

TCO D Module – Recognize Composite Damage Types and Sources

D1: Identify Sources and Characteristics of Damage to Composite Sandwich and Laminate Stiffened Structures

Sources of damages to composite structures can be roughly sorted into three types:

Processing anomalies and in-process handling damages

Incorrect processing can result in defects in composite parts, and carelessness during post-cure handling and assembly can also result in damages. The anomalies resulting from these kinds of processing mistakes are usually discovered by rigorous inspections by the OEM; however some can go undetected and may show up during routine maintenance. Similar mistakes during 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 ingress.

a) 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, or loss of vacuum.

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 layup, 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.

b) 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.

c) Inclusions can occur when insufficient care is taken during ply layup. Objects such as pencils have shown up as a result of post-cure inspection. Other items found in cured parts have been separator film and other cure aids, left there by careless technicians and undetected by sloppy in-process inspection.

Post-processing inspection will detect the vast majority of manufacturing anomalies, and for those that are detected, liaison dispositions will be made prior to part delivery. These dispositions may be repairs or defect acceptance. In any event that the defects were determined to be acceptable, it is essential that these liaison records are kept available.

In the maintenance arena, inspection techniques are currently not, in general, 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 that anomalies are kept to a minimum. There are ongoing efforts to bring more affordable portable sophisticated inspection techniques to operators and maintenance providers so that both damage assessments and post-repair 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 Boeing B787.

In-service damages

- a) 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, and debris thrown up during take-off and landing (FOD). Damages from events such as these are considered the most important in-service damages to detect. Damages from the above sources can range from minor to critical to flight safety, therefore it is essential to be able to detect them before flight loads are imposed on the damaged structures.
- b) Solvents and other fluids can be absorbed by composites causing degradation of mechanical properties. Aircraft parts can be contacted by all manner of different fluids: grease, fuels, oils, hydraulic fluids, water, cleaning and de-icing fluids and salt spray. Property reductions due to fluid or moisture absorption into otherwise undamaged composite components are typically taken into account during part design, therefore repair is not usually required.
- c) Structural components located near engines or sources of aerodynamic noise are susceptible to sonic fatigue. Examples of composite components that may be subject to sonic fatigue are engine cowl, duct and strut components. Others can include 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 is dependent on the actual structural configuration, analysis/test programs are performed to validate the designs. In the event that high frequency noise produced by propulsion units and aerodynamic disturbance is higher than designed for, damage such as loosened or broken fasteners, disbonds, delaminations and through thickness cracking emanating from attachment details may result.

d) High heat sources can affect composite parts. Examples of high heat sources are: thermal de-icing ducts (typically located in the leading edges of wings), power plants and auxiliary power units (APU), hot air feed ducts, air-conditioning units and hot air duct failures. As in the discussion above, composite parts that are expected to be exposed to high heat sources are designed for this exposure. For example Boeing aircraft wing fixed leading edge panels are fabricated with materials cured at 350°F, while the wing fixed trailing edge panels 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 compromised, but the epoxy resin may burn resulting in exposure of the fibers, and cracking that can provide moisture or fluid ingress paths. Apart from obvious burn damage, discoloration of the part finish may give an indication of a high temperature exposure.

Environmental damages

Hail, lightning strikes, ultraviolet (UV) radiation, high intensity radiated fields (HIRF), rain erosion, moisture ingress and ground-air-ground cycles (temperature, pressure and moisture excursions) can all cause damages to composite components.

a) Ground hail can seriously damage sandwich components which have relatively thin facesheets. A severe storm at the Dallas-Fort Worth 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 judged to have been between 240 and 360 inch-pounds (in-lb). The level of energy generated by impact of the 2 inch diameter hail balls during this storm, also resulted in denting of metal fuselage skins. Some of the aircraft damaged by this hail storm were not returned to service for two weeks while on-airplane repairs were performed. As a result of this severe storm and other more recent ones, cost-benefit studies were performed at Boeing and other OEMs, trading increases in operating costs due higher structural weight against lost revenue for down-time to perform structural repairs. It was decided that interchangeable parts (flight control components) were acceptable as they were, because of the ease of replacement. Due to the lost revenue caused by on-airplane repairs, minimum sandwich facesheet gages were established for permanently attached composite structural parts.

b) Lightning strikes can inflict severe damage to composite components unless protection systems are employed. Composite materials are either not conductive at all, or are significantly less conductive than aluminum. Unless protected, composite structural components will suffer more damage due to lightning strikes than counterpart aluminum structures. Also, composite materials allow significant portions of lightning current to flow (arc) into onboard systems and provide less shielding of onboard electronic systems than do metal parts. The materials, design configuration and interfaces necessary to achieve the necessary damage resistance for a composite structural component exposed to lightning is dependent on the specific 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. Composites used on

components in high intensity zones must be protected. Extensive damage can be prevented by 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 all the way from surface ply burns to complete laminate burn through. 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.

c) While ultraviolet radiation has little effect on carbon fibers, it can degrade epoxy resins and as a result the integrity of the composite can be compromised. Ultraviolet radiation can cause surface embrittlement of unprotected polymeric composite material. Although carbon fibers restrict the penetration of UV, the long-term exposure to UV radiation inevitably leads to the formation of a degraded surface which may serve as a site for brittle crack initiation. If composite parts are subject to UV radiation they are typically protected by an opaque finish layer which must be restored after repair.

d) Damaged composite parts can ingress fluids from surrounding environments. The ingress 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 ingress into the core. In the case of 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 can also provide paths for moisture ingress.

D2: Describe Damage Types and their Significance to Structural Integrity

The differing types of damage in a composite component 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 inspection techniques such as ultrasonic methods. While ultrasonic NDI methods can map out delaminated areas, and visual inspections can usually determine the extent of through thickness cracks; such damage as matrix crack, 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 following discusses the various types of damage that can occur, the failure modes associated with these damages, and the assumptions required when performing an allowed damage assessment.

Matrix Imperfections (Cracks, porosity, blisters, etc.)

These 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 cracks, a.k.a. micro-cracks, can significantly reduce properties dependent on the resin or the fiber/resin interface, such as inter-laminar shear and compression stiffness and strength. For high temperature resins, micro-cracking can have a very negative effect on properties due to lost oxidative stability associated with increased surface areas. Matrix imperfections may develop during service into delaminations, which are a more critical type of damage. Porosity usually occurs during the fabrication cycle, whereas blisters and micro-cracks 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 ingress especially for sandwich parts with thin facesheets.

Delaminations

Delaminations typically form at the interface between the layers in the laminate, along the bondline between two elements, and between face sheets 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, geometrical discontinuities, etc.
- 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.

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 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.

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. 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). Unlike metals, which can fatigue crack due to fairly low level loading that occur constantly in normal operations because of the presence of an anomaly (a small imperfection in 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.

Nicks, Scratches, Gouges

These damages 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 then they must be treated as a crack in the affected plies. The overall effect depends on the length and depth of this type of damage as related to the total laminate or facesheet thickness.

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 face sheets and core. Dents in thin face sheets over core areas often only involve core damage. Dents in solid laminate areas fastened to substructure (e.g., edgeband areas of sandwich panels) can have associated damage to the substructure. In cases such as these, if accessible, it is essential to examine the backside of the part and any adjoining substructure.

Punctures

A puncture is defined as an impact damage that causes a penetration of 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. In either case, there may be associated delaminations, matrix damage and fiber breakage outside of the puncture edges.

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 (i.e. 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 have been detected visually will yield more extensive damage when inspected by NDI techniques.

It is essential to understand that any visual indication of damage to a composite component needs to be further investigated. Knowledge of the structural configuration of

the damage part will provide clues as to potential non-visible damage, such as that to an adjacent stiffener or rib, a disbanded sub-structural element or crushed sandwich core.

Some experimental evidence suggests that, for relatively small damage sizes, impact damage is more critical than other defects. Some test results for panels containing damages greater than 2 inches indicate that large holes or penetrations are at least as severe as equivalent sizes of impact damage.

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 reduction in in-plane bypass, fastener pull-through and bearing strengths.

Edge Erosion

Damage to the edge of a panel laminate due to the effects of air flow over the structure, and the impingement of debris, rain, etc. on the laminate, resulting in erosion of the laminate material both through the thickness, and in the direction parallel to the air flow. This can expose the fibers leading to potential fiber microbuckling and a path for moisture ingress.

D3: Understand the information and analysis necessary for repair design and process development/substantiation

Repairs performed while an aircraft is in service are controlled by the Federal Aviation Regulations (FARs) and Advisory Circulars (ACs) devoted to the maintenance of civil aircraft. Of particular interest are FAR, Part 43 Maintenance, Preventive Maintenance, Rebuilding, and Alterations; and FAR, Part 145 Repair Stations.

FAR, Part 43 and the ACs associated with Part 43 specify methods which have been 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 FAR, 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.

As stated in C1 the intent of repairs to structural components is to restore the original mechanical properties, such as strength and stiffness, regardless of the component loadings and regardless of whether the repair is to be bonded or bolted. This requires the use of the strength and stiffness data for the original material and repair materials, and strength and stiffness data for the repair fasteners. Original part strength and stiffness data used for repair designs is derived from the database used for the original type certificate.

Variability, for both the original component materials and the repair materials, is accounted for in the material allowables through a 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 MIL-HDBK-17 that may be used to design a repair. If these data are used by an operator or MRO to design a repair, the repair design must be approved by a DER.

Nearly every repair consists of a patch of some kind and a joint through which the loads are transferred to the patch. Joints are basically either bonded or bolted or, sometimes a combination of both. Analytical procedures for bolted or bonded joints tend to be complex and can require computer programs for their solution.

Analysis procedures for bolted and bonded repair joints are described in MIL-HDBK-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 OEM SRM are followed.

To ensure that required standards are met, all materials and all processes used for the repair of aircraft structural components must conform to approved specifications. This means that sufficient research and testing have been performed to ensure that the materials and processes employed by the end user will; if the materials have been 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. In the case of Boeing SRMs, these specifications are BMS numbers (e.g. **BMS 8-256 for 350°F cure carbon fabric prepreg and BMS 8-168 for 250°F cure carbon prepreg.**). Process instructions in the SRM are usually laid out in detail step by step..

Approval of a repair to an aircraft structural component requires that:

- a) The repair was designed by a qualified engineer who used approved data to restore the part original strength and stiffness capabilities. Repair design criteria for permanent repairs are fundamentally those that were 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 to on the aircraft for a considerable time (e.g. an interim repair).

- b) The repair materials have been stored and handled within the bounds set down in the approved documentation. The use of improperly stored or handled adhesives, sealants, prepregs or wet layup ingredients may result in structurally unsafe aircraft components.
- b) The repair has been 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.
- c) The material and in-process records and the post-repair inspection have been judged acceptable. Once a repair is completed, it must be inspected and approved by the FAA or its representative (DER) before the part is returned to service.

D4: Distinguish differences in repair disposition procedures for those damages covered by source documentation, and those that aren't

When damage has been discovered and mapped, the first step in any damage/repair disposition is to consult the source documentation (such as the SRM) for ADLs and repair designs.

If the damage is less 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. 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 and any protection system.

If the damage is larger than the ADL and there is an approved repair available, the procedures set down for damage clean-up, moisture removal, surface preparation, repair processing and inspection should be followed precisely.

Typically the SRM will contain a number of repair designs for specific components. For primary structural elements (PSE) designated structures the similar 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 the repair cure temperatures. For example:

- a) Repairs cured at room temperature or 150°F will be typically classed as temporary or time-limited with small repair size limits.
- b) Wet layup repairs cured under vacuum at 200°F, or repairs using prepreps cured under vacuum at 250°F or higher, will usually be classified permanent. Larger size limits will be allowed for these repairs.
- c) Repairs using the original component material prepreg and cure processing parameters are classified as permanent. Within the Boeing SRM, for sandwich structural components that are easily removed from the aircraft, there is typically no size limit on these repairs except in critical areas such as highly loaded areas (e.g. local to fittings). For example, in the SRM repair

instructions for the 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 are: “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 autoclaves typically used by operators or MROs.

If there is no approved repair available for the specific damaged component, either the maintenance engineer contacts the OEM for an approved repair, or he or she designs a specific repair and obtains design approval from a DER. In the event of repair limits established by operators or MROs, these must be approved by the OEM, 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 cases such as these, 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 this repair must be obtained from the OEM..

D5: 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:

- a) **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 the damage is to a stiffened skin panel, inspection of the backside 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 that is 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. In the event

that this 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 ensue unless there are significant delays in obtaining either materials or the approved design from the OEM.

- b) 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 then be shipped to operator's maintenance depot, or to an MRO for a permanent repair. This repaired part can be then put into storage for redeployment at a convenient time, or stored until needed to replace a damaged part on 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 parts, or re-manufactured or after-market parts. Any re-manufactured or after-market parts that are used must be approved by the delegated authority.

- c) Prepare a specific repair for the damage not covered.** Repair schemes may require one or more drawings 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 for the repair. 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 that the appropriate planning action can be taken. In some instances, such as damage to a PSE 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 of the above options 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 are eventually incorporated into the SRM for specific components. This process may take a while, but 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 gestation time for OEM approved non-SRM repair designs can be, even if coordination is expedited, two or more days before the repair designs are provided and actual repairs performed.

D6 [LAB #1]: Damage laminate coupons in a controlled laboratory environment and visually inspect the extent of the front and any back side surface damage

Laboratory #1 equipment list:

Drop tower

10 LB tupp

Support frame

Four 2.0 LB weight bags

Flashlight

Magnifying glass

Black marker pen

Twelve 15" by 15" eight-ply laminate test panels painted with grey primer and grey enamel on front-side, painted with yellow primer on back-side

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.

When the instructor is satisfied with the damage state and has calibrated the drop height and tupp weight, the students will pair up to damage eight panels as follows:

Damage Instructions:

- Step 1 Place laminate test panel flat on support frame with the panel center (marked by X) directly under drop tower tube
- Step 2 Insert the Tupp into the drop tower tube.
- Step 3 Place the weight into the drop tower, raise the weight to a height of NN¹ inches, then set the locking mechanism.
- Step 4 Ensure that one student is in position to activate the catching mechanism that will stop the weight before it can rebound onto the specimen.
- Step 5 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 can rebound onto the specimen a second time.
- Step 6 Place weight bags on each corner of test panel
- Step 7 Remove weight bags from test panel
- Step 8 Remove damaged test panel from impact apparatus

¹ 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.

When the students have damaged all eight 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:

Visual Inspection Instructions:

- Step 9 Inspect the front surface of the test panel visually using flashlight and magnifying glass to inspect for damage.

- Step 10 Mark the perimeter of the damage area on the front surface of the panel with a marker pen

- Step 11 Inspect the back surface of the test panel visually using flashlight and magnifying glass

- Step 12 Mark the perimeter of any visible damage area on the back surface of the panel with a marker pen.