

Service Newsletters are issued by OEMs to address troublesome components or systems. The object of Service Newsletters is to make operators aware of problems or potential problems, and solutions, if any are available. Advice or methods of troubleshooting are often given in Service Newsletters.

MPD or AMM documents are issued by OEMs to operators and MROs to provide specific inspection requirements and overhaul instructions for a specific aircraft. MPDs and AMMs lay out preventive maintenance programs, which set fixed intervals for the overhaul and inspection of specific parts of the aircraft. They cover all parts of the aircraft, including structure, avionics, electrical and hydraulic systems, flight controls, powerplants, and landing gear. These documents are consulted when an aircraft comes into the depot for scheduled maintenance, such as a C or D check.

For structures, the MPD and AMM provide the inspection intervals for specific components. As an example, for composite parts, the MPD may require that the critical laminate areas adjacent to hinge fittings and actuators on rudders, elevators, and ailerons be inspected at regular intervals, such as during D checks, using a specific NDI method to ensure that no delamination has occurred. This type of MPD instruction would typically be the result of the area designated for inspection being considered critical to flight safety.

Component Maintenance Manuals (CMM) typically cover specific large components or parts, such as radomes and engine parts. For 14 CFR Part 25 aircraft, the CMM is arranged in ATA

order and contains part lists, plus a list of materials and their suppliers. As an example, the A300 radome is covered in CMM 53-51-11.

The SRM, issued and maintained by an OEM, is the source document that is consulted when damage to a structural component is discovered. Complete typical damage/repair instructions are listed in the SRM.

A.2.2.3 Be Exposed to and Identify and Demonstrate the use of Regulatory Documents.

Regulatory authorities, with jurisdiction for a particular repair, issue documents that specify requirements and compliance guidance for the maintenance of civil aircraft.

The FAA provides federal aviation regulations (14 CFR).

In Europe, the legal basis for design, production, and maintenance of aeronautical products, parts, and appliances, including related personnel and organizations, is covered by Regulation 1592/2002 issued by the European parliament and council. In addition, Implementing Rule 1703/2003 lays down airworthiness and environmental certification standards, and Implementing Rule 2042/2003 addresses continuing airworthiness.

The Transport Canada Civil Aviation (TCCA) provides Canadian Aviation Regulations (CAR), which cover all aspects of aviation, including production, certification, and maintenance of aeronautical products.

U.S.-registered aircraft operating to or within foreign jurisdictions must meet the FAA requirements as well as the requirements of the country in which the aircraft will be operating. Many countries have bilateral agreements in place to minimize duplication in the certification process.

Bilateral agreements facilitate the reciprocal and initial airworthiness certification of civil aeronautical products imported or exported between two signatory countries.

Bilateral Airworthiness Agreement (BAA) or Bilateral Aviation Safety Agreement (BASA) with Implementation Procedures for Airworthiness (IPA) provides for airworthiness technical cooperation between the FAA and its counterpart civil aviation authorities. BAA and IPA contains only type certification provisions, whereas BASA contains both type and operation provisions. BAA and BASA are treaties that the U.S. establishes with other countries and are not considered to be aviation regulatory requirements.

Specific information on individual country agreements can be obtained on the following FAA website:
http://www.faa.gov/aircraft/air_cert/international/bilateral_agreements/baa_basa_listing/

A.2.2.3.1 Identify Regulatory Documents.

- FAA—14 CFR:

    - Part 43 prescribes general maintenance requirements applicable to various categories of aircraft.

    - Part 65 [Subparts D and E] prescribes requirements for the certification of mechanics and repairmen, respectively.

    - Part 145 prescribes requirements for the certification of repair stations.

    - Part 183 describes the requirements for designating private persons to act as representatives of the FAA Administrator in examining, inspecting, and testing persons and aircraft for the purpose of issuing airman, operating, and aircraft certificates.

    - Parts 91 [Subpart E], 121 [Subpart L], 125 [Subpart G], and 135 [Subpart J] provide additional maintenance requirements for specific types of operations.

- EASA

    - The Annex to European regulation 1702/2003 (Part 21) prescribes procedural requirements for environmental protection and certification, including design and production organization approval, repairs, and continued airworthiness.

    - In Europe, implementing rule 2042/2003 Annex III (Part 66) prescribes requirements for certifying staff.

    - In Europe, implementing rule 2042/2003 Annex II (Part 145) and Annex IV (Part 147) prescribes requirements for maintenance and training organization approvals, respectively.

    - The European/EASA system differs from the FAA. The requirements for responsible individuals are embedded, both directly and less directly, throughout the appropriate requirements. For example, Part 66 for licensed certifying maintenance staff, and Part 21 Subpart J for DO CVE (Design Organization Compliance Verification Engineer).

- TCCA

    - TCCA CAR 571 prescribes requirements in respect to the maintenance and elementary work performed on aircraft.

    - TCCA CAR 403 prescribes requirements for holders of and applicants for Aircraft Maintenance Engineer licenses and ratings as well as requirements for approved training organizations.

- TCCA CAR 573 prescribes the maintenance of aeronautical products or the provision of maintenance services.

- TCCA CAR 505 contains the procedures and conditions where, pursuant to the Aeronautics Act, the Minister may authorize persons to act on his behalf with respect to the airworthiness of aeronautical products.

- TCCA addresses the continued airworthiness process through CAR Part VI, "General Operating and Flight Rules," and CAR Part VII, "Commercial Air Services."

ADs are issued to address the mandated requirements that relate to the safe operation of aircraft (FAA per 14 CFR Part 39, EASA per EC1702/2003 Part 21A.3B, and TCCA per CAR 593).

A.2.2.3.2 Guidance Documents.

The FAA issues guidance that provides supportive information showing compliance with regulatory requirements. Guidance may include ACs and Policy Statements. In general, an AC presents information concerning acceptable means, but not the only means, of complying with 14 CFR.

- AC 145-6, "Repair Stations for Composite and Bonded Aircraft Structure"

- AC 65-31A, "Training, Qualification, and Certification of Nondestructive Inspection (NDI) Personnel"

Many foreign authorities may have their own guidance that supports the compliance of regulatory requirements. EASA publishes guidance, i.e., AMC (Acceptable Means of Compliance), or uses corresponding ACs. TCCA publishes guidance, such as ACs, Policy Letters and Staff Instructions.

All regulatory authorities issue their own documents that specify requirements or give advice for the operation of civil aircraft.

The following are the regulatory documents issued by the FAA and EASA:

- CFR
- EASA CS

Other governments have their own documents, but they are usually the same as or based on the CFRs or CSs. Repairs performed while a U.S.-registered aircraft is in service within the U.S. are controlled by the CFRs that are relevant to the maintenance of civil aircraft. These maintenance-related regulations may include:

- 14 CFR Part 43 (maintenance, preventive maintenance, rebuilding, and alteration) prescribes general requirements applicable to various categories of aircraft and operations. 14 CFR Part 43 and the associated ACs specify methods that have been approved for repair and alteration. If a specific repair, which is designed and performed

by an operator or repair technicians, is not already approved for use (i.e., described in the OEM's SRM), it must be transmitted in detail to the OEM for approval or be given special approval by a DER.

- 14 CFR Part 65, Subpart D (mechanics) and Subpart E (repairmen) prescribes requirements for the certification of mechanics and repairmen. Subpart D also includes the requirements for the certification of inspectors.

- 14 CFR Part 145 (repair stations) prescribes requirements for the certification of repair stations. To obtain FAA certification, a repair station must submit documentation to demonstrate the skills of personnel, inspection procedures, and the necessary facilities and equipment.

14 CFR 145.5 details certificate and operations specifications requirements. These are:

- No person may operate as a certificated repair station without, or in violation of, a repair station certificate, ratings, or operations specifications issued under this part.

- The certificate and operations specifications issued to a certificated repair station must be available on the premises for inspection by the public and the FAA.

The requirements that are associated with various specific operations may include:

- General operating and flight rules—14 CFR Part 91, Subpart E

- Operating requirements: Domestic, flag, and supplemental operations—14 CFR Part 121, Subpart L

- Certification and operations: Airplanes having a seating capacity of 20 or more passengers or a maximum payload capacity of 6000 pounds or more—14 CFR Part 125, Subpart G

- Operating requirements: Commuter and on-demand operations and rules governing persons onboard such aircraft—14 CFR Part 135, Subpart J

European-registered civil aircraft operated in Europe are controlled in a slightly different way than in the U.S. For example, Europe does not use the same operating rule structure as the U.S. However, similar intent is captured with respect to maintenance and repair in Europe. The continued airworthiness process is controlled by European Commission (EC) Regulation No. 2042/2003 (see EASA website www.easa.eu.int). If U.S.-registered aircraft operate to or within Europe, they must meet the standards of both the U.S. and European CFRs and CSs. Similarly, European-registered aircraft operating to or within the U.S. must meet the requirements of both. The requirements of the CFRs and the CSs are mainly similar, but there are some differences.

Note that the interaction of regulatory agencies is often governed by a BASA.

EASA documents of interest include:

- Part 21 Subpart G—Production Organization Approval
- Part 21 Subpart J—Design Organization Approval
- Part 21 Subpart M—Repairs
- Part M—Continuing Airworthiness
- Part 66—Certifying Staff
- Part 145—Maintenance Organization Approval
- Part 147—Training Organization Requirements

MROs and repair technicians that maintain aircraft registered by both the FAA and foreign governments must be cognizant with the requirements of the specific regulatory agencies and any differences between them.

Other important regulatory documents are ACs and ADs.

ACs are issued by the FAA and EASA to give guidance on an acceptable means of compliance to the regulations. ACs provide information and guidance concerning acceptable means, but not the only means, of demonstrating compliance with the requirements of the CFRs.

Some important ACs for composite structure are:

- AC 20-107A, "Composite Aircraft Structure." This AC sets forth an acceptable, but not the only, means of showing compliance with the provisions of 14 CFR Parts 23, 25, 27, and 29, regarding airworthiness type certification requirements for composite aircraft structures, involving fiber-reinforced materials, e.g., carbon (graphite), boron, aramid (Kevlar), and glass-reinforced plastics. Guidance information is also presented for associated QC and repair aspects.

- AC 23-13, "Fatigue and Fail-Safe Evaluation of Flight Structures and Pressurized Cabin for 14 CFR Part 23 Airplanes." This AC was developed from experience with the certification of metallic aircraft structures, but much of it is applicable to composite aircraft structure. Unlike AC 20-107A, which is applicable to all composite aircraft, this AC is applicable to 14 CFR Part 23 aircraft only.

- AC 145-6, "Repair Stations for Composite and Bonded Aircraft Structure." This AC provides information on the repair and fabrication of composite materials and adhesive-bonded components, and on the inspection systems, equipment, and facilities that a certificated repair station with the appropriate ratings should have available to perform repairs or alterations on such materials and components. These guidelines supplement the procedures in the OEM SRMs. The AC provides good, broad information and guidance for operating a composites shop.

- AC 21-26, "Quality Control for the Manufacture of Composite Structure." This AC provides information and guidance concerning an acceptable means of demonstrating compliance with the requirements of 14 CFR Part 21, Certification Procedures for

Products and Parts, regarding QC systems for the manufacture of composite structures that involves fiber-reinforced materials.

- AC 23-20, "Acceptance Guidance on Material Procurement and Process Specifications for Polymer Matrix Composite Systems." This AC provides guidance for material and process specifications, or other documents, used to ensure sufficient control of composite prepreg materials. It was released by the FAA Small Airplane Directorate but contains information useful for all categories of aircraft.

AD, also issued by the FAA and EASA, are instructions that must be complied within the time scale set down in the specific AD. They are issued when urgent action is required to address safety issues concerning structural alterations and operating or maintenance procedures. In extreme cases, they could require grounding an aircraft type pending investigation. For example, FAA AD 2002-07-08, issued in May 2002, directs the inspection of lap splices on B737-200/300/400/500 aircraft at certain locations on the fuselage. This particular AD requires that the inspection prescribed in Boeing SB 737-53A1177 be performed before the accumulation of 50,000 flight cycles of each aircraft. If cracking is discovered by the requisite NDI method, the AD directs the repair prescribed in SB 737-53A1177 be performed. This AD was issued as the result of cracks found on a number of aging B-737 aircraft and an in-flight incident.

It is essential that composite repair stations have access to 14 CFRs and Joint Aviation Requirements, ADs, and ACs, particularly the documents specific to composite aircraft structure. It is essential that copies of these documents are available to maintenance engineers, and they are fully cognizant of their contents. Inspectors and technicians should also be aware of these documents and their contents.

A.2.2.4 Describe the Requirements and Engineering Approvals Necessary for Valid Sources of Technical Information and Maintenance Instructions.

Repair of composite structural parts must be in compliance with the airworthiness regulations associated with various categories of aviation products.

- 14 CFR Part 23, EASA CS-23, and TCCA Airworthiness Manual (AWM) Chapter 523 provide airworthiness requirements applicable to normal, utility, acrobatic, and commuter category small airplanes. In addition, EASA has CS-VLA, Very Light Aeroplanes, and CS-22, Sailplanes and Powered Sailplanes; and TCCA has AWM Chapter 522, Gliders and Powered Gliders, and AWM Chapter 523-VLA, Very Light Aeroplanes.

- 14 CFR Part 25, EASA CS-25, and TCCA AWM Chapter 525 provide airworthiness requirements applicable to transport category and large airplanes.

- 14 CFR Part 27, EASA CS-27, and TCCA AWM Chapter 527 provide airworthiness requirements applicable to normal category and small rotorcraft.

- 14 CFR Part 29, EASA CS-29, and TCCA AWM Chapter 529 provide airworthiness requirements applicable to transport category and large rotorcraft.

- 14 CFR Part 33, EASA CS-E, and TCCA AWM Chapter 533 provide airworthiness requirements applicable to aircraft engines

- 14 CFR Part 35, EASA CS-P, and TCCA AWM Chapter 535 provide airworthiness requirements applicable to propellers.

All steps in composite repair, such as design, materials, and processes must be substantiated to meet airworthiness regulations. Composite inspection and repair design and materials and processes that have not been previously approved must follow the regulatory processes for structural substantiation.

Important regulations for the structural substantiation of a composite repair may include:

- Strength and deformation
- Proof of structure
- Inspections and other procedures
- Materials and workmanship
- Fabrication methods
- Protection of structure
- Accessibility provisions
- Material strength properties and design values
- Damage tolerance and fatigue evaluation of structure
- Flutter (aeroelastic stability)
- Fire protection of flight controls, engine mounts, and other flight structure
- Electrical bonding and protection against lightning and static electricity
- Instructions for continued airworthiness

The FAA issues guidance to show compliance of airworthiness requirements for composite structures. Guidance includes:

- AC 20-107A, "Composite Aircraft Structure" (EASA has AMC No. 1 to CS 25.603, and TCCA has AC 500-009)

- AC 21-26, "Quality Control for the Manufacture of Composite Structures"

- AC 23-20, "Acceptance Guidance on Material Procurement and Process Specifications for Polymer Matrix Composite Systems"

- AC 35.37-1, "Guidance Material for Fatigue Limit Tests and Composite Blade Fatigue Substantiation"

The FAA will also issue PS, supporting the certification of composite structures. PSs are listed on the FAA website.

For the safe use of composite materials in the manufacture and repair of aircraft structural parts, the FAA and EASA require compliance with the airworthiness standards associated with various categories of aviation products. For the FAA:

- 14 CFR Part 23 applies to normal, utility, acrobatic, and commuter category airplanes
- 14 CFR Part 25 applies to transport category airplanes
- 14 CFR Part 27 applies to normal category rotorcraft
- 14 CFR Part 29 applies to transport category rotorcraft
- 14 CFR Part 31 applies to manned, free balloons
- 14 CFR Part 33 applies to aircraft engines
- 14 CFR Part 35 applies to propellers

EASA has similar CSs that cover the above aircraft and rotorcraft categories (e.g., CS-23, CS-25, CS-27, and CS-29, respectively). Furthermore, EASA has a CS specific to small aircraft <750 kg, i.e., CS-VLA, and a CS specific to sailplanes, i.e., CS-22 (see EASA website www.easa.eu.int).

Each category of aircraft may have different requirements, but not all of them will be covered here. Some of the important sections of 14 CFR Part 25 that cover materials and processes for transport aircraft are as follows:

- Section 25.571: Damage tolerance and fatigue evaluation of structure. An evaluation of the strength, detail design, and fabrication must show that catastrophic failure due to fatigue, corrosion, or accidental damage will be avoided throughout the operational life of the airplane. This evaluation must be conducted in accordance with the provisions of paragraphs (b) and (e) of this section, except as specified in paragraph (c) of this section, for each part of the structure, which could contribute to a catastrophic failure (such as wing, empennage, control surfaces and their systems, the fuselage, engine mounting, landing gear, and their related primary attachments).

- AC 25.571-1 contains guidance information relating to the requirements of this section.

The damage tolerance evaluation must include a determination of the probable locations and modes of damage due to fatigue, corrosion, or accidental damage. For metal aircraft structure, the determination is made by analysis that is supported by test evidence and (if available) service experience. Generally, composite aircraft structure has a greater dependence on tests. The evaluation must incorporate repeated load and static analyses supported by test evidence. The extent of damage for residual strength evaluation at any time within the operational life must be consistent with the initial detectability and subsequent growth under repeated loads. The residual strength evaluation must show that the remaining structure is able to withstand loads (considered as static ultimate loads).

Damage tolerance analyses must be performed on any repairs made to principal structural elements. Section 25.603, Materials, requires that the suitability and durability of materials used for parts, the failure of which could adversely affect safety, must

- be established on the basis of experience or tests.

- conform to approved specifications that ensure their having the strength and other properties assumed in the design data.

- take into account the effects of environmental conditions, such as temperature and humidity, expected in service.

14 CFR 25.605, fabrication methods, requires the following:

- The method of fabrication must produce a consistently sound structure. If a fabrication process (such as gluing, spot welding, or heat treating) requires close control to reach this objective, the process must be performed under an approved process specification.

- Each new aircraft fabrication method must be substantiated by a test program.

14 CFR 25.613, material strength properties and design values, requires the following:

- Material strength properties must be based on enough tests of material meeting approved specifications to establish design values on a statistical basis.

- Design values must be chosen to minimize the probability of structural failures due to material variability.

EASA's CS-25 for transport aircraft contains sections that are similar to these requirements. Similar sections are contained within the other CFRs and CSs, depending on the specific requirements of each.

Section 5 of AC 20-107A sets out material and fabrication development recommendations. It states that

> "To provide an adequate design database, environmental effects on the design properties of the material system should be established.
>
> a. Environmental design criteria should be developed that identify the most critical environmental exposures, including humidity and temperature, to which the material in the application under evaluation may be exposed. …Experimental evidence should be provided to demonstrate that the material design values or allowables are attained with a high degree of confidence in the appropriate critical environmental exposures to be expected in service. …

b. The material system design values or allowables should be established on the laminate level by either test of the laminate or by test of the lamina in conjunction with a test validated analytical method.

c. For a specific structural configuration of an individual component (point design), design values may be established which include the effects of appropriate design features (holes, joints, etc.).

d. Impact damage is generally accommodated by limiting the design strain level."

Section 9.h of AC 20-107A states:

"When repair procedures are provided in FAA approved documents or the maintenance manual, it should be demonstrated by analysis and/or test that methods and techniques of repair will restore the structure to an airworthy condition."

While the requirements and guidance given above were established to cover the original structure materials, design values, and fabrication methods, they apply equally for repair of these structural parts. Materials and processes used in repair of aircraft composite structural parts must be the result of qualification programs that provided the detailed information and engineering practices to help ensure the control of repeatable material properties and processes. The qualification programs must generate statistically based material properties for critical loading modes and environmental conditions.

Repair instructions defined in an SRM are based on material and processing qualification testing that ensures repeatable material properties and processes. These instructions, which are approved by the regulatory agency or a delegated authority within the OEM (i.e., a DER), must be followed. Structural repair materials are subject to material specifications that detail storage and handling instructions to ensure that, if processed per the SRM instructions, a repair will meet the specified design strength, stiffness, and durability. Unlike typical metal materials, such as 20204-T3 or 7075-T6 aluminum alloy sheet or plate, composite material mechanical properties depend on correct processing in the repair arena. Many metal repair materials can be obtained in the final condition, whereas composite materials are processed in the repair shop. Therefore, it is essential that composite material ingredients are shipped, stored, handled, prepared, and processed per source document instructions. In the case of bonded repairs, the actual repair processing provides the final material condition.

Bolted repairs of composite parts are very similar to bolted repairs of metal components. Precured composite repair plates or metal repair plates are used along with metal fasteners. Repair process instructions for bolted repairs are very similar to bolted repairs to metal components, with similar QC and postrepair inspection requirements.

<u>Safety Message</u>

The approved sources of technical data, maintenance and repair instructions, and guidelines and regulatory requirements contain information vital to proper aircraft maintenance and repair. It is essential to be familiar with the specific aircraft structure drawing system and approved maintenance methods for the particular component in question.

The use of inappropriate or unapproved maintenance instructions for a given part, including inspection and repair processes or alternate repair materials used on other structures, is not allowed. While it is not included in the roles of inspectors and technicians to be fully cognizant with the CFRs, ACs, and ADs, it is appropriate that all members of the repair team be aware that these regulatory requirements exist and are understood by at least one person in the repair process.

Any lack of understanding of the structural detail, or deviation from the approved data, maintenance and repair instructions or regulatory requirements can lead to unacceptable maintenance procedures and a defective repair.

<u>A.2.3 Demonstrate an Awareness of Course Principles Applied to Composite Damage Disposition and Repair</u>.

<u>A.2.3.1 Identify a Typical Structural Component and Understand the Specific Configuration and Materials Used for Fabrication of the Damaged Component</u>.

Suggested laboratory equipment that would help in presenting this material are:

- A video presenting examples of field repairs to an actual configured structure

- Various damaged configured composite panels

- Data book containing component information, component master worksheet, and SRM data for each damaged panel

- Marker pens

- Flashlight

- Tap hammer

- P/E ultrasonic equipment

- Worksheets for recording repair dispositions, repair processes and in-process QC plans, and postrepair approval plans.

Divide the class of students into repair teams each with a designated maintenance engineer, inspector, and technician.

Each student repair team will identify the component, location on the aircraft, component construction, and details of materials used. They are expected to select, from the laboratory data book, the correct data for the damaged component. They are expected to understand the proximity of the damage to details such as core ramps and stringers and the implications that any close proximity might bring about.

All members of each team are expected to understand the component configuration and construction details, including materials used (e.g., sandwich facesheets and core, or stiffened laminate lay-ups, and adjacent substructure such as a stringer or a rib).

The maintenance engineer must determine from the provided component master worksheets if previous maintenance or repairs could complicate any new repair to the damaged area.

A.2.3.2 Perform a Damage Assessment and Map the Damage as Accurately as Possible Using Visual Inspection, the Tap Test, or P/E Ultrasonic Equipment.

The designated technician must first visually discover the damage on the exterior surface of the damaged panel. A flashlight should be used to enhance visual detection capability. The extent of the damage should be assessed and marked. The designated technician will also examine the back side of the component to determine if there is any through-penetration or damage to any substructure. The damage findings will be provided to the designated inspector for a more complete damage assessment.

First, the designated inspector will consult the SRM documentation for any specific inspection procedure required for the component. The inspector will then re-assess and map the extent of the damage using a tap test or an instrumented NDI procedure. The inspector is expected to use the P/E ultrasonic equipment, but a tap test method may be used if the source documentation allows this method. When using the P/E equipment, the inspector is expected to understand what signals an undamaged structure will return before mapping the extent of the damage. The source documentation may allow a tap test if it has been shown that a defect or damage that is less than or equal to the component maximum ADL can be reliably found. Both sides of the component are to be examined to fully understand the extent of the damage. The extent of the damage is to be mapped and the findings reported to the designated maintenance engineer.

Identify and interpret the SRM information to understand the component ADL, and review any repair options contained in the SRM, based on the mapped damage (for the designated engineer only).

The designated maintenance engineer is expected to consult the specific SRM information to determine if the damage is within the ADLs for the component. The engineer is expected to determine if the damage is within any designated critical area of the component. If the damage is determined to be beyond the ADLs for the component, the engineer is expected to choose an approved permanent repair from the SRM data to restore the component stiffness and strength. The repair damage size limits contained in the SRM data will be consulted to ensure that the repair is appropriate for the present damage. The repair disposition is to be communicated to the designated technician with written instructions.

If the selection is for a bonded repair, the engineer will ensure that the technician has the correct repair lay-up instructions for the approved repair so the component strength and stiffness can be restored with a well-executed repair. The engineer will list the orientation of each ply and the order in which they are to be laid down.

If a bolted repair is selected, the engineer will list the appropriate repair plate required (e.g., material, thickness, and dimensions) and fastener details (e.g., type, sizes, and grip lengths). The technician will be provided with a diagram of the fastener pattern to be used. All of this information is either in the data book provided for the laboratory.

The maintenance engineer is expected to record the damage details and repair disposition in the component master worksheet for the specific component.

A.2.3.3 Use Source Documentation Represented by a Training Repair Manual That Simulates an OEM's SRM to Understand the Component.

The designated technician is expected to consult the specific training repair manual (TRM) information to write an appropriate repair procedure and in-process QC plan for the chosen repair. It is expected that the designated technician will consult the appropriate TRM section for the specific component repair instructions.

If a bonded repair was chosen by the maintenance engineer, the technician is expected to list the details of moisture removal; surface preparation; damage removal, including ply scarf machining and core cutting (if applicable); and preparation of repair materials. The technician is also expected to list the safety equipment required.

If the repair disposition is for a bolted repair, the process steps and special equipment (e.g., drill bits, reamer type and size, backing plate, safety equipment, etc.) necessary for such a repair will be listed. Details of surface preparation, protruding damage removal, hole drilling, and fastener installation are to be listed.

For either type of repair, the technician is expected to write a repair process plan, together with the required in-process QC plan, on the provided worksheet.

A.2.3.4 Write an Appropriate Repair Procedure and In-Process QC Plan Based on the Chosen Repair Option.

The designated inspector is expected to check the in-process repair QC data that is provided after a bonded or bolted repair. The inspector will write a postrepair inspection and approval plan, assuming that a repair has been performed. The inspector is expected to refer to the information that pertains to this subject, listing the in-process QC information that would be recorded to ensure knowledge of an actual repair. The inspector will record details of ply lay-up and dimensions, debulk cycle (if used), cure cycle with details of vacuum and temperature profiles, fasteners (types, size, grip lengths, and fastener patterns), and repair plates (material type and dimensions) if it was a bolded repair. The inspection techniques to be used to determine the

adequacy of a damaged component repair are to be listed. The inspector will be expected to understand the available inspection technique limits, and the limitations must be recorded.

A.2.3.5 Write an Appropriate Postrepair Inspection and Approval Plan.

The designated engineer is expected to record what procedure must be followed in the event that an anomaly is detected in the repair, consulting the appropriate TRM section.

In the event the repair is deemed acceptable, the technician is expected to record, in the provided worksheet, the procedure that is necessary to restore the repaired component to service, after consulting the provided data book for the restoration instructions.

Safety Message

All aspects of composite repair are interlinked, and teamwork is essential to provide a safe damage or repair disposition. It is essential that each team member recognizes his/her limitation to:

- Disposition a damage
- Determine the repair design and process that is substantiated by approved data
- Perform the approved repair process
- Inspect the completed repair

If any of the above repair aspects contain errors or flaws, the aircraft may be dispatched with a deficient or unapproved repair.

A.3 DAMAGE DETECTION AND CHARACTERIZATION.

A.3.1 RECOGNIZE COMPOSITE DAMAGE TYPES AND SOURCES.

A.3.1.1 Identify Sources and Characteristics of Damage to Composite Sandwich and Laminate-Stiffened Structures.

Sources of damages to composite structures can be roughly sorted into two types; those occurring during fabrication and processing, whether during original manufacture or repair, and those in service.

Fabrication and in-process damages can be:

- Processing anomalies and in-process handling damages

- Fabrication processing can result in defects in composite parts, and carelessness during postcure handling and assembly can also result in damages. The anomalies resulting from these kinds of processing variations and handling mistakes are usually discovered by rigorous inspections by the OEM and are typically subjected to a Material Review Board (MRB) for disposition. Any detected processing anomalies or handling damages

are compared to acceptable defects, and dispositions are made. Accepted processing anomalies or handling damages can be expected to be discovered in service and will be covered by ADLs. Similarly, repair processing or careless handling in the maintenance depot can result in the same kind of anomalies or damages that can occur during the original manufacturing process. Some of the defects listed below may exacerbate in-service damages, or grow as a result of temperature excursions or mechanical loadings. They also can grow as a result of, or lead to, moisture ingression.

- Processing anomalies such as voids, delaminations, and porosity can occur during the cure process, and may be the result of poor tooling, insufficient ply consolidation, low autoclave pressure, loss of vacuum, or nonconforming material (e.g., improper resin content or formulation).

Poorly designed or installed cure tooling can lead to bridging and lack of proper pressure. This can result in areas of porosity, voids, and delaminations.

Some of the latest, tougher materials require ply consolidation prior to the cure cycle. This is due, in part, to the material viscosity, and depending on the number of plies in a lay-up, several compaction cycles may be necessary. In these cases, insufficient ply consolidation or compaction prior to cure can lead to voids, porosity, and delaminations.

During the cure cycle, any loss of vacuum, autoclave pressure, or temperature can result in anomalies, such as voids and porosity, producing potentially lower than required mechanical properties. An improperly cured part may also have lower than required thermal stability and associated long-term durability.

Inclusions can occur when insufficient care is taken during ply lay-up. Objects, such as pencils, rulers, and backing paper, have been found in laminates during postcure inspection. Other items found in cured parts, such as separator film and other cure aids, were left there by careless technicians and undetected by sloppy in-process inspection.

Defects, such as edge damages, dents, delaminations, fastener hole damage, and damage to LSP systems, can result after cure, during part handling, machining, and assembly. Composite parts are more vulnerable to impacts than metal parts, and care must be taken during handling. Incorrect machining and drilling parameters, such as feed rates and drill speeds, can lead to delaminations in and around holes and part edges.

Postprocessing inspection will detect the vast majority of manufacturing anomalies, and for those that are detected, liaison dispositions will be made prior to part delivery. Dispositions could be repairs, defect acceptance, or scrap the part. In the event that the defects were determined to be acceptable, it is essential that the associated liaison records are kept available.

The airlines and repair stations generally do not have inspection techniques as sophisticated as those available to the OEM. Hence, anomalies present in repair patches and bondlines are more likely to go undetected. To make up for the less capable inspection techniques, repair in-process controls must be carefully implemented and monitored so anomalies are kept to a minimum. There are ongoing efforts to bring more affordable, portable, sophisticated inspection techniques

to operators and maintenance providers so both damage assessments and postrepair inspections can be more accurate. These efforts are, in large part, due to the increasing use of composite materials in critical components of new, large commercial aircraft, such as the Airbus A380 and the B-787.

A.3.1.1.1 In-Service Damages.

Aircraft parts can be damaged on the ground or during flight operations. These damages can be the result of dropped tools, service vehicle impacts, aircraft handling accidents, impacts from maintenance stands, dropped parts, local pressure from being walked on, incorrectly installed removable fasteners, bird strikes, foreign objects or debris thrown during takeoff and landing. Impact damages from these events are considered the most important in-service damages to detect. In-service damages can range from minor to flight safety critical, the latter is seen in figure A-8. It is essential to detect flight safety-critical damage before flight loads are imposed on the damaged structures.

Figure A-8. Damaged In-Service Composite
(Courtesy of Abaris Training Resources Incorporated)

Solvents and other fluids, such as grease, fuels, oils, hydraulic fluids, water, cleaning fluid, deicing fluid, and salt spray, can be absorbed by composites, reducing its mechanical properties. Property reductions, due to fluid or moisture absorption into undamaged composite components, are typically taken into account during part design; therefore, repair is not usually required. However, if a part is damaged, the effects of moisture can be much more severe and may lead to reduced properties that may require repair.

Structural components located near engines or sources of aerodynamic noise are susceptible to sonic fatigue, although the composite parts are more resistant than metal parts. Some composite components may be subject to sonic fatigue, such as engine cowl, duct, and strut components, as well as trailing-edge panels and flaps. The sonic environment is taken into consideration during the design phase of these components, and since the sonic fatigue performance of any component depends on the actual structural configuration, analyses and tests are performed to validate the

designs. In the event that high-frequency noise produced by propulsion units and aerodynamic disturbance is higher than the designed allowable, damage may result from loose or broken fasteners, disbonds, delaminations, and through-the-thickness cracking emanating from attachment details.

High-heat sources can affect composite parts, such as thermal deicing ducts (typically located in the leading edges of wings), powerplants and auxiliary power units, hot-air feed ducts, air-conditioning units, and hot-air duct failures. Composite parts that are expected to be exposed to high-heat sources are designed for this exposure. For example, some fixed-wing, leading-edge panels are fabricated with materials cured at 350°F, while the fixed-wing, trailing-edge panels, which are not exposed to high-heat sources, are typically fabricated with materials cured at 250°F.

If a composite part is heated above its cure temperature, not only are mechanical properties, such as stiffness and compression strength, seriously compromised, the epoxy resin may blister or even burn. This could result in fiber exposure and laminate cracking that could provide moisture or fluid ingression paths. Apart from obvious burn damage, discoloration of the part finish may indicate a high-temperature exposure. If paint discoloration is detected, there may be damage beyond what is visible. In these cases, a tap tester or an instrumented NDI technique should be used to determine the extent of the damage, although the total amount of damage may be difficult to assess.

Damage may also arise from maintenance activities such as cleaning and paint stripping. For varying reasons, an aircraft may undergo paint removal several times during its operational life. There are various methods used to remove paint, and for most of them, utmost care must be taken to avoid damage to the composite. Damage may include resin loss and exposed or loosened or damaged fibers.

A.3.1.1.2 Environmental Damages.

Hail, lightning strikes, ultraviolet (UV) radiation, high-intensity radiated fields, rain erosion, moisture ingression, and ground-air-ground cycles (temperature, pressure, and moisture excursions) can all cause damage to composite components.

Ground hail can seriously damage sandwich components that have relatively thin facesheets. A severe storm at Dallas-Fort Worth International Airport in February 1984 resulted in extensive damage to aircraft elevators, ailerons, and fixed structural components such as trailing-edge panels. The hailstone energy was between 240 and 360 in.-lb. The level of energy generated by the 2-in.-diameter hailstones also dented the metal fuselage skins. Some of the aircraft damaged by the hailstorm were not returned to service for 2 weeks while on-airplane repairs were performed. As a result, cost-benefit studies were performed at Boeing and other OEMs, which compared the increase in operating costs due to higher structural weight to the lost revenue for downtime to perform structural repairs. Due to the lost revenue caused by on-airplane repairs, minimum sandwich facesheet gages were established to provide an appropriate level of impact resistance for permanently attached composite structural parts that are likely to be subject to ground hail.

Lightning strikes can inflict severe damage to aircraft components unless protection systems are employed. As an example, a Pan American B-707 aircraft was struck by lightning on approach to Washington Dulles International Airport on 8 December 1963. The aluminum sheet of the wing surface was vaporized by the intense local heating caused by the lightning discharge. This event caused the fuel vapor in the wing fuel tank to ignite, and the aircraft broke up and crashed. The federal regulations for protection from lightning strikes can be found in 14 CFR 23.867 and 23.954 for commuter and light aircraft; 14 CFR 25.581 and 25.954 for transport aircraft; and 14 CFR 27.610 and 27.954 for rotorcraft.

Unless protected, composite structural components will usually suffer even more damage from lightning strikes than aluminum structures. Composite materials are typically significantly less conductive than aluminum. This low conductivity allows significant portions of lightning current to flow (arc) into onboard systems and, therefore, provides less shielding of onboard electronic systems than metal parts. The materials, design configuration, and interfaces necessary to achieve the desired damage resistance for a composite structural component exposed to lightning depends on the component's location on the aircraft. Commercial aircraft are zoned for the likelihood and magnitude of direct lightning strikes. Maximum lightning strike energy levels have been established, per aircraft zone, based on the likelihood and magnitude of a direct attachment and the ensuing swept stroke.

Composite components used in high-intensity zones must be protected. Extensive damage can be prevented by using protective systems such as metal picture frames, expanded metal foil, and embedded metal fibers. Without protection from lightning strikes, not only can arcing of current into the interior occur, but damage to the composite can range from surface ply burns to complete laminate burnthrough. If metal fasteners are present in the composite, the lightning strike can attach to them. Therefore, it is necessary to prevent arcing or sparking between them by encapsulating the fastener nuts or sleeves with plastic caps or polysulfide coatings. After repairs, if any of these protective systems are present, they need to be restored.

While UV radiation has little effect on carbon fibers, it can degrade epoxy resins and, as a result, compromise the integrity of the composite part. UV radiation can cause surface embrittlement of unprotected polymeric composite material. Although carbon fibers restrict the penetration of UV radiation, the long-term exposure to UV radiation can inevitably lead to a degraded surface, which may serve as a site for brittle crack initiation. If composite parts are exposed to UV radiation, they are typically protected by an opaque finish layer, which must be restored after repair.

Damaged composite parts can ingress fluids from surrounding environments. The ingression can come through loss of protective paints and impact damage to the laminate or facesheet. On occasion, sealant systems break down on sandwich components and unforeseen damage occurs due to moisture or fluid ingression into the core. For aluminum core in metal bond parts, the moisture or fluid can lead to corrosion, resulting in loss of the core material. In aramid cores of composite sandwich parts, the moisture or fluid can seriously degrade mechanical properties such as stiffness and shear strength. Paint cracking, caused by temperature excursions and freeze-thaw cycling can also provide paths for moisture ingression.

A.3.1.2 Describe Damage Types and Their Significance to Structural Integrity.

Different types of composite component damage can significantly affect the residual strength of the structure and the resulting damage size that can be allowed. The damage state cannot usually be conclusively determined from visual inspection of the part, nor can it be conclusively determined from in-service NDI techniques such as the tap test or P/E-instrumented method. While these NDI methods can map out delaminated areas and visual inspections may determine the extent of through-the-thickness cracks, damage such as matrix cracking, fiber breakage, and multiple plane delaminations cannot be reliably mapped. For this reason, various assumptions about the extent of damage in a part have to be made when determining if a given damage is acceptable for continued flight without repair.

The diagrams shown in figure A-9 illustrate potential damage based on impact energy. What is not shown in the diagrams is the additional effect of velocity associated with impacts. For example, low-speed, high-energy impacts can leave large-area delaminations and substructure damage without much exterior damage. On the other hand, high-speed, low-energy impacts (such as bullets) penetrate without leaving wide-area delaminations and potential substructure damage.
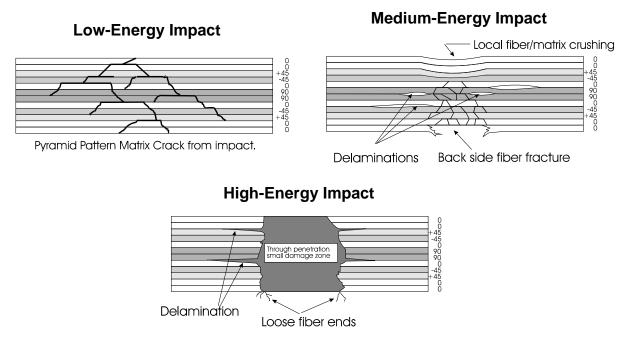

Figure A-9. Impact Damage Types
(Courtesy of Abaris Training Resources Incorporated)

A.3.1.2.1 Matrix Imperfections.

Matrix imperfections (cracks, porosity, blisters, etc.) usually occur on the matrix-fiber interface or in the matrix parallel to the fibers. These imperfections can slightly reduce some of the material properties but will seldom be critical to the structure, unless the matrix degradation is widespread. Accumulation of matrix cracks can cause the degradation of matrix-dominated

properties. For laminates designed to transmit loads with their fibers (fiber dominant), only a slight reduction of properties is observed when the matrix is severely damaged. Matrix imperfections may develop into delaminations during service, which is a more critical type of damage. Porosity usually occurs during the fabrication cycle, whereas blisters and microcracks can occur as a result of temperature excursions, such as those produced by local heat sources or freeze-thaw cycles.

The greatest concern with these types of damage is the associated breakdown of surface paints and protection layers, thus providing paths for moisture and fluid ingression, especially for sandwich parts with thin facesheets.

A.3.1.2.2 Delamination.

Typically, delamination forms at the interface between the layers in the laminate, along the bondline between two elements, and between facesheets and the core of sandwich structures. Delaminations may form from matrix cracks that grow into the interlaminar layer, from processing non-adhesion, or from low-energy impact. Under certain conditions, delaminations can grow when subjected to repeated loading and can cause catastrophic failure when the laminate is loaded in compression. The criticality of delaminations depends on:

- Length and width dimensions

- Number of delaminations at a given location

- Location—in the thickness of laminate, in the structure, proximity to free edges, stress concentration region, and geometrical discontinuities

- Loads—delaminations behavior depends on loading type. They have little effect on the response of laminates loaded in tension. Under compression or shear loading, however, the sublaminates adjacent to the delaminations may buckle and cause a load redistribution mechanism, which may lead to reduced strength or stiffness and possibly structural failure.

A.3.1.2.3 Fiber Breakage.

This defect can be critical because composite structures are typically designed to be fiber-dominant (i.e., fibers carry most of the loads). Fortunately, fiber failure is typically limited to the zone of impact contact and is constrained by the impact object's size and energy. One exception can be a high-energy, blunt impact that breaks internal structural elements, such as stiffeners, ribs or spars, but leaves the exterior panel laminate intact.

A.3.1.2.4 Cracks.

Cracks are defined as a fracture of the laminate through the entire thickness (or a portion of the thickness) and involve both fiber breakage and matrix damage. Typically, cracks are caused by impact events, but can be the result of excessive local loads (either in the panel acreage or at a fastener hole). Unlike metals that fatigue crack at fairly low-level loads, which occur frequently

in normal operations at small imperfections, at a fastener hole or a scratch or gouge in an edge, composite laminates tend to crack under less-frequent high loadings. Cracks without associated delaminations in the same area may have a higher-stress concentration than a circular hole. Damages that include delaminations or matrix damage along with cracks can have less severe stress concentrations and behave more like a soft inclusion in the laminate.

A.3.1.2.5 Nicks, Scratches, and Gouges.

Nicks, scratches, and gouges are not critical if the damage is limited to the outer layer of resin without any damage to the fibers. If the fibers are damaged, they must be treated as a crack in the effected plies. The overall effect depends on the length and depth of this type of damage as related to the total laminate or facesheet thickness.

A.3.1.2.6 Dents.

Dents are typically caused by an impact event. Damage to the structure can consist of one or more of the following: sandwich core damage, facesheet delaminations, matrix cracks, and fiber breakage and disbonds between the facesheets and core. Dents in thin facesheets over core areas often only involve core damage. Dents in solid laminate areas fastened to substructure (e.g., edge band areas of sandwich panels) can have associated damage to the substructure. In these cases, if accessible, it is essential to examine the back side of the part and any adjoining substructure.

A.3.1.2.7 Punctures.

A puncture is defined as an impact damage that penetrates the facesheet or laminate. The edge of the puncture may be relatively clean, or may be ragged, depending on the type and energy of the impact event (figure A-10). In either case, there may be associated delaminations, matrix damage, and fiber breakage outside the puncture edges.

Figure A-10. High-Energy Impact Damage on In-Service Composite
(Courtesy of Abaris Training Resources Incorporated)

A.3.1.2.8 Combinations of Damages.

In general, impact events cause combinations of damages. High-energy impacts by large objects (i.e., turbine blades) may lead to broken elements and failed attachments. The resulting damage may include significant fiber failure, matrix cracking, delaminations, broken fasteners, core damage, and disbonded elements. Damage caused by low-energy impact (caused by tire bursts, dropped tools, or on-ground hail) is usually more contained, but may also include a combination of broken fibers, matrix cracks, core damage, and multiple delaminations. Most damage sites that were detected visually will yield more extensive damage when inspected by NDI techniques.

There is a potential for in-service impact events to cause damage that is not detectable by visual inspection. Low-velocity impact events, particularly those caused by large-diameter, blunt impactors, such as tool drops or vehicle collisions, may cause internal laminate delaminations and fiber damage that would not be visible. To compensate for impact damages that may go undetected, OEMs, such as Airbus and Boeing, design primary composite structural components to be able to carry regulatory loads with barely visible impact damage (BVID). BVID is the smallest damage that may be detected by the naked eye from a specified distance (e.g., 5 feet) in good lighting conditions. Damage that is considered easily visible is termed visible impact damage (VID) and may be serious enough to reduce the structural capability below design loads. For VID damages to primary structural components, it is essential that these damages be detected during routine maintenance. Due to the increased use of laminate-stiffened composites in aircraft primary structures, visual and more rigorous NDIs of critical components may be directed by MPDs.

It is essential to understand that a known damage event to a composite component needs to be further investigated. Knowledge of the damaged part's structural configuration will provide clues as to potential invisible damage, such as to an adjacent stiffener or rib, a disbonded substructural element, or crushed sandwich core.

Some experimental evidence suggests that, for relatively small damage sizes, impact damage is more critical than other defects. On the other hand, some test results for panels containing damages greater than 2 inches indicate that large holes or penetrations are at least as severe as equivalent-sized impact damage.

A.3.1.2.9 Damaged Fastener Holes.

Improper hole drilling, poor fastener installation, and missing fasteners may occur in manufacturing or maintenance. Hole elongation damage can occur due to repeated load cycling in service. Damage to fastener holes can also happen during maintenance when removing or replacing screws or quick-release fasteners. Such issues can effectively extend the size of the hole and result in reduced in-plane bypass, fastener pull-through, and bearing strengths. Repair will require replacement with the next size fastener.

A.3.1.2.10 Edge Erosion.

The edges of laminates may be damaged as the result of the effects of airflow over the structure and the impingement of debris, rain, etc. This type of damage is typically erosion of the laminate material both through the thickness of the part and in the direction parallel to the airflow. This can expose the fibers, leading to potential fiber microbuckling and a path for moisture ingression.

A.3.1.3 Understand the Technical Data and Structural Analysis Necessary for Repair Design and Process Substantiation.

Repairs performed while an aircraft is registered in the USA are controlled by the CFRs and ACs devoted to the maintenance of civil aircraft. EASA has similar regulations for commercial aircraft registered in Europe. For an example of the CFRs, the following are of particular interest: 14 CFR Part 43 Maintenance, Preventive Maintenance, Rebuilding, and Alterations and 14 CFR Part 145 Repair Stations.

14 CFR Part 43, and the ACs associated with it, specify methods that were approved for repair and alteration. If a repair is not already approved, it must be described in the OEM's SRM or be given special approval by a DER. Repair station certification requirements are given in 14 CFR Part 145. To obtain FAA certification, a repair station must submit documentation to demonstrate the skills of personnel, inspection procedures, and the necessary facilities and equipment.

The intent of repairs to structural components is to restore the original mechanical properties, such as strength and stiffness, regardless of the component loadings or whether the repair is to be bonded or bolted. This requires using the strength and stiffness data for the damaged part material and repair materials, and the strength and stiffness data for the repair fasteners, if used. Original part strength and stiffness data used for OEM repair designs is derived from the database used for the original type certificate.

For repairs designed by a DER rather than by the OEM, proper knowledge of the original part structural details is required. This knowledge should include ply and core material properties, ply lay-ups, and core orientation. In some cases, reverse engineering is employed to design an appropriate repair for a damaged part. In this process, the engineer, with knowledge of the part structural details, materials, and lay-ups, calculates the original part strengths and stiffness and then designs a repair to restore those properties. This practice is not recommended due to the difficulties in obtaining approved data for these repair designs.

The CACRC has a task group working on analytical repair techniques intended to provide means of repair design validation. To date, this task group has yielded some educational materials that provide an introduction to repair analysis. It is important to realize that all analysis methods must be substantiated for the specific repair design and process details of interest. Qualified materials and supporting structural repair data collected at sufficient size scales are needed for this substantiation.

Variability, for both the original component materials and the repair materials, is accounted for in the material allowables through statistical analysis of the coupon or element test data. For design values, material variability is accounted for by reducing the test data by scatter factors appropriate for the failure modes and for the size of the database. Variations allowed in material and process specifications are usually accounted for through coupon tests on pieces cut from panels that were fabricated with a range of process variables (process temperature ramp rates, material out-times, etc.).

Typically, design values for both the original materials and the repair materials are proprietary and reside at the OEM. There is information available from other sources, such as the Composite Material Handbook 17, that may be used to control qualified materials and design a repair. As mentioned previously, the repair design must also be structurally substantiated and approved by a DER.

Nearly every repair consists of a patch and a joint through which the loads are transferred to the patch. A bolted patch is shown in figure A-11. Joints are usually either bonded or bolted, though sometimes a combination of both is used. Analytical procedures for bolted or bonded joints tend to be complex and can require computer programs for their solution.
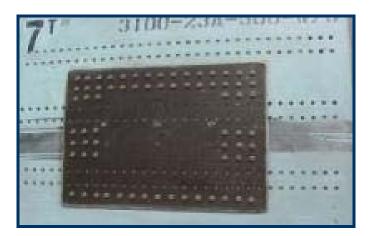

Figure A-11. Bonded-Bolted Repair
(Courtesy of Abaris Training Resources Incorporated)

Analysis procedures for bolted and bonded joints are also described in CMH-17. Selected procedures that represent typical generic repairs are given for monolithic skins, sandwich structures, and substructures. There are numerous variations of these procedures for specific situations. It may be necessary for maintenance personnel to modify the procedures as required for specific materials or geometric conditions. However, it is imperative that the procedures in the approved repair documentation are followed. Approved repair documentation may be found in an SRM, engine manual, AMM, or MPD. Some 14 CFR Part 23 OEMs may provide approved repair documentation other than the aforementioned.

To ensure that required standards are met, all materials and processes used to repair aircraft structural components must conform to approved specifications. This means that sufficient research and testing were performed to ensure that the materials and processes employed by the

end user will, if the materials were stored and handled correctly and the processes strictly adhered to, provide adequate repairs per the repair designs. Approved sources of repair documentation, such as the SRM, typically call out materials by the OEM specification numbers. Process instructions in the SRM are usually laid out step by step.

Approval of an aircraft structural component repair requires the following:

- The repair is designed by a qualified engineer who used approved data and materials to restore the part's original strength and stiffness capabilities. Repair design criteria for permanent repairs are fundamentally those that are used to design the part that is to be repaired. Repair design criteria for temporary repairs can be less demanding, but may approach permanent repairs if the temporary repair is on the aircraft for a considerable amount of time (e.g., an interim repair).

- The repair materials are stored and handled within the bounds of the approved documentation. Using improperly stored or handled adhesives, sealants, prepregs, or wet lay-up ingredients may result in structurally unsafe aircraft components. AC 145-6 defines repair station requirements for approved repair material purchasing and QC (e.g., testing, storage, and handling).

- The repair is processed per the approved process instructions and the required in-process controls were carried out. It is essential that the approved in-process controls be strictly carried out. All equipment used during repair processing, including damage and repair inspection equipment, must be certified and maintained to the required specifications.

- The material and in-process records and the postrepair inspection are judged acceptable. Once a repair is complete, it must be inspected and approved by an authorized maintenance organization inspector before the part is returned to service.

A.3.1.4 Distinguish Differences in Repair Disposition Procedures for Those Damages Covered or not Covered by Source Documentation.

When damage is discovered and mapped, the first step in any damage/repair disposition is to consult the approved repair documentation (such as the SRM) for ADLs and repair designs.

If the damage is less than, or equal to, the ADL for the specific component, the source documentation procedure for sealing the damage and restoring the component to operation should be followed. As an example, for ADLs within Boeing SRMs, the instructions will typically include moisture removal, the use of aluminum speed tape and sealant, and restoring the paint and any other protection system.

If the damage is larger than the ADL and there is an approved repair available, the procedures defined for damage clean-up, moisture removal, surface preparation, repair processing and postrepair inspection should be followed precisely.

A typical Boeing SRM will contain a number of repair designs for specific components. For PSE-designated structures, the categories are A, B, and C repairs. For non-PSE structure, these

are categorized as temporary, time-limited, and permanent. These repair categories are based on the repair materials and their exhibited mechanical properties and durability. For example:

- Repairs cured at room temperature or 150°F typically will be classed as temporary or time-limited with small repair size limits. Repairs cured at these temperatures will usually result in durability and, in some cases, mechanical properties that are less than desired for a permanent repair.

- Wet lay-up repairs cured under vacuum at 200°F, or repairs using prepregs cured under vacuum at 250°F or higher, may, for some components, be classified as permanent. Repairs cured at these temperatures often result in mechanical properties and durability that are considered acceptable for permanent repairs. Larger size limits will be allowed for these repairs.

- Repairs using the original component material prepreg and cure processing parameters are typically classified as permanent. Repairs made using these materials and cure processes (e.g., an autoclave cure cycle), if performed correctly, will result in mechanical properties and durability similar to those of the original part. Within a typical SRM, for sandwich structural components that are easily removed from the aircraft, there are typically no size limits on these repairs except in critical, highly loaded areas (e.g., local to fittings). For example, in the SRM repair instructions for the Boeing 777 rudder skins, there are three zones. For rudder skin zones 1 and 2, there are no size limits for repairs using the original prepreg and process. For zone 3 (those areas around the hinge and actuator fittings), the repair instructions state "Critical area, ask the Boeing Company for the repair information."

- Autoclave-cured repairs are not normally offered within the SRM for composite components that are not easily removed from the aircraft, or may be outside the size limits of the autoclaves typically used by operators or MROs.

- If no approved repair is available for the specific damaged component, the maintenance engineer either contacts the OEM for an approved repair or designs a specific repair and obtains design approval from a DER. In the event a repair limit is established by operators or MROs, it must be approved by the OEM or DER, and in due course, may be added to the repair instructions for the specific components in the SRM.

- In the event of an operator- or MRO-designed repair, approved data must be used for the design, and approved materials and processes must be used for the repair.

- An SRM repair design for a specific component may not be used on another component unless approved by the OEM. In some cases, the SRM may not contain a repair design for a specific component. In such cases, it is probably due to no previous damages having been reported for that component. The damaged component may employ the same base materials and lay-ups as another component for which an approved repair is available, but approval to use another component's repair must be obtained from the OEM.

A.3.1.5 Describe the Regulatory Approval Process for Damages not Covered by Source Documentation.

In the event that the damage is larger than the ADL in the SRM, and an approved repair design is not available, the maintenance engineer has several options, as described in the following sections.

A.3.1.5.1 Contact the OEM for an Approved Repair.

In this case, the damage evaluation is transmitted to the OEM, and a specific repair will be designed by the OEM using approved data. In critical areas (e.g., the areas around a heavily loaded fitting), a more detailed damage evaluation may be required before the OEM can design a repair. In the event that a stiffened skin panel is damaged, inspection of the back side and adjoining substructure may be requested. This process can take several days just to obtain the approved repair, and the actual repair may take another day. To expedite the process, operators are urged to provide the most comprehensive damage evaluation possible.

This option usually involves revenue loss if the damage was discovered on the ramp during operations. In some cases, approval may be granted to fly the aircraft, without passengers, to a maintenance base for the repair. If this is not possible, qualified repair personnel must travel to the damaged aircraft with the appropriate repair equipment and materials.

If the damage is discovered during a routine maintenance event, then all appropriate personnel, equipment, and materials will typically be on hand, and no operator revenue loss will be ensued, unless there are significant delays obtaining either materials or the approved design from the OEM.

A.3.1.5.2 Replace the Damaged Part.

This is an option that is only suitable for removable parts such as flight control panels and easily removed secondary structural panels.

If damage is discovered on the ramp and replacement parts are available, the most likely course of action is to replace the damaged part with a spare or leased part. The damaged part will be shipped to the operator's maintenance depot or to an MRO for a permanent repair. Then, this repaired part can be put into storage for redeployment at a convenient time or stored for another occasion.

In the event that a leased part is used, the part would be replaced at the earliest time, due to the high cost of part leasing. This option requires the use of approved replacement parts. These can be repaired, remanufactured, or after-market parts.

Any remanufactured or after-market parts must be approved before use by the cognizant regulatory body or their delegated authority (e.g., DER).

A.3.1.5.3 Prepare a Specific Repair for the Damage not Covered.

Repair schemes may require one or more drawings to be created and may cover large repairs. They may be produced under delegated design authority or with the assistance of the OEM. These repair schemes may be for a specific, single, large damage or for repetitive repairs to cracks, wear, or deterioration on all similar aircraft in the fleet.

In the event of an operator- or MRO-designed repair, any non-SRM repair must use approved data for the design and approved materials and processes. They are usually classified as permanent repairs. However, they may be time limited or temporary, in which case, it is essential that this information is clearly stated on the repair drawings so the appropriate planning action can be taken.

In some instances, such as damage to a PSE that is not covered by the SRM, even though there is an in-house repair scheme available, an adequate damage disposition or repair design will require evaluation by the OEM.

In some cases, if the damage is discovered on the ramp, the maintenance engineer may determine which option has the least impact to the operator's revenue, and choose that option. However, regardless of the option chosen, the repair design, repair materials and process, or replacement part must all be approved by the delegated authority.

As non-SRM repair designs are created and approved, they may eventually be incorporated into the SRM for specific components. This process may take a while, but it is done to provide all operators with approved designs, rather than have them run through the lengthy repair design and approval process for non-SRM repairs. The time to obtain OEM-approved, non-SRM repair designs can be, even if coordination is expedited, two or more days before the repair designs are provided and the actual repairs are performed.

A.3.1.6 Damage Laminate Coupons in a Controlled Laboratory Environment and Visually Inspect the Extent of the Front and any Back Side Surface Damage.

Helpful laboratory equipment is listed below.

- Drop tower

- 10-lb (4.5-kg) tupp

- Support frame

- Four 2.0-lb (1.0-kg) weight bags

- Flashlight

- Magnifying glass

- Black marker pen

- 15- (38-) by 15-inch (38-cm) 16-ply laminate test panels painted with grey primer and grey enamel on the front and yellow primer on the back.

The laboratory will commence with the instructor damaging flat laminate panels with sufficient energy to promote damage to all plies in the laminate without through penetration. The steps needed to create impact damage are illustrated in figures A-12 and A-13.

Figure A-12. Preparation for Impact Damage
(Courtesy of Edmonds Community College)

Figure A-13. Dropping Tupp to Inflict Damage
(Courtesy of Edmonds Community College)

When the instructor is satisfied with the damage state and has calibrated the drop height and tupp weight, the students will damage the panels as follows:

- Place the laminate test panel flat on the support frame with the panel center (marked by X) directly under drop-tower tube.

- Place weight bags on each corner of the test panel.

- Insert the tupp into the drop-tower tube.

- Place the weight into the drop tower, raise the weight to a height of NN^2 inches (cm), and set the locking mechanism.

- Ensure that one student is in position to activate the catching mechanism that will stop the weight before it can rebound onto the specimen.

- Release the locking mechanism to allow the weight to fall onto the tupp causing damage to the specimen.

- Activate the catching mechanism to stop the weight before it can rebound onto the specimen a second time.

- Remove the weight bags from the test panel.

- Remove damaged test panel from the impact apparatus.

- When the students have damaged all the panels, the instructor will demonstrate the visual inspection of one panel. When the instructor has inspected both surfaces of a test panel and outlined the damaged area, the students will each inspect and map the damage on one panel as follows:

  - Visually inspect the front surface of the test panel for damage using a flashlight and magnifying glass, as shown in figure A-14.

  - Mark the perimeter of the damage area on the front surface of the panel with a marker pen, as shown in figure A-15.

  - Visually inspect the back surface of the test panel for damage using a flashlight and magnifying glass.

  - Mark the perimeter of the damage are on the back surface of the panel with marker pen.

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[2] Prior to conducting this laboratory, the appropriate test height will be determined by experimentation to produce an appropriate amount of damage necessary for student learning.
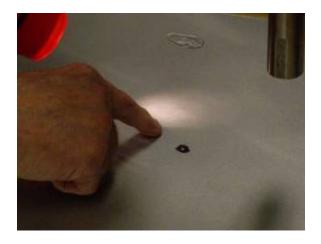
Figure A-14. Assessing Damage to Composite
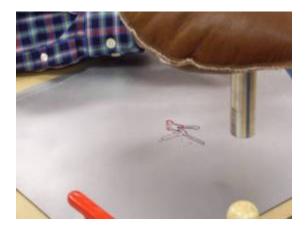(Courtesy of Edmonds Community College)

Figure A-15. Initial Mapping of Damage
(Courtesy of Edmonds Community College)

Safety Message

Composite aircraft components can be damaged by many different sources. In many cases, these components are designed to withstand damages up to levels within the bounds of performance and cost. When damages are above these levels and approved data exists for repair, source documentation (e.g., SRM) will typically provide the necessary instructions for maintenance actions. This will include the appropriate NDI methods to accurately map the extent of the damage and an approved repair that can be performed so the aircraft can be returned to service. If the level of damage is not covered by previously approved and documented data, the OEM must be contacted for repair disposition or a DER with the appropriate authority will have to develop a repair design and generate the data needed to substantiate the repair.

Damage disposition and subsequent repair designs and processes must be based on approved data, which substantiates the structural integrity. Without such data, the airworthiness of the structure is in question.

A.3.2 Describe Composite Damage and Repair Inspection Procedures.

A.3.2.1 Describe NDI Techniques Currently Available in the Field, Including an Assessment of Their Strengths and Weaknesses.

NDI techniques used in the field for composite part damage detection, damage characterization, and postrepair inspection are typically less sophisticated than those employed by OEMs for their postprocessing inspection. Some of the NDI techniques used in the field for the inspection of damage or defects in structural parts are:

- Visual inspection (surveillance and detailed)
- Coin tap or hammer (mechanical and instrumented)
- P/E ultrasonic equipment
- Bond testers
- Moisture meters
- Eddy current
- Radiography
- Air-coupled TTU
- MAUS
- Shearography
- Thermography

Operators and MROs use visual inspection as their main technique for initial detection of field damages, unless NDI techniques are specified by the maintenance planning manual or aircraft maintenance manual. Once damage is detected visually, other NDI methods are needed to map the full extent of damage. If visual inspections do not detect any damage during either surveillance (e.g., on the ramp between flights) or directed detailed visual inspections during planned maintenance events, follow-up NDIs are rare. Special inspections defined by engineering personnel after known service events (e.g., tire or engine bursts, bird strikes, structural overloads, etc.) may require both visual and other NDI methods to determine the full extent of the damage.

While OEMs typically use water-coupled TTU methods for detecting anomalies in composite bonded parts; many operators and repairers use the tap test and the P/E NDI technique for mapping damages and for postrepair acceptance inspections. Some of the major airline maintenance bases and repair technicians use the air-coupled TTU technique. This equipment is relatively new and has advantages when compared to P/E in that no couplant or contact is required, and therefore, is noncontaminating. The MAUS technique is a Boeing-developed, large-area inspection system, and is in use by such aircraft operators as the United States Air Force. This system is portable, easily set up, and integrates several traditional NDI techniques into one package. Detection capabilities include P/E, ultrasonic resonance, and eddy current. The latest version, MAUS-V has increased capacity for data processing of linear and phased ultrasonic and multifrequency eddy-current scans, and allows for even faster setups.

Some operators and repair technicians use bond testers to inspect for delaminations and adhesive disbonds in large sandwich components.

Repair technicians and operators use moisture meters, radiography, and thermography techniques to detect the presence of moisture in sandwich parts.

Most operators and repair technicians have eddy-current equipment, which is generally used to detect cracks emanating from fastener holes in metal structure without removing the fasteners. This technique is very limited when used for detecting damages within composite structures and for inspecting repairs for integrity.

Similarly, radiography (x-ray) is sometimes used by aircraft MROs to inspect critical metal parts for minute cracks. For inspection of composite parts, x-raying CFRP is difficult, because the absorption characteristics of the fibers and resin are similar and the overall absorption is low.

A.3.2.1.1 NDI Techniques Most Likely to be Used in Service to Support Inspection and Mapping of Damage and for Postrepair Acceptance Inspection.

Visual Inspection

Most, if not all, aircraft operators and MROs use visual inspection for surveillance of structural components on the ramp between flights. For those composite structures designed to be able to carry expected loads with damage that is not visual to the naked eye, this is an appropriate technique for detecting damages that may be critical to flight safety. However, there may be a rogue event (e.g., a ground service vehicle collision with the fuselage) that may cause invisible damage that could be a threat to flight safety, and to which the part has not been designed for.

If damages to structural components are not visible and may be critical to flight safety, these areas should be designated as critical, and NDI should be specified in addition to visual inspection.

During planned maintenance checks or directed inspection after known damage events, detailed visual inspections are performed. MPDs or engineering personnel may edict ultrasonic inspection of specific areas of critical components (e.g., areas adjacent to elevator and rudder actuator and hinge fittings), but all directed inspections start with a visual inspection. After a repair has been performed, the repaired area is visually inspected before using a tap test or ultrasonic techniques. The edges of the bonded repair are visually inspected for adhesive bleed out. For fiberglass repair patches, visual inspection can often detect areas of disbonded plies within the fiberglass patch. These anomalies can show through the laminate as discolored areas.

Tap Test

Most commercial and military aircraft operators use the tap test for damage detection in composite sandwich components. Tap testers can be a simple coin, a tap hammer, or an instrumented tap hammer. The tap coin and tap hammer are simple to use, and many seasoned inspectors can map damages accurately in sandwich structure with thin facesheets. When inspecting sandwich parts with thin facesheets, a gentle tap on the surface of a good area will yield a clear, high-frequency sound. A similar tap on the surface of a delaminated or disbonded area will yield a dull, low-frequency sound. As the number of plies increases, the accuracy of

the tap method decreases. For mapping damages in the edge bands of sandwich parts and laminate-stiffened parts, the tap tester tends to be ineffective. Tap testers are also used for detecting disbonds or damage in thin-faced, metal-bond sandwich or metal doublers. But similar to detection of delaminations or disbonds in composite parts, as the thicknesses of the aluminum facesheet and doubler go up (approximately above 0.04 in.), damage detection accuracy goes down. Generally, the tap test underestimates the extent of damage with an increase in facesheet thickness.

Pulse-Echo NDI

P/E is a portable ultrasonic technique that uses a liquid gel as the couplant and is particularly suitable for fieldwork, i.e., damage characterization and mapping and postrepair inspections. Similar to TTU, P/E inspection can detect small defects through the thickness of a laminate and disbonds between facesheets and honeycomb core. Compared to factory TTU inspection, P/E is substantially slower to use and is not as useful for inspection of large components. It is more useful for inspecting areas that have already yielded visual damage indications. Most commercial aircraft operators will have P/E equipment available at their maintenance bases. To obtain accurate readings and correctly interpret the results, the inspectors must be thoroughly trained in using the equipment. All ultrasonic NDI equipment must be calibrated against known standard defects before use. Each time the equipment is used, the appropriate standards should be checked with the P/E equipment. At the OEM, a standard may be the smallest defect in a specific component that can be reliably found with the specific piece of equipment. More typically, in service, the P/E equipment is calibrated before each inspection using a reference standard or on a known good area of the structural component before being used for damage mapping or repair assessment. Some of the larger repairers or MROs store composite parts that contain known defects to use as standards for calibrating their NDI equipment. The CACRC inspection working group has developed calibration standards for a variety of NDI techniques.

X-Ray

X-ray is often used to detect moisture ingression in honeycomb core of radomes and other sandwich parts and is sometimes used to detect transverse cracks in laminates.

Detection of delaminations by x-ray is difficult because they tend to be normal to the x-ray beam and thus make little difference to overall absorption. However, if a surface crack is present, delaminations can be detected by x-ray, if a radio opaque penetrant is introduced to reach the delaminated layers. Penetrant use is usually restricted to the laboratory due to potential contamination.

X-ray can detect foreign inclusions and voids if they are large enough, and water can be detected in the honeycomb cells of sandwich core.

The need for associated safety precautions, such as personnel shielding or local area personnel evacuation, may be an inconvenience for some MROs and repair technicians.

Bond Testers

Bond testers are typically used to detect defects such as delaminations and adhesive disbonds. Bond testers are instruments that use the mechanical, velocimetric, and resonance impedance methods. They measure the change in local impedance produced by a defect when the structure is excited in the 1- to 10-kHz and higher frequency range. Bond testers are readily portable; thus, they are attractive for field service. They are well suited to the inspection of sandwich structures for facesheet separation from the core if small anomalies are not considered to be important. Gross defects, such as wide-spread environmental degradation and facesheet disbonds in sandwich structure, produce readily measurable changes in resonant frequencies. Large areas can be inspected for gross defects in a very short time.

Moisture Meters

Moisture meters are often used by operators and repair technicians to detect the presence of moisture when making repairs to GFRP or aramid materials. Typical moisture meters rely on radio frequency dielectric power loss. This power loss is attributed to an increase in the conductivity of the composite due to moisture absorption. Therefore, the techniques cannot be used with carbon or any other conductive material, such as metal, or with antistatic coatings that contain carbon. This technology was supplanted more recently by thermography (digital and liquid crystal) techniques, which do not suffer from the limitations of moisture meters, as shown in figures A-16 to A-18.

Figure A-16. Moisture Meter Assembly
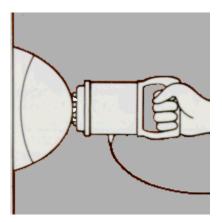(Courtesy of Abaris Training Resources Incorporated)

Figure A-17. Illustration of a Moisture Meter in Use
(Courtesy of Abaris Training Resources Incorporated)

Figure A-18. Moisture Meter Dial
(Courtesy of Abaris Training Resources Incorporated)

Thermography

Thermography is often used by airline operators for detecting moisture in the form of ice or water in honeycomb core of sandwich structures. This method can be used to detect disbonds in adhesive joints, delaminations, and inclusions in which conductivity differs significantly from the base material.

A more detailed discussion of the common nondestructive test methods can be found in the supportability chapter of CMH-17 (i.e., Chapter 14 of Rev G) and the CACRC Inspector Handbook. A discussion of improvements to in-service inspection methods can be found in a report by Dennis Roach and Kirk Rachow of Sandia National Laboratories titled "Improving In-Service Inspection of Composite Structure: It's a Game of CATT and MAUS."

Figure A-19 is extracted from the supportability chapter of CMH-17.

METHOD	STRUCTURE	DAMAGE DETECTED	RELIABILITY
Visual	All	Surface damage	Good
Tap test	Thin laminate Thin face sheet	Delaminations near surface	Good
		Lack of bond	Good
		Disbond near surface	Good
		Voids	Poor
		Blown core (core damage)	Poor
		Lack of tie-in at closure	Good
		Lack of tie-in at core splice	Poor
Ultrasonics	All	Delaminations	Good
		Lack of bond	Good
	Sandwich	Crushed core	Poor
		Blown core (core damage)	Poor
		Water in core	Poor
Radiography	All	Disbonds/delaminations	Poor
		Delaminations in corners	Good
	Sandwich	Node separation	Good
		Crushed core	Good
		Blown core (core damage)	Good
		Water in core	Good
Shearography	All	Disbonds/delaminations	Good
Thermography	All	Disbonds/delaminations	Good
	Sandwich	Water in core	Good

Figure A-19. Nondestructive Test Methods
(Courtesy of CMH-17 Task Group)

This figure compares the reliability of common in-service NDI techniques in detecting damages. The capabilities of these techniques to detect poor-quality repairs are somewhat less than their damage detection capabilities. As an example, the current in-service NDI techniques have little or no capability in assessing bondline integrity. There is encouraging research being conducted on methods to assess bondline strength, but assessment of bondline durability is another question.

A.3.2.1.2 Demonstrate and Perform Various Damage Assessments, Including Visual Inspection, Tap Test, and Ultrasonic Inspection.

Students will participate in various NDI assessments of damaged panels in a controlled laboratory environment. The damaged panels will include those damaged in the laboratory, and several configured panels, both sandwich and stiffened laminate panels. The inspection techniques to be used will be visual inspection, the tap test, and a P/E ultrasonic inspection method.

- Damaged configured panels (one thin-faced sandwich panel and one thicker-stiffened laminate panel, as available)

- Previously damaged laminate panels

- Tap coins

- P/E instrument and transducer

- Liquid-gel couplant

- Black, red, and blue marker pens

The instructor will demonstrate three inspection techniques on the damaged configured panels.

The instructor will first visually inspect the damaged areas of both panels using a flashlight. He will map the periphery of the damages with a marker pen, as shown in figure A-20.
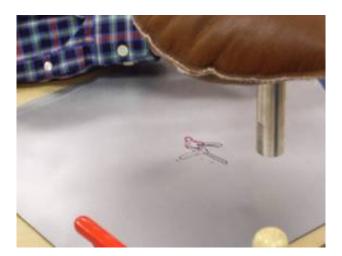

Figure A-20. Mapping Periphery of Damage
(Courtesy of Edmonds Community College)

Next, the instructor will demonstrate how to use the tap coin, as shown in figure A-21, (or hammer) and the P/E instrument in mapping damage on the thin-faced sandwich panel. He will use a blue marker pen to outline the periphery of the damage mapped using the tap coin, and a red marker pen to outline the periphery of the damages mapped with the P/E instrument. It is anticipated that there will be little difference in indications of the damage periphery, showing that the tap coin is a reliable NDI technique for mapping damage on thin-faced sandwich panels.

Figure A-21. Tap Coin Used for Mapping Damage
(Courtesy of Edmonds Community College)

Next, the instructor will demonstrate how to use the tap coin and the P/E instrument in mapping the damage on a stiffened laminate panel. He will outline the damage periphery mapped with the tap coin with a blue marker pen and the damage periphery mapped using the P/E instrument with a red marker pen. It is anticipated that the P/E instrument will result in a larger mapped damage periphery than the tap coin. This is due to the thicker laminate and presence of the stiffener on the back side of the panel. This demonstration will show that for increased laminate thickness and the presence of substructure, the P/E instrument is a more reliable NDI technique, as shown in figure A-22.

Figure A-22. Mapping Results Using P/E, Tap Test, and Visual Inspection
(Courtesy of Edmonds Community College)

The demonstration will show that the visual inspection technique can be used for initial damage detection but not for mapping damages.

The instructor will demonstrate how to use the tap test and the P/E equipment for mapping damage.

The instructor will outline the extent of the damage after each NDI technique. He will show the differences in damage outlines from the visual inspection, the tap test, and the P/E ultrasonic techniques. It is anticipated that the damage periphery indicated by the tap coin (or hammer) will be larger than what was indicated visually and the damage periphery indicated by P/E will be the largest. The use of P/E is shown in figure A-23.

Figure A-23. Assessing Damage With P/E
(Courtesy of Edmonds Community College)

When the instructor is satisfied that the students are familiar with the NDI equipment, each student will assess and map the damage on a previously damaged laminate panel using the tap test and the P/E ultrasonic equipment as follows:

- Inspect the front surface of the test panel using the tap coin (or hammer) to assess the extent of the damage already indicated by the black marker pen.

- Mark the perimeter of the damage area indicated by the tap coin (or hammer) on the front surface of the panel with the blue marker pen.

- Smear liquid gel over the front surface damage area.

- Inspect the front surface of the test panel using the P/E equipment and transducer, moving the transducer on the liquid gel.

- As the P/E instrument indicates the damage perimeter, clean away the gel and indicate the perimeter with dots using the red marker pen. When the damage area is surrounded, clean off all the gel and link up the red dots with a continuous red line.

- Compare the damage areas indicated by the black, blue, and red markers.

Safety Message

In-service inspections of composite components are necessary for safe flight operations just as they are for components fabricated from metals. In-service damages to composite parts from various sources are likely to occur during an aircraft's operational life. Per maintenance instructions, damages may be detected using visual inspection or by directed NDI. Visual indications of outside surface damage should be followed up with a back side inspection, if accessible. Even when damage is detected visually, NDI will likely be needed to determine the full extent of damage. The correct use and interpretation of NDI is required to accurately define the extent of damages so that correct damage dispositions can be made.

In the event of in-service damages, it is crucial for safe flight operations that these damages are discovered, either by operations or by directed maintenance inspections, before they become critical. After damage is discovered, the correct damage disposition must be made so the damage can either be determined to be acceptable or the damaged component can be repaired and the aircraft returned to safe flight operations. If damage is first detected using visual methods, back side surface inspection and NDI techniques, such as P/E or even a simple tap hammer, will generally be needed to determine the full extent of damage and make the correct disposition.

A.3.2.1.3 Related Documents.

SAE References (Commercial Aircraft Composites Repair Committee—CACRC)
ARP5089: Composite Repair NDT/NDI Handbook

A.3.2.2 Describe Critical Steps Necessary for Making Damage Dispositions, Including Inspection and a Draft Process for Quality Control Plan for Repair.

A.3.2.2.1 Damage Disposition.

Regardless of where damage is discovered, e.g., during routine maintenance on the ramp, or as the result of a known event, such as a tire or engine burst, bird impact, or ground vehicle collision, the damage must be assessed and the extent of the damage mapped. When damages are detected during routine inspections in the maintenance depot, the instructions within the SRM must be strictly followed, and component and material records must be maintained. When damage is found on the ramp during normal operations, the AMM or SRM may not be available, but the damage must be reported to a maintenance engineer for evaluation. Also, the normal work documents may not be available on the ramp, but records of damages, inspection results, repair dispositions, and materials used must be kept for entry into the component and materials records. Because approved repair documentation is unlikely to be available on the ramp, it is essential that operations personnel have access to qualified maintenance engineers so that proper damage or repair dispositions can be made.

To accurately map any damage discovered visually, an instrumented NDI procedure should be used. A tap test method may be used if it can be shown that a defect or damage that is less than or equal to the maximum allowable damage size can be found. If damage is found on the outside surface, the inside surface should be inspected, if accessible. It is also important to extend the inspection a distance away from the damaged area to ensure that there are no adjacent anomalies (e.g., OEM-allowed processing anomalies or other invisible damages) or previous repairs that may complicate a repair disposition.

A.3.2.2.2 Approved Repair Documentation.

After any damage is discovered and mapped, the first step in any damage/repair disposition is to consult the approved repair documentation (such as the SRM) for ADLs and repair designs. If the damage is less than or equal to the ADL for the specific component, then the source documentation procedure for sealing the damage and restoring the component to operation

should be followed. As an example, for ADLs within Boeing SRMs, the instructions will typically include moisture removal, the use of aluminum speed tape and sealant, and restoration of the paint system and any other protection system. Most allowable damages that are sealed with speed tape are required to be inspected at regular intervals to check on the condition of the seal.

A typical detailed SRM instruction for allowable damages sealed with aluminum speed tape includes:

- Check the condition of the sealing tape at a maximum of 400 flight cycles, if deterioration is found, replace the sealing tape.

- Make a permanent repair after a maximum of 4000 flight cycles or 2 years, make a permanent seal and restore protective system (primer, enamel, and any conductive coating).

- If the damage is larger than the ADL and there is an approved repair available, the approved procedures defined for damage cleanup, moisture removal, surface preparation, repair material preparation, repair processing, and inspection must be followed precisely.

- If the damage disposition is to perform a repair, a QC plan should be followed to ensure that the repair is processed correctly per the approved repair documentation.

A.3.2.2.3 A Quality Control Plan for a Bonded Repair.

To verify that all the repair process steps were performed correctly, a QC plan is recommended so a technician or inspector can ensure that the approved repair instructions were followed. The approved repair documentation (e.g., SRM) step-by-step instructions are often used for quality checks, or the technician provides his own checklist, which another technician or inspector checks off. The ideal situation is for a second person to perform the inspection activities, be responsible for verifying the correctness of the process step, and signify on the work instructions that he has verified the process step. In some smaller GA maintenance organizations a second person may not be available, so the repair technician must approve his own work. In such cases, it is important that a quality checklist is made and used.

Due to the limitations of current postrepair inspection techniques to assess bondline integrity (e.g., check for bondlines with the required strength and durability), it is essential that repairs are performed per the approved repair documentation, and that the separate repair steps are checked as having been performed correctly. QC of the repair process is the basis of repair acceptance.

The following are examples of a typical OEM SRM instruction (verify prior to the actual repair cure cycle).

- The protective coating (e.g., conductive coating if present, paint enamel, and primer) must be removed per the approved instructions.

- The part may contain moisture and must be dried per the specific drying instructions.

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- The damage must be removed and the repair scarf or steps must be cut accurately.

- The repair materials (e.g., prepreg or wet lay-up resin, film and paste adhesives and dry preforms) must be collected and prepared per the approved instructions. The prepreg or dry preforms and adhesive film must be cut to the correct orientations and size to fit the repair steps or scarf.

- The adhesive ply and the prepreg or dry preforms must be laid down in the correct order and orientation. In the event of a wet lay-up repair the dry preform plies must each be wetted-out appropriately per the instructions.

- If a debulk cycle is called for, it must be performed using the required breather cloth, vacuum, and heat for the prescribed period of time.

- After all the repair plies have been laid down, the repair must be prepared for cure by assembling the correct bleeder, breather cloths, thermocouples, and heat application instructions.

A.3.2.2.4 Related Documents.

SAE References (Commercial Aircraft Composites Repair Committee – CACRC)
AIR5279: Composite and Bonded Structure Inspector: Training Document
ARP4977: Drying of Thermosetting Composite Materials

A.3.2.3 Describe the Critical Steps Necessary for Inspecting a Completed Bonded Repair, Including NDI and Interpretation of Results.

The postrepair inspection and interpretation of inspection results should be performed in accordance with the approved repair documentation and NDI manuals using approved inspection standards.

After the hot-bonding equipment, vacuum bag, breather plies, and parting films are removed, the cured repair is visually inspected for anomalies. A flashlight or intensive light source should be used to aid the visual inspection. The inspector will look for any discolored areas or indications, such as bubbles or bulges, that may indicate an anomaly within the repair patch or the repair bondline. Due to the translucent nature of the cured patches, a visual inspection of fiberglass repair patches can often reveal areas of disbond or delamination. Carbon repairs are less revealing of anomalies to visual inspection because of the color and lack of transparency of the carbon fibers.

Print out the cure parameters so they can be checked for the correct temperature and vacuum profiles. Vacuum or temperature profiles outside the limits for a correct cure cycle are indicators of a bad repair, and the repair may have to be removed and replaced with a correctly processed repair.

The repair will be NDI inspected using the tap hammer or P/E ultrasonic equipment. For laminate and sandwich parts with facesheets of more than two to three plies, it is recommended

that the NDI be performed using the P/E technique, if available. If any anomalies are detected, they must be compared to any allowable bonded repair defect limits that have been established for the specific component in the source documentation. If no defects are found, or if any detected anomalies (e.g., porosity or delaminations) are within acceptable limits, then the repair is considered acceptable. If any anomalies are detected, they must be compared to any allowable bonded repair defect limits that have been established for the specific component in the source documentation. A printed record of the repair's ultrasonic scan is helpful, if available. If available, the postrepair inspection information should be added to the information recorded during the cure cycle and retained for the component repair log.

If the repair is found defective (i.e., any detected anomalies or defects are judged to be outside acceptable limits, such as allowed processing anomaly limits), the repair must be removed and a new repair prepared and cured in its place.

If the repair is found satisfactory, any required protective coatings are to be restored to the repaired area (e.g., primer, enamel paint, and any conductive coating such as aluminum flame spray or aluminum mesh installed for lightning protection, or an isolation coating such as a glass fabric ply). The restored protection system should be checked for the correct (e.g., initial) conductivity.

If the repaired component is a flight control panel, the mass balance of the part must be checked against operational balance limits. If the repaired area is adjacent to other structure or movable parts, clearances should be checked.

A.3.2.3.1 Related Documents.

SAE References (Commercial Aircraft Composites Repair Committee—CACRC)
AMS2630:   Inspection, Ultrasonic Product Over 0.5 Inch (12.7 mm) Thick
AMS2632:   Inspection, Ultrasonic, of Thin Materials 0.50 Inch (12.7 mm) and Under in Cross-
                     Sectional Thickness
ARP5605:   Solid Composite Laminate NDI Reference Standards
ARP5606:   Composite Honeycomb NDI Reference Standards

A.3.2.4 Demonstrate and Perform Various Postrepair Acceptance Inspections, Including Visual Inspection, Tap Test, and/or Other Inspection Methods.

Students will participate in various NDI assessments of bonded repairs to laminate panels in a controlled laboratory environment. The NDI techniques to be used will be visual inspection, the tap test, and a P/E ultrasonic inspection method. The following laboratory equipment will be required:

- Previously repaired flat laminate panels
- Flashlights
- Tap coins
- P/E instrument and transducer
- Liquid-gel couplant

- Black, blue, and red marker pens

The instructor will first demonstrate how to effectively visually inspect the bonded repair.

A flashlight will be used to aid in the visual inspection.

The instructor will be looking for any discolored areas or indications, such as bubbles or bulges, that may indicate an anomaly within the repair patch or the repair bondline.

The instructor will know if the adhesive has wetted-out correctly by checking for adhesive bleed out all around the edge of the repair patch.

The instructor will then demonstrate the use of the tap test and the P/E equipment for inspecting the repaired area of the flat laminate panel.

The instructor will outline the extent of any potential defects in the repair after the use of each inspection technique.

The instructor will then show any differences in potential defect outlines from the visual inspection, tap coin, and P/E ultrasonic techniques.

It is anticipated that if defects are detected in the repair, the defect periphery indicated by the tap coin will be larger than what was indicated by the visual inspection, and the defect periphery indicated by P/E will be the largest.

When the instructor is satisfied that the students are familiar with the NDI equipment, each student will assess the state of the bonded repair using visual inspection, tap coin, and P/E ultrasonic equipment as follows:

- Visually inspect the surface of the repair on the laminate panel, using a flashlight to illuminate any suspicious areas of the bonded repair. Look for any discolored areas or indications, such as bulges or blisters, that may indicate anomalies in the repair patch or bondline.

- Mark any suspicious areas with the black marker pen.

- Inspect the surface of the repaired panel using the tap coin (or hammer) to inspect the entire repaired area of the laminate panel.

- Mark any potential anomaly or defect with the blue marker pen.

- Smear liquid gel over the entire repaired area of the laminate panel.

- Inspect the test panel repair surface using the P/E equipment and transducer, moving the transducer on the liquid gel.

- If the P/E instrument indicates any defect, clean away the gel and indicate the defect perimeter with dots using the red marker pen. When the defect area is adequately surrounded by red dots, clean off all the gel and link up the red dots with a continuous red line.

- Compare any defect perimeters indicated by the black, blue, and red markers.

Safety Message

Defects may be present in the bondline or within the repair patch due to poor surface preparation, material storage and handling, and cure process mistakes. It is essential to use the appropriate inspection methods for specific types of bonded repair (i.e., to sandwich, monolithic-stiffened, or metal components). Visual inspection is, in some cases, just as valuable as NDI methods for detecting flawed bonded repairs.

Misuse of equipment during postrepair inspections, or misinterpretation of postrepair inspection results may be detrimental to safety.

A.4 REPAIR PROCESSES.

A.4.1 Describe Composite Laminate Fabrication and Bonded Repair Methods.

A.4.1.1 Understand the Basics of Composite Laminate Fabrication.

Structural composite fabrication consists of fiber-reinforced composite components that have been fabricated from uncured material, e.g., epoxy prepreg tape and fabric materials, and formed while curing into the final configuration using heat and pressure. The heat and pressure are supplied by an autoclave, in an oven with the parts being bagged to provide vacuum pressure, or in a press that is heated. While that is a simple description of the process, the actual fabrication process is quite complicated with many parameters that can affect the quality of the composite component.

A.4.1.1.1 Comparison of Cure Processes.

OEMs often use different processes for curing epoxy prepreg laminates. Some prefer to use an oven instead of an autoclave for the composite cure cycle. Ovens only cure components using vacuum pressure. OEMs using ovens rather than autoclaves are somewhat limited in the scope of parts that they can produce because some materials and part configurations have complex shapes and contours that require the full pressure of an autoclave in addition to a vacuum. Autoclaves are much more expensive and complicated than ovens but are capable of curing more complex shapes as well as being better suited for cocuring assembled parts into subassemblies. Both processes still require the same fabrication steps with the only difference being the added consolidation pressure that an autoclave can provide.

A.4.1.1.2 Design and Processing for Low Cost.

Reliable, repeatable composite laminate fabrication starts with the design of the part, design and fabrication of the tooling, and ends with the assembly of the part to the main component or aircraft. Manufacturing processes and tooling are the elements that control the success and cost of a composite component and are integral parts of the design process.

The basic problem facing the OEM engineers is producing reliable composite hardware at an acceptable cost. Basic material costs are high compared to raw material forms of metal (e.g., aluminum sheet, plate, extrusions, and forgings). Labor-intensive steps, such as ply cutting and lay-up, have been mostly eliminated for large parts through automation. As an example, during the ply-by-ply, lay-down process, plies for the skins of major composite components are laid down on the female lay-up tool by automatic tape-laying machines, eliminating the human labor. However, after each ply has been laid down on the tool, it is inspected for orientation and area by an inspector, so the automatic lay-up process stops and starts each time.

A.4.1.1.3 Emerging Developments in Composite Processing.

Manufacturers continually search for technologies that reduce cost and labor. Designers often create composite parts configured in difficult-to-manufacture shapes, with performance-enhancing features, such as through-thickness reinforcements. Some of the newer composite-manufacturing technologies are discussed in sections A.4.1.1.4 through A.4.1.1.8.

A.4.1.1.4 Tow Placement.

Tow placement is a method for laying down epoxy prepreg material. With tow placement, the material is stored on preimpregnated spools of string-like composite tows. (A tow is comprised of a string of 3000-6000 individual fibers.) Multiple spools are held on a creel that aligns the tows in parallel rows that are fed into the tow-placement machine head and then lays the material down on the cure tool or mandrel. The advantages of tow placement are (1) the creels can contain tow material with different fibers so hybrid (e.g., carbon and glass) laminates can be created automatically, (2) the material can be laid down on tools to create complex curved parts, unlike tape-laying machines, which only lay down material to create flat or gently curved parts, and (3) material lay-up rates can far exceed those of manual lay-up.

A.4.1.1.5 Thermoplastic Forming.

This process uses thermoplastic materials that have no cross-linking during cure, unlike epoxy thermoset materials. This allows reforming if a part has been cured with anomalies. Thermoplastic materials are more difficult to lay-up and cure, because uncured, they are often stiff and boardy and because they require higher cure temperatures (650° to 750°F) than epoxy thermosets. The advantage for parts fabricated from thermoplastics is they tend to be more resistant to environmental effects when cured. The prepregs can be purchased as precured, individual ply sheets that can be formed into various shapes in a heated press.

A.4.1.1.6 Resin Transfer Molding.

Resin transfer molding (RTM) is a resin injection process in which fiber preforms are impregnated with resin inside a rigid, closed mold. Resin is injected at high pressure, and the mold is heated via imbedded heaters without needing an oven or autoclave. Due to extremely high tooling costs and lead times, the RTM process is cost-effective for producing large quantities of parts or for producing parts that have complex geometry. One issue that must be considered with RTM fiber preforms is the ability to repair off-axis fibers. Because the preforms will often create plies with curved fibers and fibers oriented at odd angles, the designer must accommodate these nonuniformities in the repair design.

A.4.1.1.7 Vacuum-Assisted Resin Transfer Molding.

Vacuum-assisted resin transfer molding is a liquid-forming process similar to the RTM process, but it uses a vacuum to assist in moving the resin through the heated mold.

A.4.1.1.8 Nonautoclave Curing of Composites.

Processes employing only vacuum pressure or no pressure are often used for aerospace applications. The Seemann Composite Resin Infusion Molding Process (SCRIMP) infuses resin into a dry preform under a vacuum bag on a hard outside mold line-type mold. Heating is accomplished in either an oven or an unpressurized autoclave. Some SCRIMP use room temperature curing resins where no heating is required. This process is especially useful for curing large parts with complex shapes (e.g., boat hulls and armored vehicle structure). Another nonautoclave process widely used is simple oven curing of prepregs under vacuum. Oven curing of prepreg or wet lay-up parts using only vacuum pressure is especially popular among GA OEMs because it uses lower-cost tooling and less capital intensive equipment.

A.4.1.2 Understand the Basics of Composite Bonded Repair.

Similar to the original part fabrication process, a simple view of composite bonded repair is that damaged composite components are repaired with either prepreg or wet lay-up materials when the repair patches are cured onto the damaged component using heat and pressure. The purpose of bonded repairs to structural composite components is either to restore structural capability or seal a damaged area from environmental effects.

During the original part fabrication, the OEM has the responsibility of producing reliable repeatable composite components that have the required performance. The repair shop has the responsibility of performing reliable, repeatable repairs that restore structural performance, fit, and form, while minimizing operator revenue losses. The repair shop must reliably restore aircraft safety and perform the repair with minimum cost and service interruption without the sophisticated tooling and fabrication equipment of the OEM. Many bonded repairs must be performed on the aircraft; therefore, the bonding process must be performed without autoclave pressure. Damaged components that can be easily removed from the aircraft can be repaired using an autoclave, providing there are no size restrictions.

Bonded repair patches may reduce load-carrying capability because nonautoclave processing of patches does not enable the ply consolidation that autoclave processing provides. Porosity is typically greater in nonautoclaved repairs, so material allowables are reduced. When designing a composite repair, care should be taken to consider the effects of patch interfaces and nonautoclave allowables.

Patch ply thickness may increase in nonautoclave processing. This is due to the reduced consolidation pressure during cure when the ply thickness can increase by an additional 0.002 inch. Repair designers should consider whether or not the additional thickness will be detrimental to aerodynamic or aesthetic concerns.

A.4.1.2.1 Repair Molds.

In some cases, a specialized mold may be required to complete a repair to a composite component. This usually occurs when a repair is especially large (as in the reskinning of an elevator) or when off-component patch debulking or curing is required (as is the case with high-outgassing patch materials). In these cases, the original mold or a custom-fabricated mold is used to provide a caul for the bonded patch. Custom-fabricated molds can be created by lifting an impression from the same area on another aircraft or similar area of the same aircraft, machining a custom mold, or even producing a mold using stereo lithography.

A.4.1.2.2 Material Selection for Bonded Repairs.

Materials are selected to create a repair that is both sufficiently strong and practical to process. The patch can be configured with either materials that match the parent component or nonparent materials. Since parent material properties usually cannot be attained, due to reduced consolidation, fiber volume mismatch, and thermal anomalies during typical vacuum cures, nonparent materials may be more appropriate for creating a nonautoclave bonded patch. Factors that must be considered when selecting a material include (1) materials that are available to the repair site, (2) materials that have allowables generated when replacing the parent material in repair, (3) the level of stiffness required for the repair, and (4) thermal nonuniformities of the parent part that will require a more robust processing material. In all cases, only the material called out in the repair documentation should be used for the composite repair.

A.4.1.2.3 Scarfing.

Scarf, or stepped joints are normally required for flush bonded repairs, as shown in figure A-24. The scarf requires the removal of the damaged material so the repair material plies can be laid down in a stepped process. Scarf angles of 30-50/1 are normal for bonded repairs to composite laminates. The part strength is maintained when adequate surface area is available for each repair ply to overlap its underlying parent ply so the load can transfer between the two. Scarfs should be as smooth as possible (elimination of waviness) to enable even pressure distribution during debulk and consolidation. Scarfing angle allowables must be maintained per the repair documentation to ensure proper load transfer and overall repair strength.
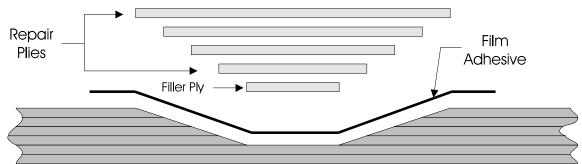
Scarf Removal

Figure A-24. Scarf Setup
(Courtesy of Abaris Training Resources Incorporated)

A.4.1.2.4 Prepreg Repair.

These bonded repairs use either the original prepreg material or a substitute prepreg material. Prepreg repairs can be performed using an autoclave if the damaged part can be easily removed from the aircraft. In this case, providing the repair bonding procedure is the same as the original part procedure, the repair plies will have the same properties as the plies removed or damaged. Prepreg repairs can also be performed using vacuum pressure and heat blankets. Repair plies using this process usually will not have the same properties as the original autoclaved prepreg plies. Prepreg materials, during original part fabrication, are subjected to autoclave pressure between 35 and 90 psig, where volatiles are vented and porosity is low. Vacuum bag prepreg repairs may contain additional to normal porosity that can reduce compression strength and stiffness, when compared to the original autoclave-cured part. This may not be the case when performing vacuum cure repairs to composite parts that were originally cured in an oven where no pressure other than that provided by vacuum is applied. The main difference between vacuum cure repairs and original parts cured in an oven is probably the control of the heating source. Heat provided in an oven is potentially more consistent throughout the laminate compared to the heat provided by a hot bonder and heat blanket. Prepreg repairs, both vacuum bag and autoclave, processed on commercial transport structural composite components are typically processed at temperatures ranging from 250° to 350°F. Prepreg repairs can use either tape or fabric forms, which unfortunately have a limited shelf life. They must be kept in a freezer in specified conditions, and after their expirations dates, they must be either scrapped or retested for acceptance.

In the field, special techniques often must be employed to ensure adequate repair. One technique is shown in figure A-25, when there is no access to the component's back side. The patch is held in place with a wire or clamp as the repair is cured.
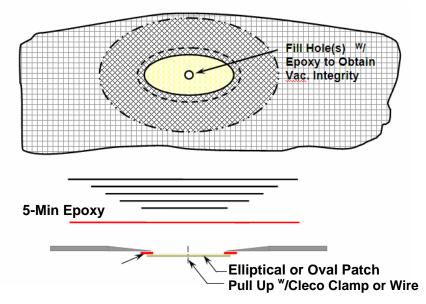
Figure A-25. Repair of Composite With no Back Side Access
(Courtesy of Abaris Training Resources Incorporated)

A.4.1.2.5 Wet Lay-Up Repairs.

These bonded repairs use special two-part epoxy resins and dry-fiber fabrics. The resin ingredients are kept sealed in separate containers at room temperature. These resin ingredients have long shelf lives, as long as the containers remain sealed. The dry fabrics, woven similar to fabric prepreg materials, are also kept at room temperature. The mats have no expiration date as long as they are kept in reasonably dry conditions and free of contamination. Each wet lay-up repair ply consists of a fiber mat cut to the specified size and laid down in a specific direction, and impregnated with the correct amount of mixed resin. This process is repeated for all subsequent plies. The lay down process is slower and potentially more variable than for prepreg, but the wet lay-up repair materials have more convenient and less restrictive storage requirements. Additionally, the fiber and resin volume of a wet lay-up patch is less repeatable since the wet lay-up process is so technician dependent. With the less repeatable fiber resin content, the design of the wet lay-up patch must compensate for nonoptimal resin content.

A.4.1.2.6 Repair Classifications.

There are often three classifications of repair in approved repair documentation. As an example, the following repair classifications are typical for OEM SRMs:

- Permanent repairs: Permanent repairs are repairs that restore either original part capability or sufficient capability to carry the required loads. These are considered terminating repairs, meaning that, under normal circumstances, no further action is required. Prepreg repairs processed at 250° and 350°F are usually permanent repairs. Wet lay-up repairs processed at 180° to 200°F are also often classified as permanent.

- Temporary repairs: Temporary repairs are either performed to seal damages detected on the ramp, or are repairs that are considered not to have sufficient durability to remain on the aircraft for an indefinite length of time. Temporary repairs are usually only allowed for one or a few flights. Many temporary repairs use wet lay-up materials processed at room temperature up to 150°F.

- Time-limited repairs: Time-limited repairs are repairs that may restore sufficient part capability, but may not have fully proved durability. These repairs may be allowed to remain on the aircraft for many flights, but have an inspection schedule with a final remove and replace with a permanent repair requirement. Many time-limited repairs use wet lay-up materials processed at 150° to 200°F.

A.4.1.2.7 Repair Categories.

There are two categories of repair:

- Depot level (military) or maintenance base (commercial): These repairs are performed at a major maintenance base in a controlled environment. Repairs in this environment can range from simple on-aircraft wet lay-up vacuum bag repairs to larger, more complex autoclave prepreg repairs to critical areas of removable components. In either case, repairs performed at the depot or maintenance base are typically permanent repairs.

- Field level (military) or line station (commercial): These repairs are performed at a forward-operating base with limited facilities. Repairs in the field are usually limited to small wet lay-up vacuum bag bonded repairs and bolted overlay repairs. Due to the limited facilities and time, most field or line repairs are temporary or time-limited.

A.4.1.3 Describe the Detailed Processing Steps Necessary for Laminate Fabrication (Factory), Bonded Repair (Maintenance Base or Line Station), and Disposition by the Original Manufacturer's Material Review Board (MRB).

A.4.1.3.1 Laminate Fabrication.

To more completely demonstrate the steps in the bonding process of composite laminates in the factory, the following is presented as an example of an autoclave-cured skin panel bond assembly at a major OEM:

- Composite material is received and accepted based on tests meeting the specifications.

- A lay-up manufacturing plan is prepared per the part drawing.

- The female lay-up and cure tool is cleaned and its surface is prepared with a release agent so the first ply laid down does not stick to the tool during cure.

- Material is removed from the freezer and prepared for use.

- Each ply is laid down on the lay-up tool by a tape-laying machine programmed to the manufacturing plan, each ply is manually inspected for orientation, periphery and correct order per the manufacturing plan. Note: In some cases, as the experience and confidence bases increase, the ply-by-ply inspection, after lay down and cutting, may be changed to a ply-sampling plan. This will significantly decrease the manufacturing time cycle for each part.

- The laminate is vacuum bagged and debulked to eliminate air that is trapped between plies during lay down and to prevent potential wrinkling. In many cases, the debulk cycle is preformed every six plies or so, in others, after every ply. The double vacuum debulk (DVD) cycle is sometimes used to removed entrapped air and potential wrinkles from the total laminate in one operation.

- The debulk bag is removed.

- Precured stringers are located in positions on the debulked skin with film adhesive. Note: The precured stringers have been previously laid up and cured in a separate process. Other OEMs may precure the skin and then bond the uncured stringers to the skin in another cure cycle. Either way, these processes are termed cobonding. The classic method of fabricating a stiffened skin panel is to cure both the skin and stringers together, and is termed cocuring. In many cases, when assembling uncured stringers to relatively thick parts, during the cocure process, the stringers sink into the skin due the presence of heavy stringer tooling bearing down on the skin during the cure cycle. That is one of the reasons that some OEMs have turned to the cobonding process. In this process the stringers do not need heavy tooling (or mandrels) to ensure the correct shape for the stringers. There is a critical issue with the cobonding process. That is the issue of surface preparation of the precured part. To provide for a good bondline, the precured part mating surface will need to be prepared so that (1) it does not have a smooth surface, and (2) it may need to be cleaned to ensure all potential contaminates are removed. Some OEMs cure a special peel ply to the precured part mating surface during its cure cycle. This peel ply is removed immediately prior to the assembly of the precured part to the uncured part, and helps to provide for a slightly roughened (i.e., the imprint of the peel-ply fabric) and uncontaminated surface for bonding.

- The whole skin assembly is vacuum bagged for cure with appropriate stringer tooling, breather and bleeder plies, edge dams, and bag sealant.

- The vacuum bagged part is installed in the autoclave and the cure process is programmed.

- The cure is started and in-process data, such as temperature, autoclave pressure, and vacuum, are monitored and recorded on strip charts.

- The recorded process data record is inspected periodically for conformity to the cure plan.

- The cure cycle ends and the part is allowed to cool down.

- The vacuum bag, bleeder, and breather plies are removed, and the part is visually inspected for defects.

- The part is subjected to TTU inspection and the results of the scan are recorded.

- Inspectors examine the printed scan results and record any detected anomalies for MRB action.

- The part is edge trimmed to the final assembly periphery by water jet cutters and the correct geometry is verified by an inspector for accuracy.

There are many steps in the bond assembly process, and many places for errors. It is essential that errors be kept to a minimum and any anomalies that do result are detected. The automatic equipment helps to reduce human errors. The in-process QCs and inspections and postfabrication inspections are provided to ensure that errors are detected.

After the above process, the part is primed, assembled to the main aircraft component, a surface protection system is applied (aluminum flame spray or aluminum mesh), and it is enamel painted.
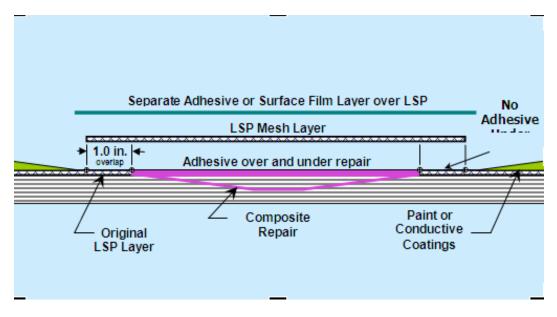
A.4.1.3.2 Bonded Repair.

A typical on-aircraft, hot-bonded repair includes a number of the basic steps that are similar to the original component fabrication process (material preparation, lay-up, vacuum bagging, cure, and inspection). Many of the individual bonded repair steps can occur in parallel when performing a time-critical repair. Repair material control and handling are also similar to the original material control at the OEM. One difference may be that some repair shops do not perform material acceptance tests. These shops typically only use small quantities of material on an infrequent basis; therefore, they often procure materials from third-party suppliers who perform all receiving inspection. A typical bonded repair is shown in figure A-26.

Figure A-26. Tapered Scarf Repair
(Courtesy of Abaris Training Resources Incorporated)

The actual steps necessary to perform a permanent hot-bonded repair for a vacuum bag cure repair of the outside facesheet and core of a sandwich component using prepreg material are as follows:

- Perform the damage assessment and cleanup as outlined in section A.2.1.1.

- Clean the abraded surface and tapered area with a soft cloth moistened with an approved solvent. See section 5.4.2 for details.

- Prepare and clean a core plug the same size as the cutout in the facesheet.

- Install a core plug with foaming adhesive and cure using a vacuum bag, thermocouples, and heat blanket. After cure, sand it flush with the first ply of the facesheet.

- Place the film adhesive down first, then each replacement repair ply, and finish with additional ply (or plies) over the tapered repair. Sweep each ply to remove any wrinkles

- If required by SRM or approved repair documentation, compact repair plies using a temporary vacuum bag.

- Place a parting film over the entire repair and install a vacuum system, a minimum of three thermocouples, the required surface bleeder cloths, and a heat blanket. Apply the vacuum seal around the repair.

- Apply a minimum vacuum of 22 inches of mercury, and cure the repair per the specified cure cycle. Ensure that the vacuum does not fall below the minimum of 22 inches of mercury. Monitor the heat-up and cool-down rates, dwell temperature, and vacuum pressure throughout the cure cycle.

- After the repair has been cooled down to a prescribed temperature, remove the vacuum pressure.

- After cooldown is complete, remove the heat blanket, breather cloths, thermocouples, vacuum seal and bag, and parting film.

- Inspect the repair for voids and anomalies using approved inspection methods. Make sure to inspect the area around the bonded repair, up to 6 inches away from the edge of the repair.

- If the inspection proves the repair to be satisfactory, restore the protective coatings over the repaired area, per the approved documentation, and return the component to service. Some composite components are protected from excessive damage from lightning strikes by systems such as flame spray coatings, aluminum mesh, or picture framing. If damaged or removed during a repair, these protection systems need to be restored per the SRM or other approved documentation. Some composite components that contact aluminum parts are isolated by corrosion-prevention systems such as a layer of glass epoxy. If these isolation systems are damaged or removed during the repair, they must be

restored per approved documentation. Below is an example of lightening strike repair (figure A-27).

Figure A-27. Lightning Strike Repair
(Courtesy of Abaris Training Resources Incorporated)

A quality permanent hot-bonded repair to a sandwich part currently takes at least 14 hours from damage discovery to paint restoration. If damage is found in a solid laminate, the OEM SRM procedure may require that the technician completely dry the damaged area, which requires 24 hours; therefore, completeing a permanent repair under these circumstances can take over 30 hours. Temporary and time-limited bonded repairs may take less time, depending upon the cure temperature; however, any moisture present should be eliminated.

A.4.1.3.3 Material Review Board

Most major OEMs have a factory process called the Material Review Board. It is a process that is intended to make dispositions concerning reported defects or unsatisfactory raw material and take corrective actions, as necessary. Dispositions may include repair or rework, scrapping the part or material, or using the part or material as is. Corrective action may be taken to reduce the number of repetitive errors or defects in the fabrication process.

The material review process starts with an inspection report (rejection notice), which is provided by an inspector. The rejection notice is reviewed and entered into the Quality Management System (QMS) database. The rejected material or part is reviewed to determine its disposition. It may be used as is, scrapped, reworked, or, if raw material, returned to the vendor.

The purpose of the corrective action process is to prevent or reduce the rate of recurrence of defects, discrepancies, failures, or other conditions judged significantly detrimental to safety, cost, quality, or performance of the product by correcting the root cause. The process is initiated

by management request, document review, QMS thresholds, or analysis reports. Problems are investigated to determine the root cause and appropriate corrective action.

Problems that cannot be resolved at a lower level are resolved by Corrective Action Committees. These committees are composed of midlevel management personnel that address

- problems that cross departmental lines.

- problems unresolved by departmental corrective action activities, root causes of repetitive problems.

- high-dollar and high-frequency problems.

Follow-up is assigned to ensure corrective action is effective.

A.4.1.4 Describe Key Characteristics and Processing Parameters for Laminate and Sandwich Panel Fabrication.

Good repeatable composite laminate fabrication depends on the following key characteristics and parameters.

- Fresh prepreg material is essential for consistent, high-quality cured laminates. Prepreg materials have a limited freezer storage life (often less than 6 months), as well as limited allowable out-time (often less than 10 days), once the material has been removed from the freezer, but before it has been cured. If the material exceeds its storage life or its out-time, the matrix intermolecular cross-linking may not fully occur during cure, yielding a cured component that will not meet strength requirements. Porosity and delaminations are often the result of using old material. Old material may loose its tackiness, feel boardy to the touch, and be difficult to manipulate during lay-up and forming operations. The material must also be kept free of debris and fluids during processing to prevent strength degradation through contamination.

- Cure tools must be designed correctly and fabricated from materials with matching coefficients of thermal expansion with the composite material to produce laminates that will not warp during the cure cycle. Mandrels for drape-forming stiffeners, spars, and ribs must be designed with slightly open angles so they can be easily removed after cure. They also must be designed so the material can be draped around them easily.

- Most OEMs use some method of automatic ply lay down to speed up the process and eliminate human errors. Tape material can easily be laid down by machines, whereas fabric is more difficult and, in a majority of cases, is laid down by hand. An important aspect of ply lay-up is inspection by a qualified inspector to ensure that the plies have been laid down in the correct orientation and order.

- Many composite materials may require debulking every five or six plies to remove any air that is trapped between plies during lay down. Air trapped within a laminate can

cause wrinkles and porosity during cure. The debulking procedure is typically performed under vacuum and, for some materials, may require heating. The DVD method was developed to obviate the need for a debulk cycle every six plies. DVD has successfully removed entrapped air from laminates with more than 40 plies prior to the cure cycle.

- OEMs developed a reusable vacuum-bagging system that removes trapped air and escaping volatiles (through venting) and encourages consistent resin flow throughout the laminate. The bag must be vacuum tight to ensure proper ply consolidation during cure. Typical autoclave pressures range from 36-45 psig for sandwich parts to more than 75 psig for laminate-stiffened parts to ensure a well-consolidated part without porosity.

- During cure, a monitoring system is required to review all processing parameters throughout the cure cycle to ensure a good-quality part that meets the desired structural performance and environmental durability. Thermocouples, placed at critical locations within the vacuum bag, are monitored to ensure correct cure temperature. The autoclave pressure and vacuum are also monitored to ensure there are no critical variations in the cure cycle that may lead to anomalies in the cured laminate.

- Inspection of the cured part is an essential part of the fabrication process. Both visual and NDI methods of inspection are used by the OEM to ensure that any defects in the cured parts are detected. A cured part is visually inspected after cooldown, after which it is subjected to a rigorous ultrasonic inspection. Many OEMs use TTU ultrasonic equipment in the factory for postfabrication inspection. The signal-transmitting medium is achieved with water jets for large components or immersion in a tank of water for smaller parts. Very large, thick parts can be inspected rapidly with TTU, and defects as small as 0.5-inch diameter can be discovered in flat or gently curved parts. TTU equipment has limitations for inspecting parts with tight bend radii (e.g., corners on T- and I-section components). In these cases, P/E ultrasonic equipment with a small hand-held head for the transducer is used to detect anomalies within these bend radii.

A.4.1.5 Identify Typical Processing Defects That Occur in Composite Laminate Fabrication and Bonded Repair.

Processing anomalies, such as voids, delaminations, and porosity, typically occur during the cure process and may be the result of poor tooling, insufficient ply consolidation, low autoclave pressure, loss of vacuum during the cure cycle, and postcure process damages, such as edge damage, dents, delaminations, and poorly drilled holes.

Lay-up mandrels must be designed to produce components that match the designed part contour. If the tooling produces components that exceed contour allowables, then the components will require excessive preloading during assembly, which will compromise the component's ability to carry required loads. Likewise, mandrels for cocuring (e.g., stringer or stiffener mandrels) must be designed and installed with minimal mismatches to ensure proper consolidation of the cocured component.

Some of the latest, tougher materials require ply consolidation prior to the cure cycle. This is due, in part, to the matrix viscosity, and depending on the number of plies in a lay-up, several compaction cycles may be necessary. In these cases, insufficient ply consolidation or compaction prior to cure can lead to voids, porosity, and delaminations.

During the cure cycle, any loss of vacuum, autoclave pressure, or temperature can result in anomalies such as voids and porosity. An improperly cured part may also have lower than required thermal stability in addition to lower mechanical properties.

Inclusions can occur when insufficient care is taken during ply lay-up. Foreign objects, such as backing paper, release film, and even personal items, have been discovered during postcure inspection. Inclusions are physical defects that reduce the load-carrying capacity of a part. Inclusions are often attributed to a lack of diligence by the lay-up technicians or to the lack of attention by the in-process inspectors. Postfabrication NDI methods employed by the OEMs will discover the majority of inclusion anomalies.

Defects, such as edge damages, dents, delaminations, and fastener hole damage, can result after cure, during part handling, machining, and assembly. Composite parts are more vulnerable to impacts than metal parts, and care must be taken during handling. Incorrect machining and drilling parameters, such as feed rates and drill speeds, can lead to delaminations in and around holes and part edges.

The following repair process mistakes are almost certain to lead to repairs that may have less than required structural properties or may become disbonded in service.

- It is essential that the protective coating (e.g., conductive coating if present, paint enamel, and primer) is removed using a prescribed method, such as abrading or sanding. The coatings should be completely removed over an area that more than encompasses the repair. (Note: Extreme care must be taken to avoid damaging plies when removing paint and primer.)

- All fluids must be removed from the component using vacuum and heat. Failure to remove all moisture and fluids from the repair region of the component may cause a patch bondline failure. Residual moisture or fluids can seep into the patch bondline during the cure cycle, destroying the chemistry of the resin matrix, or the moisture can vaporize under the elevated cure temperatures and blow the patch off the component.

- If the abraded surface and tapered area are not cleaned sufficiently with an approved solvent, contaminates (e.g., dust) may be present in the repair bondline.

- Prepregs and adhesives that have exceeded their allowable out-time or have expired must not be used.

- It is essential that the right amount of the two-part resin be mixed correctly, otherwise the repair may not reach the desired state of cure.

- During ply lay-up, proper debulking is critical to the finished patch quality. Many materials require heated debulks to remove volatiles prior to cure. Some materials have a limit to the number of plies that can be cured to avoid entrapping volatiles in a thick lay-up. Some materials can be debulked using the DVD process, where volatiles are drawn from the patch in a heated cycle under vacuum without compaction. If a patch is not properly debulked to remove volatiles prior to cure, excessive porosity may be present in the cured patch. The DVD debulk method is widely used by the U.S. Navy on repairs to F/A-18 and AV8B aircraft composite parts.

- The materials used in a patch lay-up must be kept free of moisture. Moisture in the materials can cause high porosity or patch delamination as the water on the materials is transformed to steam during cure. As such, lay-up materials should remain covered whenever possible to avoid moisture ingression from environmental humidity. Also, prior to unwrapping frozen materials, the material must be removed from the freezer and allowed to reach room temperature in the sealed bag to avoid accumulation of condensation on the exposed prepregs. All frozen prepregs must be sealed in airtight packaging to avoid condensation accumulation in the freezer environment.

- If there are any leaks in the vacuum bag, they can be detected by monitoring the heating unit readouts, otherwise the repaired patch and bondline may contain voids, excessive porosity, and delaminations.

- One critical issue that must be pointed out is the effect of heat from the repair on substructure or underlying components. If a full-temperature repair adversely effects adjacent structure or sealants, then alternative repair processes must be considered. Also, if underlying equipment will be adversely affected by adjacent heating, the equipment must be either removed prior to the repair or protected from excessive heat. Always remember to reinstall any removed equipment.

- If the cure process is not monitored closely, wide variations in thermocouple readings, incorrect heat-up rates, cure temperatures, and loss of vacuum may result. Cure temperature of the bonding process must be properly controlled. If there is a lower or higher than acceptable thermocouple reading, then an undercured or overcured repair may result. When cure temperatures vary outside the specified limits, the patch properties may be diminished. Substructure heat sinks can cause the cure temperatures to vary by drawing heat away from the repair zone. For this reason, it is important to be cognizant of the substructure when placing thermocouples. Vacuum pressure during the bonding process must also be controlled, or a repair may have insufficient properties. Air paths must be maintained through breather layers in the vacuum bag, otherwise air may be entrapped.

Safety Message

The correct processing of composite components is critical to the elimination of processing defects. This includes all processing steps, including surface preparation, material handling and storage, material lay-up, part bagging and cure, and postprocess inspection. Inspection of

bonded composite and metal bond assemblies using appropriate NDI techniques together with in-process QC is essential to have defect-free composite components. Equally, adherence to the correct repair processing steps, in-process QCs, and postprocess inspection is crucial to successful composite repairs. In addition to the above steps for the original part fabrication, composite repair procedures will require damage removal and acceptable surface preparation for bonding. Note that these procedures may also include drying. The particular fabrication and repair methods for a given structural part will often include specific processing details that are unique to that part.

In the event of processing mistakes, misuse of equipment, or misinterpretation of in-process QCs and postprocess NDI results, composite part fabrication or repairs may be compromised. Parts delivered with unaddressed defects and compromised repairs may not be airworthy.

A.4.1.6 Describe Differences Between Wet Lay-Up and Prepreg Bonded Repairs to Sandwich and Laminate Parts.

Wet lay-up repairs employ different types of materials than prepreg repairs. Wet lay-up materials consist of fabric fiber mats that are stored at room temperature and two-part resin systems that are stored in sealed cans and mixed to prescribed compounds.

Prepreg materials must be stored in freezers, have limited storage lives, and require temperature-monitoring equipment, whereas wet lay-up materials can be stored at room temperature, have longer shelf lives, and do not require temperature monitoring.

Wet lay-up materials have a short working time (often less than an hour) when the impregnated plies must be applied to the repair. Prepregs typically have long working times (often over 100 hours). Therefore, repair technicians can often repair larger, deeper damages with prepregs. In general, wet lay-up bonded repairs are useful for sandwich parts, thin laminate areas (e.g., edge bands of sandwich parts), and minor (partial thickness) repairs to laminate parts.

Repair technicians using prepregs can achieve a repair with the same process (autoclave and higher temperature) used by the OEM for the original part fabrication, assuming the part requiring repair can be removed from the aircraft.

Prepreg materials are easier to use than wet lay-up materials, requiring no precise mixing of the resin parts.

Use of prepreg materials for repairs performed on the aircraft may be limited if autoclave pressures are needed.

A.4.2 PERFORM A SIMPLE BONDED COMPOSITE REPAIR.

Figures A-28 through A-43 depict various stages of bonded repair. (All photographs are courtesy of Abaris.)

Figure A-28. Damage Removal and Tapered-Scarf Preparation on Glass/Epoxy
Foam Sandwich Panel

Figure A-29. Performing Bonded Repair

Figure A-30. Lay Out Repair Plies and Mark Warp Fiber Direction on Warp Face Side of
Poly-Sandwiched Fabric

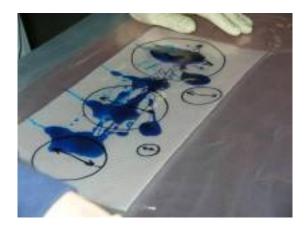Figure A-31. Fill Cavity With Rapid-Cure Epoxy Syntactic Filler

Figure A-32. Grind Cured Filler to Flush Surface

Figure A-33. Apply Repair Resin to Glass Fabric

Figure A-34. Distribute Resin Throughout Fabric With a Squeegee; Brush on an Interface Coat
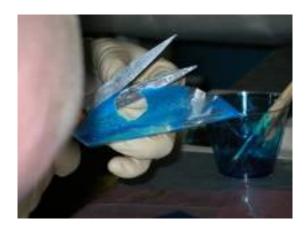of Resin in the Tapered Scarf Area Plus 1/2 Inch all Around
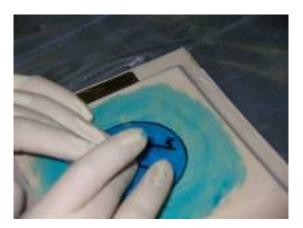

Figure A-35. Cut Repair Plies to Size

Figure A-36. Lay-Up Repair Plies to Match Original Orientation (-45 warp face up)
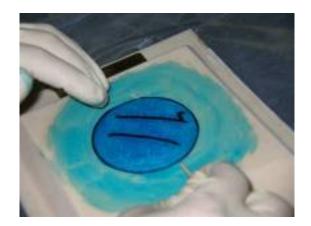
Figure A-37. Lay-Up Repair Plies to Match Original Orientation (+45 warp face up)
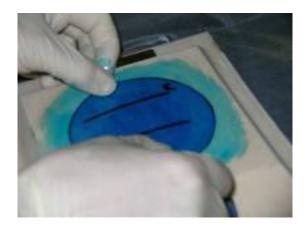

Figure A-38. Remove Backing Film Between Plies

Figure A-39. Lay-Up Repair Plies to Match Original Orientation (90 warp face up)
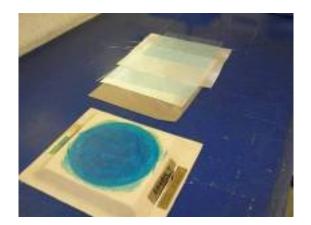
Figure A-40. Lay-Up Repair Plies to Match Original Orientation (0 warp face up)

Figure A-41. Assemble Bleeder Stack; Peel Ply, Direct Bleeder (#1581 Glass), P-3 Perforated Film, Secondary Bleeder, and Solid Film Separator Layer

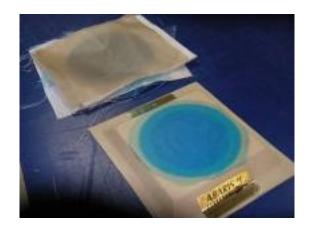Figure A-42. Vacuum Bag Repair and Check for Leaks

Figure A-43. Examine Bleeder for Uniform Bleed Pattern

A.4.2.1 Demonstrate and Apply Common Drying and Surface Preparation Techniques, Including how to Inspect for Acceptability.

Proper drying and surface preparation are essential for a successful bonded composite repair. This is due to the sensitivity of adhesives and resins to accumulated component moisture and contamination prior to and during cure. The drying times called out in the source repair documentation must be followed explicitly to ensure that the component is sufficiently dry for bonding operations. The duration of drying times recommended for composites fabricated from epoxy resins can be excessively long, so some operators may use approved revised drying times that were authorized by DERs to truncate these for expediency.

A.4.2.1.1 Drying of Composites Demonstration.

Contaminants and water must be removed from the component using vacuum and heat. The area being dried must extend well beyond the area of damage per the instructions in the approved repair documentation.   The following steps are based on a major OEM SRM drying requirements:

- Remove the protective coating (e.g., conductive coating if present, paint enamel, and primer) using the prescribed method, such as abrading or sanding. Remove the coatings over an area that will more than encompass the repair per the approved repair documentation. Be careful not to damage any composite fiber material. Use protective eyeware and a face mask, and capture dust with a vacuum system.

- If the damaged part is of sandwich construction, open the honeycomb cells and sand off any adhesive covering the cells.

- Lay down a glass fabric screen over the entire damaged area.

- Place a thermocouple in the center of the damage.

- Place a breather cloth over the entire area, and use masking tape to hold the cloth in position.

- Place the heat blanket in position over the breather cloth, making sure that the vacuum probe is above the breather cloth.

- Place the vacuum bag over the entire area and seal in place.

- Apply a minimum vacuum of 22 inches of Mercury to the vacuum bag.

- Operate the heat blanket to apply heat to the area being dried at the heat-up, hold, and cooldown rates and temperatures specified in the approved repair documentation.

- Remove the vacuum bag, heat blanket, breather cloth, and screen.

- Moisture and contaminant removal can take up to 24 hours, depending on whether the damaged part is a sandwich or solid laminate construction. Since practical methods to verify moisture removal are not yet available, the drying process must be followed stringently. After drying the repair area, ensure that the area is protected from further moisture exposure by covering the area with waterproof film and adhesive tape or sealant.

A.4.2.1.2 Repair Scarfing and Cleaning.

In all processes, the approved repair documentation must be followed explicitly. The following steps are condensed from instructions in a typical OEM SRM.

- Remove any damaged core, if present, to the same size as the cutout in the facesheet. The core should be excavated down to the far side facesheet or adhesive. Pneumatic tools with high-speed router bits can aid greatly in this process. Caution should be exercised to avoid damaging the back side facesheet.

- Make the necessary taper to the damaged plies for the bonded scarfed repair, using a taper ratio per the approved repair documentation (e.g., 50 to 1). Use protective eyeware and a facemask, and remove dust with a vacuum system. It is advisable to have another technician or inspector perform a check on the work done to this point.

- Remove any remaining dust and debris from the repair area with a vacuum cleaner.

- Wipe the scarfed surface and surrounding surfaces with solvent per the approved repair documentation. Care must be taken to avoid smearing contamination from unclean adjacent surfaces onto the repair area. Use clean dry cloths to remove the solvent until the cloth wipe is completely dry.

- All composite repairs must be conducted per the work instructions, which include steps that document successful inspection at the completion all significant processes.

A.4.2.1.3 Related Documents.

SAE References (Commercial Aircraft Composites Repair Committee – CACRC)
ARP4977:   Drying of Thermosetting Composite Materials
ARP4916:   Masking and Cleaning of Epoxy and Polyester Matrix Thermosetting Composite
            Materials
ARP4991:   Core Restoration of Thermosetting Composite Components

A.4.2.2 Demonstrate and Apply Material Lay-Up and Compaction Processes for a Simple
Laminate Panel Repair.

A.4.2.2.1 Prepreg Ply Lay-Up for Scarfed Laminate.

The instructions in the approved repair documentation must be followed explicitly.   The
following steps are condensed from instructions in a typical OEM SRM.

- Trace scarf contours onto ply templates. Ply count and orientation must be in accordance
  with the approved repair documentation. Mark the orientation, ply number, and
  crosshairs on each template to facilitate lay-up. Use these to cut out the adhesive film
  and repair plies.

- Cut adhesive and repair plies using ply templates. Make sure that each replacement ply
  fits and the orientation is in accordance with the approved repair documentation.

- Lay adhesive and repair plies into the scarf per the approved repair documentation. In
  most prepreg repairs, film adhesive will be placed between the scarfed surface and the
  repair plies. Some approved repair documentation will require an additional ply to cover
  the entire repair. Sweep each ply after lay-up to remove any wrinkles and air trapped
  between the repair plies. All template and backing films must be removed during ply
  lay-up.

- Compact the repair plies as required by the approved repair documentation, using a
  temporary vacuum bag to remove any air trapped between the repair plies. Some
  prepreg materials may need a compaction cycle for each repair ply laid down, while
  others may allow a greater number of plies to be applied between compaction cycles.

A.4.2.2.2 Wet Lay-Up Ply Lay-Up for Scarfed Laminate.

In all processes, the approved repair documentation must be followed explicitly. The following
steps are condensed from instructions in a typical OEM SRM and apply to a room temperature or
hot-bonded wet lay-up repair.

- Trace the ply contours onto a transparent sheet, such as vacuum bagging film, to create a
  ply template sheet. Ply count and orientation must be in accordance with the approved
  repair documentation. Mark the orientation and ply number on each template to facilitate
  lay-up.

- Smooth one sheet of vacuum bagging film onto the work surface. Place one layer of dry fabric onto this sheet, as specified in the approved repair documentation.

- Calculate the amount of resin required to impregnate the fabric, as specified in the approved repair documentation.

- Pour the resin over the entire fabric surface using a back-and-forth pattern to dispense evenly on top of the ply template.

- Place the ply template sheet over the fabric. Use a squeegee to wet-out and work the resin into the fabric as evenly as possible. Work the material until there is no air remaining between the vacuum bagging film and the fabric, taking care not to force the resin from the fabric edges. When the air is removed from between the ply template side, turn the parcel over and repeat the process to remove the air from between the fabric and the lower vacuum bagging film.

- Turn the parcel over so the ply template sheet is again facing up. Cut out the individual plies as marked on the ply template sheet. Verify the lay-up sequence by stacking the patch plies in numerical sequence prior to lay-up.

- Compact the repair plies as required by the approved repair documentation, using a temporary vacuum bag to remove any air trapped between the repair plies.

- All composite repairs must be conducted per the work instructions, which include steps that document successful inspection at the completion all significant processes.

A.4.2.2.3 Related Documents.

SAE References (Commercial Aircraft Composites Repair Committee—CACRC)
AIR4938: Composite and Bonded Structure Technician/Specialist: Training Document
ARP5143: Vacuum Bagging of Thermosetting Composite Repairs
ARP5256: Mixing Resins, Adhesives and Potting Compounds
ARP5319: Impregnation of Dry Fabric and Ply Lay-Up
ARP5701: Lay-Up of Pre-Preg Composite Materials

A.4.2.3 Demonstrate how to Prepare and Cure a Simple Bonded Repair to a Laminate Panel, and Explain the Types of Errors to Avoid.

A.4.2.3.1 Vacuum Bagging.

For room temperature cures, apply the same bagging procedure, except there is no need for the thermocouples and heat blanket.

Vacuum bag the repair for cure by applying perforated parting film, bleeder, parting film, caul sheet, thermocouples, heat blanket, breather, and vacuum bag per the repair documentation, as shown in figure A-44.
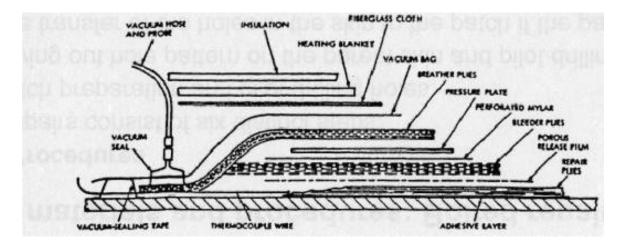
Figure A-44. Standard Lay-Up Cross-Section
(Courtesy Heatcon Composite Systems)

A.4.2.3.2 Repair Curing.

For room temperature cures, apply this curing procedure, except the hot bonder (figures A-45 and A-46), thermocouples, and heat blanket are not used.
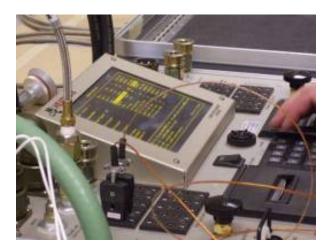

Figure A-45. Hot Bonder
(Courtesy Heatcon Composite Systems and Edmonds Community College)

Figure A-46. Hot Bonder Graphical Output
(Courtesy of Heatcon Composite Systems and Edmonds Community College)

Apply vacuum and cure using a hot bonder per the approved repair documentation. Care must be taken to ensure that the proper cure parameters are maintained, including vacuum pressure, ramp rates, and cure temperature ranges.

Visually inspect, tap test, or perform other NDI of the repaired area per the approved repair documentation.

There are a number of repair process mistakes that are almost certain to lead to inadequate repairs that may disbond or even rupture while in service.

It is essential that the protective coating (e.g., conductive coating if present, paint enamel, and primer) is removed using a prescribed method such as abrading or sanding.

All pieces of backing paper or separator film must be accounted for during ply lay-up. If the piece count is less than required, the piece may still be present in the ply stack. Under these circumstances, the repair plies must be removed until the missing piece is found or the lay-up process must be repeated.

All fluids must be removed from the component using vacuum and heat, otherwise the repair may contain moisture or moisture may be present in the bondline resulting in poor strength or even loss of the repair during operations.

If the abraded surface and tapered area is not cleaned sufficiently with an approved solvent, contaminants (e.g., dust) may be present in the repair bondline.

Prepregs and adhesives that have exceeded their allowable out-time or have expired must not be used, otherwise the resin may not wet-out the fibers properly during cure, which would lead to less than desired strength, stiffness, and durability. The use of old, boardy prepreg can also result in delaminations and bridging when used to repair curved parts.

It is essential that the right amount of the two-part resin be mixed correctly, otherwise the repair may not reach the desired state of cure, which would lead to less than desired strength, stiffness and durability.

If there are any leaks in the vacuum bag, they can be detected by leak checks and monitoring the heating unit readouts, otherwise the repaired patch and bondline may contain voids, excessive porosity, and delaminations.

The entire repair must be heated evenly to keep all sections of the bond within the parameters of the cure specification. The repair can be thermally mapped prior to scarfing to determine if the repair zone will exhibit nonuniformities. Substructure behind the repair zone can act as a heat sink and may need to be insulated or locally heated with a secondary heat blanket to enable even heating of the patch during cure. The heat blanket itself will sometimes fail to heat evenly if one of the elements within the blanket is damaged or malfunctioning. Thermal mapping of the blanket as part of scheduled blanket maintenance can alleviate most heat blanket failures.

If the cure process is not monitored closely, wide variations in thermocouple readings, incorrect heat-up rates and cure temperature, and loss of vacuum may result. If there is a lower or higher than acceptable thermocouple reading, an unevenly cured repair may result.

A.4.2.3.3 Related Documents.

SAE References (Commercial Aircraft Composites Repair Committee – CACRC)
ARP5144: Heat Application for Thermosetting Resin Curing

A.4.2.4 Prepare Bonded Repair for Cure, Including Bagging and Heating Apparatus and Cure.

A.4.2.4.1 Prepreg Bonded Repair Laboratory.

In teams of two, students will perform a 250°F prepreg composite repair in a controlled laboratory environment. The process steps to be followed are detailed in sections A.4.1.2.4 and A.4.1.3. Emphasis will be placed on following the work instructions in a step-by-step fashion to simulate a conforming repair process.

A.4.2.4.2 Wet Lay-Up Bonded Repair Laboratory.

In teams of two, students will perform a 200°F wet lay-up composite repair in a controlled laboratory environment. The process steps to be followed are detailed in sections A.4.1.2.5 and A.4.1.3. Emphasis will be placed on following the work instructions in a step-by-step fashion to simulate a conforming repair process.

Figure A-47 shows the standard heat blanket installation. Locations less than approximately 2 inches in from the edge radiate considerable heat, resulting in as much as 50° to 75°F cooler than the body of the blanket. Therefore, all heat blankets should extend a minimum of 2 inches beyond all caul plates and thermocouple locations so the midblanket temperature indications are

not effected. If this procedure is not followed, an over-temperature condition results in the repair, because the controller is sensing a cooler temperature.
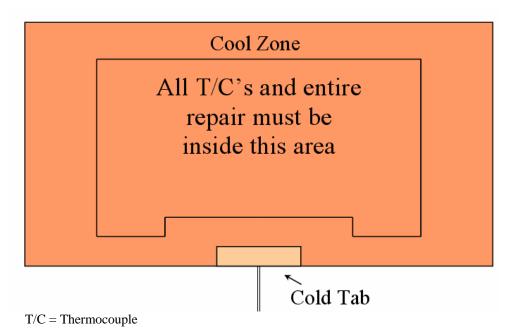

T/C = Thermocouple

Figure A-47. Heat Blanket Installation
(Courtesy of Abaris Training Resources Incorporated)

If any of the above repair process steps are not carried out correctly or missed, a defective repair may result. In the event a repair ply is omitted or laid down with the wrong orientation or if the heat blanket installation is incorrect, inadequate repair may result.

The following in-process QCs for a wet lay-up or prepreg vacuum bag repair have been condensed from detailed instructions in a typical OEM SRM.

- After the vacuum bag is installed over the repair and sealed, apply a minimum vacuum of 22 inches of Mercury, keeping that minimum level throughout the cure cycle, as shown in figure A-48.

- After 22 inches of Mercury has been attained, remove the vacuum source. Monitor the vacuum pressure, and after 5 minutes, the vacuum must be within 5 inches of the required 22 inches of Mercury. If the required level of vacuum is not maintained, the vacuum bag seal needs to be reworked and the vacuum process repeated. Two views of the vacuum bag setup are shown in figures A-49 and A-50.

- Increase the temperature of the heat blanket by 1° to 3°F per minute until the required cure temperature has been reached. Monitor the leading thermocouple for the heat-up rate.

- The cure temperature must be kept constant. As an example, for a 200°F wet lay-up repair, the temperature must be kept at 200° ±10°F for the required dwell time of 220 to 250 minutes.

- After the required cure dwell is finished, the temperature of the heat blanket must be decreased at a rate of 5°F per minute until a temperature of less than 125°F is reached.

- Print out the cure parameters, i.e., time, temperature, and vacuum profiles for repair records and for postrepair quality assessment.

- Release the vacuum pressure and remove the vacuum bag and heating equipment.

Figure A-48. Vacuum Bag Repair Preparation
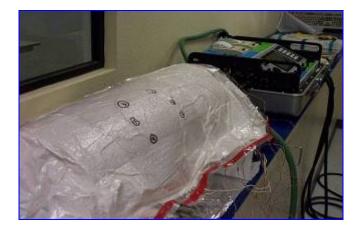(Courtesy of Abaris Training Resources Incorporated)

Figure A-49. Vacuum Bag Repair Setup (front view)
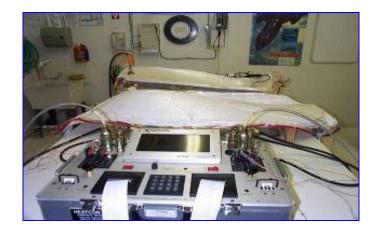(Courtesy of Abaris Training Resources Incorporated)

Figure A-50. Vacuum Bag Repair Setup (side view)
(Courtesy of Abaris Training Resources Incorporated)

Safety Message

The correct processing of composite components is critical to the elimination of processing defects. This includes all processing steps, such as surface preparation, material handling and storage, material lay-up, part bagging and cure, and postprocess inspection. Inspection of bonded composite and metal bond assemblies using appropriate NDI techniques, together with in-process QC, is essential to the delivery of defect-free composite components. Equally, adherence to the correct repair processing steps, in-process QCs and postprocess inspection is crucial to successful composite repairs. In addition to the above steps for the original part fabrication, composite repair procedures will require damage removal, and acceptable surface preparation for bonding. Note that these procedures may also include drying. The particular fabrication and repair methods for a given structural part will often include specific processing details that are unique to that part.

In the event of processing mistakes, misuse of equipment, or misinterpretation of in-process QCs and postprocess NDI results, composite part fabrication or repairs may be compromised. Parts delivered with unaddressed defects or with compromised repairs may not be airworthy.

A.4.2.5 Describe Process Parameters That Affect Bonded Repair Quality, and In-Process Controls Necessary To Avoid Defects.

The cure parameters that are critical to repair quality are:

• Maintain proper vacuum pressure—After the vacuum bag is installed over the repair and sealed, apply and maintain vacuum per the approved repair documentation. Typically, a vacuum leak check is required. If a vacuum leak is discovered, the vacuum bag must be resealed and the vacuum leak check performed again.

• Maintain proper cure ramps and hold temperatures—Increase the temperature of the heat source (e.g., heat blanket) at the specified ramp rate and hold the required cure

temperature for the specified time. At the end of the cure time, reduce the temperature at the specified rate until the part can be unbagged.

All process steps must be performed according to the instructions in the approved documentation.

A.4.2.5.1 Related Documents.

SAE References (Commercial Aircraft Composites Repair Committee—CACRC)
AIR5946: Design and Application of Composite Repairs for Thermosetting Composites

A.4.2.6 Describe Critical In-Process Quality Controls During Laboratory Bonded Repair Process Trials.

It is essential that during the entire bonded repair procedure, checks are performed to ensure that the approved repair instructions are followed and the appropriate repair materials are prepared and handled to avoid contamination.

The ply lay-up (including adhesive film ply), ply size, and orientation must be checked as each repair ply is laid down. If a debulk cycle is required, it must be performed per the approved repair documentation.

The vacuum bagging sequence must be checked at each step, and the vacuum seal must be checked by applying vacuum to monitor any vacuum loss.

The cure process must be monitored to ensure the specified cure cycle proceeds correctly. The heating apparatus (e.g., hot bonders) has controls for temperature and vacuum measurement, and some can monitor up to 10 thermocouples. If the cure process is not monitored closely, wide variations in thermocouple readings, incorrect heat-up rates and cure temperature, and loss of vacuum may result. If there is a lower or higher than acceptable thermocouple reading, an unevenly cured repair may result. A record of the cure vacuum and temperature profiles must be printed out for the postrepair approval procedure.

A.4.2.7 Describe Metal Bonded Repairs and the Differences From Composite Bonded Repairs.

A.4.2.7.1 Metal Bonded Repairs.

Metal bonded structures have been used in the aircraft industry for more than 60 years. Many aircraft still retain metal bonded parts.

For example, early models of the B-737 used metal bonding in the fuselage to increase skin thickness at joints and stringer interfaces. The latter was done to effectively prevent knife-edge conditions due to the use of countersink fasteners and to lower the stresses at the joints for fatigue reasons.

Current models of the B-737 do not employ metal bonding in the fuselage skins, but it is used in the main flaps and engine cowls, which are aluminum sandwich parts. A typical metal bond sandwich flap employs skins that have been chemically machined to decrease thickness, epoxy adhesive, and aluminum honeycomb core.

The cure process for metal bonding at the OEM usually employs an autoclave to apply pressure as well as vacuum and heat.

The repair of metal bonded parts was predominately performed by removing the damaged or corroded piece of skin or doubler and fastening a repair plate in its place. Repair of metal bonded sandwich parts usually employs a metal bonded repair plate, a replacement core, epoxy and foaming adhesives, and vacuum bag cure.

The most important parameters in effecting a good metal bond, whether for original part fabrication or for a repair, are surface preparation and environment. Early model aircraft experienced some metal bonding failures as a result of the original part bonding processes that used poor bonding procedures, including improper surface preparation and material handling. The following points are essential for effecting a successful metal bond repair:

- The surfaces must be cleaned with a pure, contaminant-free solvent per the approved repair documentation.

- A water-break test is often required per the approved repair documentation.

- Surface preparation must be correctly applied per the approved repair documentation. Often, processes such as Boeing's Sol-Gel and phosphoric acid anodizing are specified.

- Surface drying must be conducted per the approved repair documentation in a manner similar to composite structure.

- Surface preparation, cure vacuum pressure, ramp rates, and temperatures must be performed or maintained within the parameters of the approved repair documentation.

A.4.2.7.2 Differences Between Metal and Composite Bonding.

Primers are commonly used in metal bonding to wet both prepared and dried metal surfaces properly to ensure a good bond and ensure good bond durability. Primers reduce surface energy, which helps to reduce the attraction of contaminants to the surfaces and provides a surface that is highly compatible with the epoxy adhesives. Composite parts must be abraded, cleaned, and dried but do not need a coat of primer.

Metal bond requires chemical surface treatments such as Sol-Gel or phosphoric acid anodizing, whereas composite surfaces simply require solvent cleaning.

Metal bond repair surfaces usually require a shorter drying time than composites.

Metal bond and most prepreg repairs require the epoxy adhesive to enable the bond. Conversely, adhesives are not required for wet lay-up composite repair.

A.4.2.7.3 Related Documents.

SAE References (Commercial Aircraft Composites Repair Committee—CACRC)
AIR4844B: Composites and Metal Bond Glossary

A.4.3 DESCRIBE COMPOSITE LAMINATE BOLTED ASSEMBLY AND REPAIR METHODS.

A.4.3.1 Describe the Basics of Composite Bolted Structural Assembly. Show the Differences Between Composites and Metal-Bolted Assembly.

Using bonded composite parts in aircraft structures enables the elimination of thousands of mechanical fasteners that exist in similar metal components.

For example, in a metal airframe, the bonded or cobonded attachment of stiffeners or stringers to wing and fuselage skins has many thousands of mechanical fasteners. However, mechanical fasteners are still used for joining the more highly loaded composite elements and components.

The examples are wing and stabilizer skins-to-spars and skins-to-ribs, component-to-component (e.g., wing-to-side-of-body fittings and terminal fittings) and attaching fittings to elevator and aileron components. This use of fasteners in critical composite joints is due to the higher joint reliability of discrete fasteners, the improved inspection capability, and the need for possible disassembly during maintenance.

A composite mechanical joint is typically designed to be capable of transferring the maximum load capabilities of the parts being joined, regardless of the actual loads being transferred. This is common for mechanically fastened joint design and is a good rule regardless of the materials of the components being joined.

It is important to understand the effect that holes and loaded fasteners can have on the strength of the composite laminates being joined. An open hole in a composite laminate produces stress concentrations that can significantly increase the stresses at the edges of the hole compared to the stresses in the unnotched section of the laminate. The bearing stress of the fastener transferring load from one part to the other must be added to the stresses at the edge of the hole. All these stresses cause significant reduction in the laminate strength in the joint area.

A.4.3.1.1 Composite Versus Metal-Bolted Assembly.

The use of mechanical fasteners to assemble airframe structural components or elements is a mature technology. Composite part joining is no exception. Failure modes for composite-fastened joints are, in some ways, similar to metallic-fastened joints. Despite their similarities,

the behavior of composite-fastened joints differs significantly from metallic-fastened joints, and they deserve special attention because of the following reasons.

- Composite notch sensitivities—The notch sensitivity of composite materials, which result in significant reductions in the static strength due to the presence of a hole (e.g., approximately 50%-60% from a similar laminate without a hole).

- Composites have low Z-directional strength—Composite strength is largely based on the fibers, and as a result, most traditional laminates exhibit very little strength in the through-thickness Z direction. There are some composite laminate forms that contain through laminate thickness fibers (e.g., 3D braided composites), but most large composite components are laminates without this feature. A metal such as aluminum has similar properties in all three directions, although some aluminum alloys have poor through-thickness tensile strength.

- Composites are anisotropic—Composite-fastened joint strengths are a function of the actual laminate, i.e., ply-stacking sequence, percentages of plies in each direction (e.g., 0, +45, -45, and 90 degrees to the loading direction), fiber volume, etc. Composite laminates consist of anisotropic layers and the laminate lay-ups can change this anisotropy toward isotropy. A lay-up of 25% 0°, 25% +45°, 25% 90°, and 25% -45° plies yield a quasi-isotropic laminate, which almost equates to a metal isotropic part. This kind of laminate lay-up is probably the best type for composite-fastened joints. It has the same percentage of fibers in all the standardized directions, thus allowing for a load in any direction to have good load paths around the hole.

- Composites are more susceptible to fastener deformation—Fastener flexibility during loading can have a significant effect on the strength of a composite-fastened joint than a metallic-fastened joint. Excessive bolt bending is problematic for both metal- and composite-fastened joints and can lead to reduced strength. In a composite joint, the increased local bearing stresses on the outer plies of a laminate, due to bolt bending, are more critical. This can lead to delamination and microbuckling of the fibers, resulting in crushing the outer plies, which reduces joint strength.

- Composites are more sensitive to fastener clearance—Efficient load transfer from fasteners to the parts being joined can have a big effect on the joint capability. For a given configuration, the fastener fit in the holes is important to this efficiency. If fasteners have a close-tolerance fit in the holes, they not only transfer load efficiently, but can also prevent hole wear. Critical joints in metal structures usually employ close-tolerance, transition-fit and, in many cases, actual interference-fit holes. When bolts are inserted into close-tolerance or transition-fit holes in relatively thick metal parts, they often have to be firmly tapped in with a hammer. This cannot be the case for composite-fastened joints. Any kind of close-tolerance or transition-fit fasteners will cause damage to the holes in the composite laminate parts when installing the fasteners. Lack of fit can have a serious effect on the efficiency of composite-fastened joints. The need to have some clearance in the holes for composite-fastened joints called for significant research to optimize composite drilling, hole fit, and hole and fastener tolerances. The research

results led to using holes, fastener sizes, and tolerances that effectively result in a fit that is between close tolerance and what is called Class 1. This type of fastener fit has proved to be effective in providing good load sharing in multiple-fastener joints.

- Fastener clamp-up issues when assembling composite parts—Care must be taken when mechanically assembling composite components. Using the fastener clamp-up to close gaps between parts can lead to delaminations. Metal parts can bend to accommodate moderate gaps, whereas composite laminates tend to delaminate if subjected to local abrupt changes in curvature. It is essential to use some form of shimming if part tolerances or warpage cause fit-up problems. There have been several lessons learned in this area leading to the redesign of composite tooling, shimming techniques suitable for composite assembly, and reductions in material thickness variations so gaps are either eliminated or significantly reduced.

A.4.3.2 Describe the Basics of Composite-Bolted Repair. Show the Differences Between Drilling and Cutting Composites and Metals.

A.4.3.2.1 Bolted Composite Repair Basics.

Bolted or bonded repairs are made to damaged composite parts to restore the original component strength and stiffness properties.

Typically, the ideal bolted repair will follow the same design guidelines as the original manufactured component so failure of the repaired part does not occur in the bolted repair area. To this end, the selection of the repair doubler, fasteners, number of fasteners, fastener pattern, and spacing must be carefully considered.

The main consideration for repair of any aircraft component is that, in general, aircraft components, such as wing, stabilizer, and fuselage skins, are loaded in multiple directions.

A bonded repair, if performed correctly (i.e., a sufficient number of repair plies and a good bond), has the potential to restore the component's mechanical properties in all directions, without the need to understand the component design loads.

To ensure this for a bolted repair, the repair has to be carefully designed and knowledge of the component strength and stiffness is essential. Drilling holes in a component that may be loaded in multiple directions can have significant effects on the capability of that component. With the addition of bearing stresses and stress concentrations due to fasteners, the state of stresses at the fastener holes can be higher than the component was designed for.

Because most repair technicians do not have knowledge of the component design loads, when repairing a damaged component with a bolted repair, it is essential to follow the repair instructions in the source documentation.

A.4.3.2.2 Bolted Repairs to Damaged Composite Structures are Often Performed.

One reason for a bolted repair is to temporarily repair a part if the time or facilities to perform a permanent bonded repair are limited.

More importantly, due to the increased use of composite material in critical components on military and large commercial aircraft, permanent bolted repairs are being used to repair damage to thicker composite laminates that comprise highly loaded components such as wing or stabilizer skins and spars.

Bolted repairs may be smaller in area than some bonded repairs and, thus, are often used for the repair of composite parts when the thickness of the part being repaired requires a very large scarfed-out area for bonding.

Typically, a 30:1 scarf angle is used, which can cause the area of the scarf to be more than double the size of a bolted repair area. For example, for a 2-inch-diameter hole in a 0.5-inch-thick laminate, a bolted repair might cover 144 in^2, while the area for a bonded repair with a 30:1 scarf angle might cover 800 in^2.

Bonded repairs to a component with complex geometry (excessive ply drops, cobonded or cocured stiffening structure, or bolted spars and ribs) can be quite difficult to implement due to the intricate scarfing and patch lay-up requirements. This is also true for bolted repairs of composite structure with these kinds of complex structural details.

There are significant differences in drill types, lubrication, speed, and feed rates for drilling metal versus composite parts.

Drilling metal parts is typically performed using steel drill bits, a relatively slow drill speed, and a lubricant. For example, the speed for drilling metal parts (e.g., titanium) is usually in the order of 450 revolutions per minute (rpm) for drilling a 0.25-inch-diameter hole with the drill bit lubricated with Boelube or a similar lubricant such as cetyl alcohol.

Drilling composite parts requires unique procedures. Fiber breakout or delaminations can occur during drilling composite parts if proper procedures are not followed.

Drill bits for composites are usually solid carbide or diamond tipped, with different tip geometries. A carbide drill bit is used because steel drill bits wear out very quickly when drilling carbon- and glass-fiber materials, which are very abrasive.

Speeds for drilling composites should be much higher than for metals. Using the same example, for drilling a 0.25-inch-diameter hole in a CFRP part, drill speed needs to be 5000 rpm and drilling is often performed without a lubricant. There are several reasons for the higher drill speed: (1) eliminate splintering of the edges of the hole, (2) reduce pressure, and (3) use low feed rates. Excessive pressure (i.e., high feed rates) during the drilling operation can cause delaminations and fiber breakout.

To eliminate defective holes (delaminations and fiber breakout), high drill speeds and slow, controlled feed rates are used. Fiber breakout of the last ply to be penetrated by the drill bit can be avoided by using a wooden backing block during the drilling operation.

Carbon composite parts fabricated from unidirectional tape are very difficult to drill without fiber breakout. Typical production parts that are constructed with tape or tows will employ fabric for the surface plies to eliminate or greatly reduce fiber breakout when drilling. It is highly recommended to refer to the drilling procedure section of the source documentation (e.g., SRM) before performing a drilling operation in composite materials.

Drilling titanium and composite stackups: The low speeds and high feeds of metal will cause overheating and fiber breakout in composite laminates. As such, the metal and composite materials are quite often drilled separately using feeds and speeds as previously indicated. A final ream of the hole through both parts is then conducted to ensure proper hole quality.

A.4.3.2.3 Composite Materials Behave Differently Than Metals During Machining.

Cutting and machining composite parts requires different processes and equipment than metals.

When performing composite repairs, surfaces are often prepared with sanders and grinders. Sanders and grinders are high-speed hand tools used for light sanding, feather edging, and cutting of tapered and stepped scarfs for bonded repairs. A high skill level is required when using these tools because of their high speeds (e.g., 18,000 rpm or faster). Metal repairs are often repaired with routers and various metal-working tools.

Parts made with aramid fibers require special machining and cutting techniques. Aramid fibers tend to fuzz when cut. This can also happen when cutting or machining aramid honeycomb core.

Special efforts are made to keep machining and drilling temperatures down when working on composite parts. Machining temperatures should be kept below the T_g of the material being worked on. For many epoxy materials, the T_g is the temperature at which a material transitions from being rigid and brittle to becoming rubbery and flexible. Drilling or cutting at temperatures above the material T_g may cause clogging. Typically, this means keeping the temperature at the cut edge below approximately 150°F to ensure clean cutting. High temperatures at the cut edges are caused by high speeds and pressures. The high speeds used for machining or drilling composites requires that pressure be kept low by significantly reducing the feed rate below that typically used for machining or drilling metals.

It is important to follow the drilling instructions in the source documentation, because they typically detail feed and speed rates that will ensure the temperatures at the cutting edge will be below the material T_g.

A.4.3.2.4 Debris and Shavings Created During Machining.

When machining, cutting, or drilling metals, the debris that is removed from the material comes away in the form of shavings or chips. These metal shavings are often carried away by the lubricant or forced air that is typically used when machining, cutting, or drilling metals.

For composite materials, the higher cutting and machining speeds cause the debris to come away in the form of a fine dust. To prevent the repair technicians ingesting this dust, approved vacuum equipment must be used with all composite cutting, grinding, or drilling tools. Pneumatic tools with rear exhausts are recommended, because air is directed away from the work and dust is not blown around the shop. Vacuum attachments can also be used for dust extraction.

A.4.4 PERFORM A SIMPLE BOLTED COMPOSITE REPAIR.

A.4.4.1 Demonstrate Composite Drilling Versus Metal Drilling.

The instructor will demonstrate drilling operations for titanium sheet and carbon composite laminate. The differences needed in drill speeds and feeds required to successfully drill holes in the CFRP laminate will be shown. The following is an example for setting up a demonstration.

Titanium sheet drilling operation: Mark the fastener pattern on the titanium sheet, then place the sheet in a fixture and, using a #30 ST10-907-J2 steel drill bit, drill (at 750 rpm) the fastener holes in the doubler, using Boelube to lubricate the drill bit. When all the fastener holes have been drilled, remove all burrs.

Carbon laminate contrasting drill of feeds and speeds: Mark the fastener pattern on the composite laminate, then place the laminate in a fixture. Using the same drill bit (i.e., #30 ST10-907-J2 steel drill bit) and the same drill speed (750 rpm) used for the titanium sheet, drill two fastener holes in the composite laminate. (A dust capturing vacuum will be used during all drilling operations.) Feed the drill at a much slower rate than the first two holes. Next, replace the drill bit with a #30 ST1257B solid carbide drill bit, and drill the next two holes using a 5000-rpm drill speed and a wooden backing block.

Compare hole quality: Remove the composite laminate from the fixture and examine the holes. (A borescope may be used to further evaluate hole quality.) The second pair of holes will be superior to the first pair of holes. The first pair of holes drilled may exhibit fiber breakout on the drill entry side and will most certainly exhibit fiber breakout and possible splintering on the drill exit side. In addition, the first pair of holes drilled may exhibit delaminations within the holes. The second pair of holes drilled with the carbide drill bit, increased drill speed, decreased feed rate, and wooden backing block should compare favorably with the holes drilled in the titanium sheet using the steel drill bit and lower drill speed.

A.4.4.2 Describe Process Parameters That Effect Bolted Composite Repair Quality and In-Process Controls Necessary to Avoid Defects.

There are a number of critical processing parameters that can affect bolted composite repair quality.

- It is essential that the surfaces are clean and there is no protruding damage (e.g., fibers) that may prevent the doubler and base composite part from mating properly. If a repair doubler is sitting up and not in contact with the part being repaired, fastener installation may be affected, e.g., effectively changing the fastener grip length.

- Good, consistent fastener fit is essential for good load sharing in a repair fastener pattern. If the holes have been drilled with varying tolerances or at off-angles, the fasteners will not all fit in the same manner, and some fasteners will unload before others, increasing the bearing stresses in those fastener holes. In extreme cases, holes with variable fits can cause the bolted repair to fail.

- If a fastener has insufficient grip length (the part of the fastener without threads), concentrated bearing stresses on the fastener shank may result. Also any threaded portion of the fastener that bears on the laminates being joined may cause damage inside the holes, as well as creating the potential for incorrectly clamped sleeves. Incorrect grip length fasteners have contributed to bolted repair failures, regardless of the materials being repaired.

- Fasteners must be installed per applicable specifications with proper clearance, sealant, sizing, and clamp up. If the fastener is incorrectly installed, i.e., with insufficient clamp or unsatisfactory sleeve formation, the designed repair strength may not be attained. In some cases where the back face of the part being repaired is not easily viewed, it is very important to perform the fastener installation correctly using the specified blind lockbolt installation tool.

- If the sealant between the repair parts and the edge seal was not applied correctly, moisture paths may result, which can lead to fastener corrosion, freeze-thaw damage, and laminate moisture absorption through the damage area.

- Due to poor tolerance control, warpage, and other geometric variations, gapping can occur when using bolted repairs on damaged composite components. It is essential to make sure that the repair plates have the same contours as the composite parts being repaired. If moderate gapping still remains after forming the repair plate, then laminated, peelable shims are recommended.

- All repairs depend on technician workmanship and the ability to comply with work instructions and applicable specifications while performing the repair. As such, it is advantageous to have two technicians (or a technician and an inspector) involved in the repair process so one technician can perform the repair step, and the other technician or inspector can verify that the step has been performed correctly.

All composite repairs must be conducted per the work instructions that include steps to document the successful inspection at the completion all significant processes. A second person must perform the inspection activities. This second person is responsible for verifying the correctness of the process step and for signifying on the work instructions that the process step was verified. This system of checks and balances creates a reliable method to ensure that the repair process has been performed correctly.

<u>A.4.4.3 Demonstrate and Apply Common Damage Removal, Surface Preparation, Drilling, and Fastening Techniques Used for Bolted Composite Repairs and how to Inspect Them for Acceptability</u>.

Students will participate in the bolted repair of damaged laminate panels in a controlled laboratory environment using titanium repair plates and fasteners and will inspect the repairs for acceptability. The following was summarized from a typical OEM SRM bolted repair equipment list.

- Damaged eight-ply carbon laminate panels (15 inches by 15 inches)
- Titanium doubler, precut and prepiloted (0.063 Ti-6Al-4V annealed sheet)
- Protruding head blind bolts (Monogram 98524-8 or Fairchild 5M902-8)
- Blind bolt installation tools
- Temporary fasteners (CBX-BF (Bigfoot))
- Cylindrical temporary fastener installation tools
- Boelube or cetyl alcohol for four drilling workstations
- Drill motors
- #30 ST1257B carbide drill bits
- #10 ST10-937-B steel drill bits
- #10 ST1257B carbine drill bits
- 0.2600-inch ST 186 4R carbide reamer
- Drill fixtures
- Solvent cleaner

The instructor will demonstrate the damage preparation, doubler, and composite drilling operations, and fastener installation procedure. The students will be paired off to perform a bolted repair.

The bolted repair procedure is as follows:

- The damage is to be cleaned up, i.e., remove any damaged materials or broken fibers.

- Ensure that the damaged area and adjacent surface of the part are smooth and flat for the repair doubler.

- Clean the area with an approved solvent.

- Seal the damage, if applicable (i.e., if called for in the source documentation).

- Retrieve a precut, #30 piloted titanium repair doubler with a surface finish of 125 microinches Ra or better. Deburr all cut edges, corners, and holes to remove sharp edges.

- Clamp the doubler onto the composite part to ensure that it does not move. Pilot drill all fasteners holes (#30 at 5000 rpm) in the repair component, using a wooden backing block, as shown in figure A-51. Use a slow feed rate to drill the holes in the composite laminate. Use a star-pattern drilling sequence, as shown in figure A-51, to avoid doubler creep due to thermal expansion. Install a temporary fastener in each hole after drilling to ensure the doubler and part do not move.

- Remove the doubler and place in a fixture. Drill out all the piloted holes (#10 ST10-937-B steel drill bit at 550 rpm) using Boelube as drill lubricant and remove all burrs.

- Reclamp the doubler onto the composite part aligning the fastener holes. Drill all holes in a star pattern using the doubler as a guide (#10 ST1257B carbine drill bit at 4500 rpm), using a slow drill feed rate and a wooden backing block to prevent fiber breakout. Install a temporary fastener in every other hole to ensure that the doubler does not move during drilling.

- Hand ream all holes to full size (0.2605- to 0.2630-inch diameter) at a 150- to 300-rpm reamer speed. Install temporary fasteners to ensure that the doubler does not move during reaming.

- Remove the doubler and deburr all the holes on both the doubler and composite part. Chamfer the edge of all holes on the fastener entrance side of the doubler to the same diameter as the radius on the underside of the fastener heads.

- Use solvent to remove any residual Boelube from the titanium repair places, carbon panel, and reamed holes.

- Apply one coat of sealant to the mating surface of the doubler and composite part.

- Place the doubler over the composite part, aligning the holes in each part. Install temporary fasteners in each corner of the fastener pattern, as shown in figure A-52, and then install temporary fasteners in every other hole so the opposite holes are clamped on each side of the symmetry line.

- Select appropriate fasteners of the correct diameter and grip lengths.

- Install fasteners in the open holes through the squeezed out sealant.

- Remove all the temporary fasteners and install the permanent fasteners in the open holes through the squeezed out sealant.

- Apply a fillet seal around the repair doubler.

Figure A-51. Drilling Operation on Doubler
(Courtesy of Edmonds Community College)

Figure A-52. Temporary Fasteners on Doubler
(Courtesy of Edmonds Community College)

A.4.4.4 Verify Correct Fastener Selection, Inspect Drilled Holes, and Check if Fasteners Were Properly Installed During Bolted Composite Repair Laboratory Trials.

Note: This laboratory exercise is meant to serve as an example for how the indicated composite repair principles can be taught. The following QC steps are condensed from detailed instructions in a typical OEM SRM.

- Inspect the repair doubler for correct material, thickness, dimensions, flatness, and finish.

- Inspect damage area for any protruding damage.

- Inspect the marked fastener edge distances and spacing on the repair doubler prior to drilling holes.

- Inspect all fastener holes in doubler and composite panel for correct size and condition. Perform this inspection for each increment of the drilling procedure.

- Inspect repair doubler and composite laminate for cleanliness after they are cleaned with the solvent.

- Inspect the selected fasteners to see that they are the correct size and grip length.

- Inspect faying surface sealant application to make sure that the sealant completely covers both mating surfaces.

- Inspect to see if all fasteners are correctly installed. Inspect the back side of the repair to see if the fastener sleeves are satisfactorily formed.

- Remove and replace any fasteners found to be incorrectly installed.

- Inspect the edge seal to ensure that there are no gaps to allow moisture to ingress the repair.

Safety Message

In general, there will be differences in the machining and fastening steps needed to complete a composite bolted repair compared to a metal-bolted repair. Specific differences in bolted repairs should also be expected for different composite structures. The tolerances for installation of fasteners during composite bolted repairs are as critical as those for installation of fasteners during the repair of metal parts. Drilling composites is more difficult, and the selected fasteners should not cause delamination during installation. Details of the bolted repair design are also of great importance. A bolted repair design that is not substantiated by approved data may cause problems for component margins of safety. Fastener installation equipment, such as torque wrenches, must be correctly calibrated and standards kept up to date.

The design of the bolted repair, selection of incorrect fasteners, poorly drilled holes, and incorrect fastener installation can all contribute to a defective or unapproved repair, and hence, flight safety may be impaired.

APPENDIX B—CURRICULUM DEVELOPMENT PROCESS AND OUTCOMES

B.1 CURRICULUM DEVELOPMENT PROCESS FOR INDUSTRY STANDARD.

The development process included much involvement by industry and academia to achieve a consensus worldwide. The Commercial Aircraft Composite Repair Committee (CACRC) had significant input. In addition, other platforms were used to provide opportunities for input by experts in the field. The most significant activities are described below.

B.2 COMMERCIAL AIRCRAFT COMPOSITE REPAIR COMMITTEE.

The CACRC has biannual meetings to provide an open forum for the exchange of ideas between the various CACRC task groups, as well as presentations from airlines, original equipment manufacturers (OEM), and regulators related to the use of composites on aircraft. The CACRC is an international organization composed of personnel from aircraft manufacturers, airlines, third-party maintenance depots, research agencies, and composite material suppliers. Its mission is to reduce the cost of maintenance and ensure the safe use of composite aircraft structures by developing new and improving existing materials, techniques, and training. The sponsoring agencies are the Air Transportation Association of America (ATA), International ATA, and Society of Automotive Engineers. The CACRC task groups are Repair Materials, Repair Techniques, Analytical Techniques, Design, Inspection, Life Cycle Modeling, and Training. There are approximately 50 meeting participants.

B.3 ADVANCED MATERIALS IN TRANSPORT AIRCRAFT STRUCTURES COMPOSITE MAINTENANCE WORKSHOP (NOVEMBER 30 - DECEMBER 2, 2004).

1.   Workshop Executive Summary

        The goal for this workshop was to achieve a consensus for composite maintenance and repair.

        Specifically, the objective of the workshop was to provide terminal course objectives (TCOs) for a 5-day survey course and laboratory workshop regarding composite repair with the following vision:

        •   Students at the end of the course will have a common foundation of understanding the maintenance, repair, and handling of composite materials. Exposure to simple repairs will prepare the students for more advanced training. At the end of the course students will be able to:

                -       Understand the basic art of composite maintenance and repair, including the design issues associated with airframe structure.

                -       Describe the materials, processes, and key quality controls (QC) used for bonded and bolted repair methods.

- Understand the roles, responsibilities, and relationship of technician, inspector, and engineer in the composite repair process.

- Identify sources of technical data and regulatory requirements.

- Produce composite laminate, damage it under controlled conditions, and make the proper repair. Identify differences in critical damage types, inspection methods, and repair procedures for composite and metal structures.

- Perform damage inspection on a composite part produced by another student.

- Assess effectiveness of the student-repaired composite part.

- Distinguish between proper procedures for repairs that are and are not included in the source documentation.

2. Overview

Over 50 individuals, selected for invitation because of their extensive backgrounds in composites repair, attended the workshop. They represented academia, industry (OEMs, airlines, and maintenance and repair organizations), the Federal Aviation Administration (FAA) and U.S. Air Force, and professional training organizations, originating from a wide geographic spectrum. The workshop was sponsored by the FAA, Advanced Materials in Transport Aircraft Structures, and The Boeing Company. The workshop was subdivided into three smaller work groups, which represented engineers, technicians, and inspectors. Each group was facilitated by Boeing professionals who provided critical expertise to the workshop process.

3. Outcomes

Each group provided vital information for the successful development of the curriculum, such as draft TCOs and a list of 150 essential skills. Equally important, the broad representation of personnel in the workshop provided a sense of ownership and responsibility to a broad spectrum of the industry for the resulting curriculum development. Educational resources were identified for potential incorporation into the class and included videos on composites repair from the U.S. Air Force, Alteon, and Heatcon Composite Systems.

- Prevention and Reporting of Damage
- Basics of Composites
- Mishap and Post-Crash Handling
- Operation of Hot Bonder

4.   Workshop Overview

- Objective: Establish fundamental baseline training for composites maintenance and repair training course(s) to achieve a common level of understanding for technicians, engineers, and inspectors.

- Format: Three subgroups to consider unique requirements in separate breakout sessions and provide feedback and conclusions.

- Vision for training students: The goal is that, at the end of the course, a common foundation of understanding the maintenance and repair of composite materials will be achieved and provide a foundation for more advanced training.

5.   Sample Terminal Objectives

- Use basic repair techniques (subordinate objectives to be determined (TBD))

- Read and follow source documentation and procedures (subordinate objectives TBD)

- Determine selection criteria through assessment of damage for alternative repair solutions (subordinate objectives TBD)

- Mitigate technical risks (subordinate objectives TBD)

6.   Workshop Goal

Each group provided training objective recommendations from the group's perspective and made presentations to the audience for discussion and comment concerning the recommended desired state of training. Each group had an overall facilitator, supported by a technical facilitator and a note taker to capture information being discussed. The process was designed to result in the following information at the end of the workshop:

- Course Objectives
- Obstacles to Overcome
- Alternative Approaches
- Parking Lot Issues (out-of-scope issues)

7.   Conduct of Class

While a structured approach was provided to the facilitators, it was made clear that the facilitators, at their discretion, could deviate, depending on the dynamics of the work group.

8.   November 30

Session 1 Objective: After the welcome comments, participants were given an opportunity to describe the current situation and why the workshop is needed. Six groups were selected randomly to discuss composite repair scenarios from actual experience and present one story to the audience. The participants were requested to select one of the following topics for a full session presentation, commenting on how the scenario could have been improved through training.

- "How could a repair process be made more efficient by avoiding unnecessary rework?"

- "Knowing what you don't know: An instance where someone didn't know when to ask for help"

- "The Temptation of Shortcuts: An instance where shortcuts could have caused a bad situation"

- "Understanding Roles: How one person didn't understand the interdependence with other organizations and other roles in the process"

- A variety of scenarios were presented, including the following topics:

    - Repair a hole in a leading edge

    - Metal overlay repairs

    - Structural Repair Manual issues

    - The difficulties of replacing metal structure with composites in a manufacturing line

Presentations were made from the representatives of the CACRC, and the Alteon and Abaris training organizations. A number of issues were identified.

Session 2 Objective: Workshop participants were divided into three work groups to address composite maintenance and repair for three types of composite construction: thick laminate, laminate and honeycomb flight-control surfaces, and honeycomb fairing panels. Groups were represented by three principal practitioners, engineers, technicians, and inspectors. Each work group outlined steps required in composites maintenance and repair and identified key personnel involved in the process. This laid the groundwork for discussing required skills in Session 3.

Each group filled out a template reflecting the discussion and listing the repair steps.

9.   December 1

Session 3 Objective: Identify the skills of each work group (engineers, technicians, and inspectors) by addressing the question, "What basic skills do we use, what are the tools, source documentation, and techniques we use, and how did we learn them?"

Based on the repair steps identified in Session 2, each work group identified specific skills to perform the repair, including the tools, documentation, and techniques for each skill.

Session 4 Objective: By comparing the current state of training for essential skills, and what the desired training ought to be, course objectives were to be decided by the work groups, addressing the question: "What should the training objectives be, considering improvements regarding current training?"

At this point in the process, it was clear that the number of essential skills were numerous (up to 150 per work group), and that the process would have to be modified to meet the objectives of the workshop. By group consensus, the decision was to categorize each skill on a level of 1 to 3, with 1 identifying exposure of students to the skill, and 3 identifying more focus for the curriculum development. From the level 3 skills and considerations of the other desired skills, which would be covered to a lesser extent, each work group identified TCOs that are summarized in section 4.

Session 5 Objective: Each group provided their conclusions, presented by their respective technical facilitators. Invitations were extended for final comments (see Essential Issues below) and solicitations were made for training resources. Educational resources were identified for potential incorporation into the class, and included, for example, videos on composites repair from the U.S. Air Force, Alteon, and Heatcon Composite Systems with the following video titles:

- Prevention and Reporting of Damage
- Basics of Composites
- Mishap and Post-Crash Handling
- Operation of Hot Bonder

The following essential issues were discussed.

- Urgent request to have an FAA-sanctioned training standard in composites repair

- Suggestion that composites repair training be mandatory (i.e., a regulatory requirement)

- Composites repair training exists—it is the implementation of training that is the concern

- Course will be universally available—by reaching consensus on the course content, standards and regulations can be developed from this baseline

    - Logistics—distance, cost, and time for students—alternative is to use web-based training

    - Involve employers to create priority and importance of training for students

    - Advanced training—certain aircraft have special paint requirements, possibly requiring advanced training

    - Five days may not be sufficient to meet the terminal course objectives

    - Perception by technicians is that this course is a final course, rather than a survey course

    - Training location logistics—who pays for this?

    - Target audience for this course is not defined sufficiently

    - Teaching materials—availability may be an issue, especially manufacturer's approved data

    - How will this course be integrated with the existing CACRC curricula?

    - Will the FAA recognize CACRC documents as acceptable/source data?

    - TCOs outcomes from 2004 Workshop

Inputs collected at the workshop are organized by TCOs, and subordinate objectives to the TCOs. It was estimated that a course, which includes all the objectives as currently worded, would exceed the current time available in a 5-day survey course. As a result, some objectives linked to a basic understanding of composite materials were moved to prerequisites to be covered in self-study or through other basic knowledge held by students entering the course. As conceived after the workshop, short, web-based learning modules, including a basic entrance test, were envisioned for any prerequisite. Some TCOs, as identified in the workshop, were identified as advanced topics, requiring study beyond the 5-day survey course.

B.4 WORLDWIDE TELECONFERENCE (APRIL 2005).

1.   Workshop Executive Summary and Essential Issues Discussed

A teleconference was conducted from Edmonds Community College, inviting attendees from the November 2004 workshop and others to discuss the progress in developing terminal objectives. Topics discussed included:

-       Using the Blackboard Academic Suite for distance learning class, with a description of how this will work in an asynchronous teaching format.

-       The potential for students not having sufficient fundamental composites knowledge in the prerequisite course and the importance of assessment tools.

-       The lack of OEM approval, to use structural repair manuals (SRM). This was discussed as important for students to appreciate the need to use approved source documentation in repairs. It was proposed that using B-757 or B-777 SRMs might reduce Boeing sensitivity to the use of SRMs.

-       Future opportunities for feedback (Chicago September 2005 workshop was discussed in this context)

-       The importance of using standardized panels in the laboratory section of the class

2.   Technical discussions.

        Topics discussed during this session included proper drying techniques prior to composites repair and scarfing as well as pulse echo versus coin tap testing.

B.5 CACRC ATTENDANCE (BREMEN, GERMANY) (MAY 2004).

1.   Workshop Executive Summary and Essential Issues Discussed

        Progress in the course development was presented first to the training committee for initial feedback, followed by a presentation to the main group. A proposal was prepared and jointly delivered by C. Seaton and M. Hoke (President, Abaris) to support the development of a generic SRM. This was presented to the main steering committee.

2.   Numerous testimonials were taped for later use in the course in order to add meaning before detail and to emphasize the importance of composites' maintenance awareness related to safety. A guidance sheet was prepared for each speaker to discuss one aspect of the safety messages to provide consistency in format and approach.

        A guidance sheet for the taped testimonials was provided, which included the following script:

                "My name is _____, and I have (description of a few roles in composite materials as related to aircraft repair and maintenance, or

design)" 1 to 2 minute testimonials to introduce course modules (below are options). Examples to illustrate points made are especially effective.

a.   Safety Message 1

Provide a personal perspective on the importance of teamwork and following procedures.

Discuss an instance whereby the teamwork among inspectors, technicians, engineers, and others either did or did not result in a satisfactory repair.

Discuss a time when following maintenance procedures enhanced the safety of an aircraft.

b.   Safety Message 2

Describe an instance whereby approved data was made available that substantiated structural integrity.

Describe options for the practitioner in the field for acquiring approved data, such as the SRM.

c.   Safety Message 3

How does "knowing what you don't know" play an important role for the practitioner?

Discuss an instance when following approved maintenance instructions for a given part was mandatory for aircraft airworthiness.

Describe how practitioners should be aware of the importance of understanding instructions by at least one member on a team, and consequences of one such person not being available.

d.   Safety Message 4

Describe the value of experience in remote areas for properly processing composites that are under repair.

Describe the various processing and nondestructive inspection (NDI) options available to a practitioner, and how the practitioner can better prepare themselves to select the appropriate and best alternatives.

e. Safety Message 5

Describe an instance whereby a trained practitioner discovered and dispositioned a damaged part, playing a critical role in aircraft safety.

Discuss the role of maintenance procedures, from your experience, for the successful maintenance of composite components.

f. Safety Message 6

Describe how experienced practitioners play an important role in bonded repairs.

Compare bonded to bolted repairs in terms of (1) application and (2) inspection techniques and difficulty, and how practitioners have responded to these situations.

g. Safety Message 7

Discuss a situation where proper interpretation of NDI resulted in a satisfactory repair and disposition of a composite part.

Describe how defects, due to poor surface preparation or bondline defects, required expertise in proper NDI procedures.

h. Safety Message 8

Describe the specific skills that are needed for bolted repairs, and how this might differ from those in bonded repair.

Talk about the need for approved data to support bolted repairs.

Discuss unique issues that practitioners should pay attention to in bolted repairs; use a specific experience (good or bad) to illustrate this point.

i. Safety Message 9

How is the understanding of one's limitations critical to successful maintenance and repair?

Where can a practitioner go for help when specialized knowledge is needed?

B.4 CACRC ATTENDANCE (CHICAGO, ILLINOIS) (SEPTEMBER 2005).

1. Workshop Executive Summary

The basis for the workshop was to ask industry and government experts to support the development of training standards. Whereas the initial workshop defined TCOs, this workshop was used to review documented modules that will be released with the TCO as industry standards as a potential precursor to FAA policy.

2.   Critical issues, as defined at the beginning of the conference, included the need to

- understand roles and responsibilities (importance of teamwork).

- recognize composite damage types and sources (proper team reaction to possible service damage).

- understand the inspection methods and procedures needed for detection, characterization, and disposition of damage.

- understand regulations and the importance of approved source documentation (and process for cases requiring new approval).

- realize the unique processing issues and QCs needed for bonded composite repairs.

- realize the unique processing issues and QCs needed for bolted composite repairs.

- realize the need for more training to acquire technician, inspector, or engineering skills (avoid working beyond skill limits).

3.   Objectives

The primary objective was to review technical details that need to be included as a basis for maintenance and repair training with a focus on critical safety issues. Secondary objectives include:

- Discussing industry engineering practices, which are needed in training modules to authenticate safety messages

- Identifying additional training development needs

- Providing directions for future research and development

4.   Process Description

a.   Preworkshop

Teams were assembled to address specific key subject areas in base knowledge, teamwork and disposition, damage detection and characterization, and repair processes. Each team reviewed the specific modules related to the assigned key subject in preparation for opening feedback comments to the workshop assembly (Session 3).

b.   September 13, Sessions 1 and 2

Sessions 1 and 2

Speakers during Sessions 1 and 2 provided an overview for the workshop. Speakers represented the following organizations:

- FAA (logistics, agenda, process, introduction, background)
- Edmonds Community College (development process)
- Moderators for each key subject

c.   September 14, Sessions 3-5

Session 3

Feedback from expert reviews and participants considered the following areas:

- Documented standards

- Modules and TCOs

- Safety messages and testimonials

- Specifically, the team leaders of each preworkshop topic (base knowledge, repair process, documentation, and damage types and sources) and the assigned participants were asked to assess consistency between TCOs and content and balance between the issues involved in the curriculum development. The comments from the assembly were sought and recorded by an administrative assistant.

Session 4 (breakout session)

Feedback continued with small group interactions in breakout sessions. Multidiscipline teams consisting of less than 20 experts were given an opportunity to provide discussion points and issues, both verbally and through the submission of feedback slips. Discussion leaders and appointed note takers rotated among four small groups.

Session 5

Scribes for each key subject summarized the findings from the workshop. Following the workshop, comments were integrated into the curriculum development and posted on a website administered by Edmonds Community College (www.mpdc.biz).

B.5 CACRC ATTENDANCE (PRESCOTT, SCOTLAND) (MAY 2006).

1.   Separately, but directly related to the course development, the CACRC held discussions on how to better serve the aviation industry.

2.   The CACRC identified the following major research needs:

- Substantiation of analysis methods

- Wet lay-up properties

- Inspection of composites (performance assessment and improvements), including detection of barely visible impact damage

- Effect of moisture on repair strength

- Inspection of repairs (measuring and assessing effects of porosity)

- Effect of training on strength of repairs

- NDI quantification of bondline strength

- Test methods for establishing material allowables

- Feedback from CACRC concerning awareness training

3.   CACRC was primarily concerned that students clearly understand that the course being developed in this cooperative agreement would not qualify them as practitioners.

4.   CACRC was presented with two concepts in relation to this course:

- Publish the teaching points in this report to provide industry support for the awareness class and a checklist by which future curriculum developers can measure the effectiveness of the class. It was agreed that these points would be considered at the next meeting.

- The need for a generic SRM as an important tool that can be published in the public domain without violating copyright or other proprietary concerns by OEMs. While many of the CACRC privately supported this concept, considerable concern existed for the source of funding for this project.

B.6 BETA CLASS (MAY AND JUNE 2006).

To further provide an opportunity for course feedback from industry and to evaluate potential application of online-learning formats, a beta class was conducted at Edmonds Community College. The goal was to provide three days of content instruction, using online computer resources, but with the students assembled in a classroom.