Composite Bending Box section Modal Vibration fault Detection

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Abstract:
One of the primary concerns with Composite construction in critical structures such as wings and stabilizers is that hidden faults and cracks can develop operationally. In the real world, catastrophic sudden failure can result from these undetected faults in composite structures. Vibration data incorporating a broad frequency modal approach, could detect significant changes prior to failure.
The purpose of this report is to investigate the usefulness of Frequency mode testing before and after bending and torsion loading on a composite bending Box Test section. This test article is representative of construction techniques being developed for the recent NASA Blended Wing Body Low Speed Vehicle Project.. The Box section represents the construction technique on the proposed blended wing aircraft. Modal testing using an impact hammer provides an 'frequency fingerprint' before and after bending and torsional loading. If a significant structural discontinuity develops, the vibration response is expected to change. The limitations of the data will be evaluated for future use as a non-destructive in-suito method of assessing hidden damage in similarly constructed composite wing assemblies. Modal vibration fault detection sensitivity to band-width, location and axis will be investigated. Do the sensor accelerometers need to be near the fault and or in the same axis? The response data used in this report was recorded at 17 locations using tri-axial accelerometers. The modal tests were conducted following 5 independent loading conditions before load to failure and 2 following load to failure over a period of 6 weeks. Redundant data was used to minimize effects from uncontrolled variables which could lead to incorrect interpretations. It will be shown that vibrational modes detected failure at many locations when skin de-bonding failures occurred near the center section. Important considerations are the axis selected and frequency range .

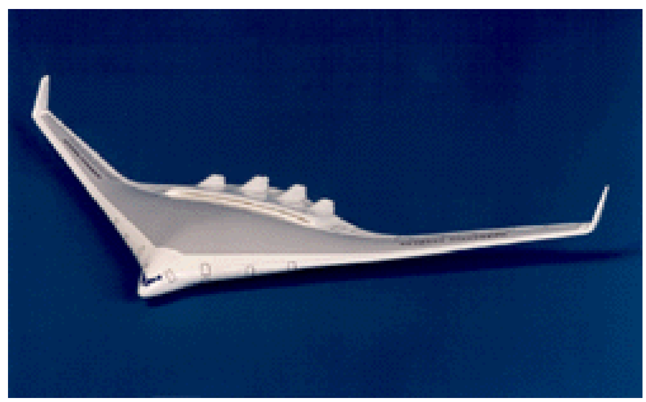


Photo 1 Model of the Blended Wing Body proposed Air-transport

Background:

Shown above in Photo 1 is the inspiration for the composite box engineering development section, The Proposed blended-wing-body (BWB) Airline configuration is a very large subsonic transport with a design payload of 800 passengers, a 7000-n.mi. Range and a cruise Mach number of 0.85. The test section is a demonstration of the wing construction. Structural flaw detection in the composite material using practical non-destructive methods will be required if such an aircraft is built, hence the use of Modal vibration as a potential tool.

Technique:
The testing described here-in is to provide a fairly broad modal picture with application to the real world, not conduct a detailed survey of all possible modes of vibration and their shapes. The Data was displayed near real time and saved to disk for later analysis.
Hardware used was a Gateway Laptop Computer with PCMIA interface card with Spectral Dynamics Bobcat 8 channel Data Acquisition unit , version 4.0 Software, force input using a Modal Hammer (PCB #C086C05) with force gage impacting in the vertical direction on the Aluminum end. Response was recorded using 6 channels of Tri-axial Accelerometers (PCB # 339801), data was collected from 3 axis and two locations at a time. Frequency response was 1-1000 hertz (hz) with 0.3 hz resolution. The accelerometers were temporally mounted using wax to allow easy removal and good frequency response. Total testing was for 5 independent loading conditions before load to failure and twice following load to failure over a period of 6 weeks. See Table One.
Three input hammer hits were averaged for each datasheet.
Locations 1-7 were along the Center Spar with Locations 8-9 and 10-11 placed on the left and right Spars to collect torsion responses with modal hammer hits applied on the right Spar.
Locations 12-17 were each centered on the Six 15 by 20 inch Unsupported areas s to collect panel responses.

Table 1 applied loads and modal testing

Date	Test number	Prior applied Load Description
7/2	1Axx	Initial Modal Test prior to load testing
7/10	2Axx	40 LB torsion and bending loading
7/17	3Axx	170 LB torsion and bending loading
8/8	4Axx	Repeat 170 LB torsion and bending loading without bracket
8/15	5Axx	Same as 4 with bracket
8/17	**6Axx**	**bending load to failure (899 LB) with bracket**
8/17	**7Axx**	**Same as test 6 without bracket**

xx = response accelerometer locations
The response accelerometer locations were selected to provide the following objectives:

Table 2 Response Accelerometer Objectives

Mode type	location numbers
Main spar Bending	1-7
Spar Torsion	8-11
Panels	12-17

A photo of the test setup is shown below (Photo 2)

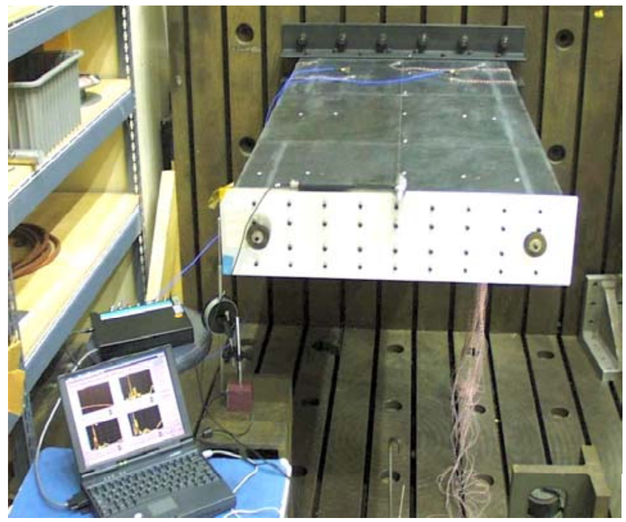


Photo 2 Test –setup showing Cantilevered Bending Box and Data Acquisition

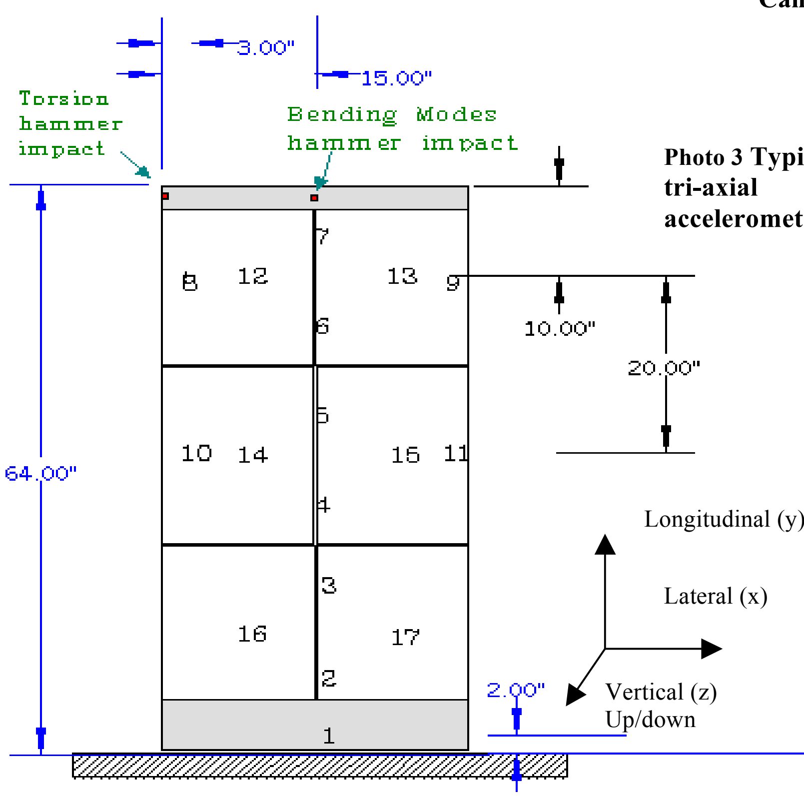


Photo 3 Typical tri-axial accelerometer

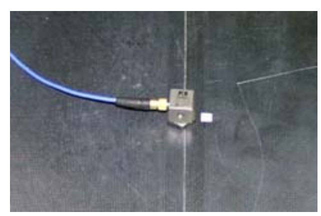


Figure 1 Bending Box force input and response accelerometer layout

The modal tests were performed following static loading on The bending box with vertical force applied with a hydraulic cylinder to the clevis bracket an up (+Z) direction bending force in Photo 4. Photo 5 shows a close-up of bracket and clevis. The torsional load was applied using pulleys and dead weights (Photo 6).

Photo 4 Bending loading with hydraulic cylinder

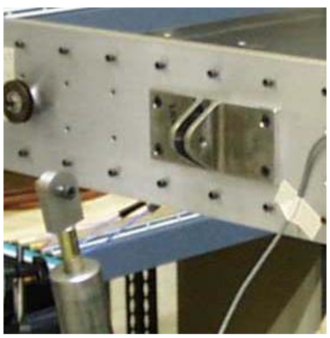


Photo 5 Cylinder and Bracket Clevis

The Test Beam Box sections consisted of Phenolic honeycomb cores with carbon cloth bonded to both sides. The design is to minimize weight and maximize strength and stiffness. Shown below in Photo 7 is the cross-section of the Main center Spar: The Carbon cloth skin was bonded to the Honeycomb with Shell Epon 901® adhesive

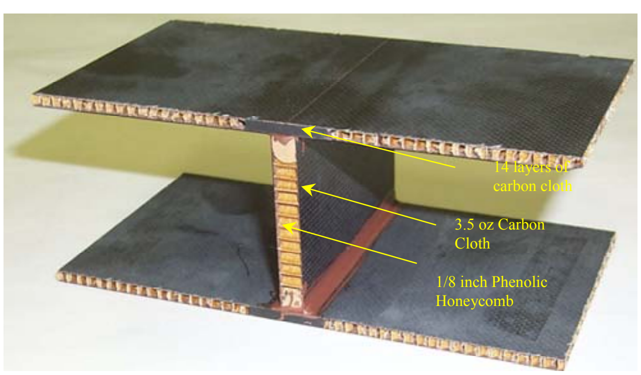


Photo 7 Composite test beam cross-section

Photo 6 Torsional loading on dummy beam using Pulleys and weights

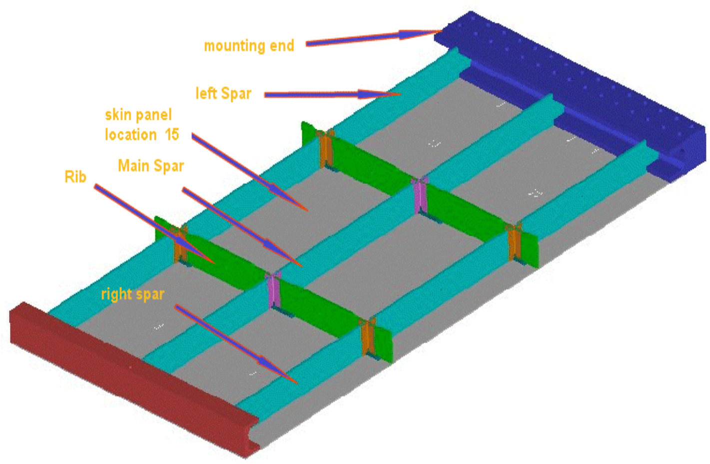
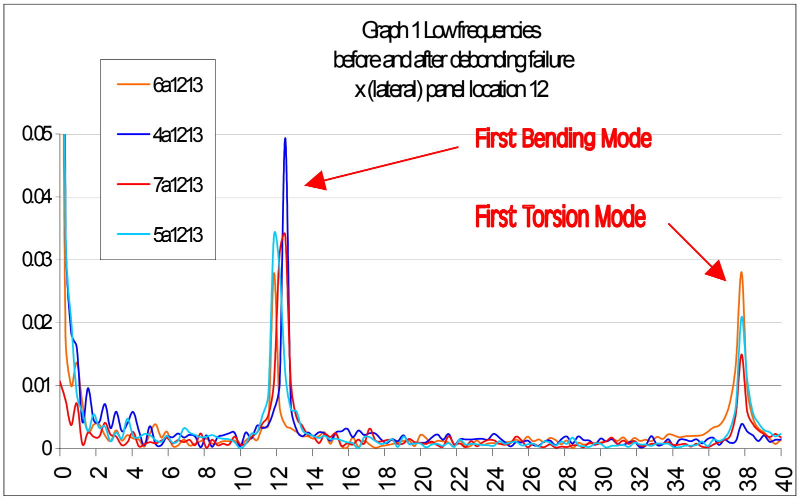
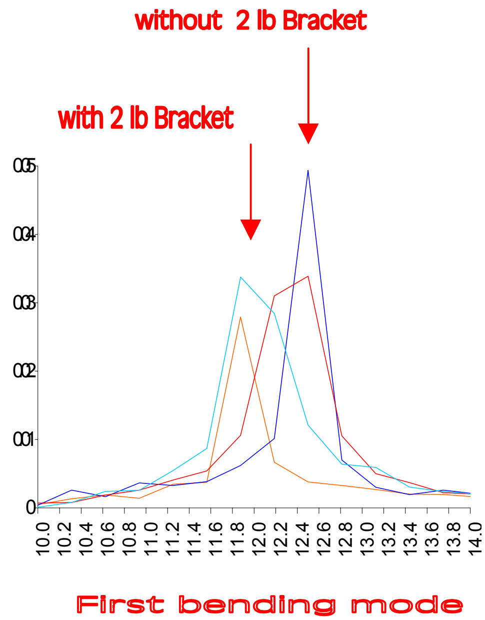


Figure 2 Inside cutaway showing structure

Figure 2 above shows the inner geometry with the top skin removed, using ribs, spars and panels made from honeycomb Phenolic with carbon cloth bonded to each side, joints use 14 layers of carbon cloth and adhesive. Ends are bonded Aluminum pieces for loading and cantilevered attachment on the mounting end to a backstop. Total weight is 53 pounds with 13.2 pounds for the composite bending box

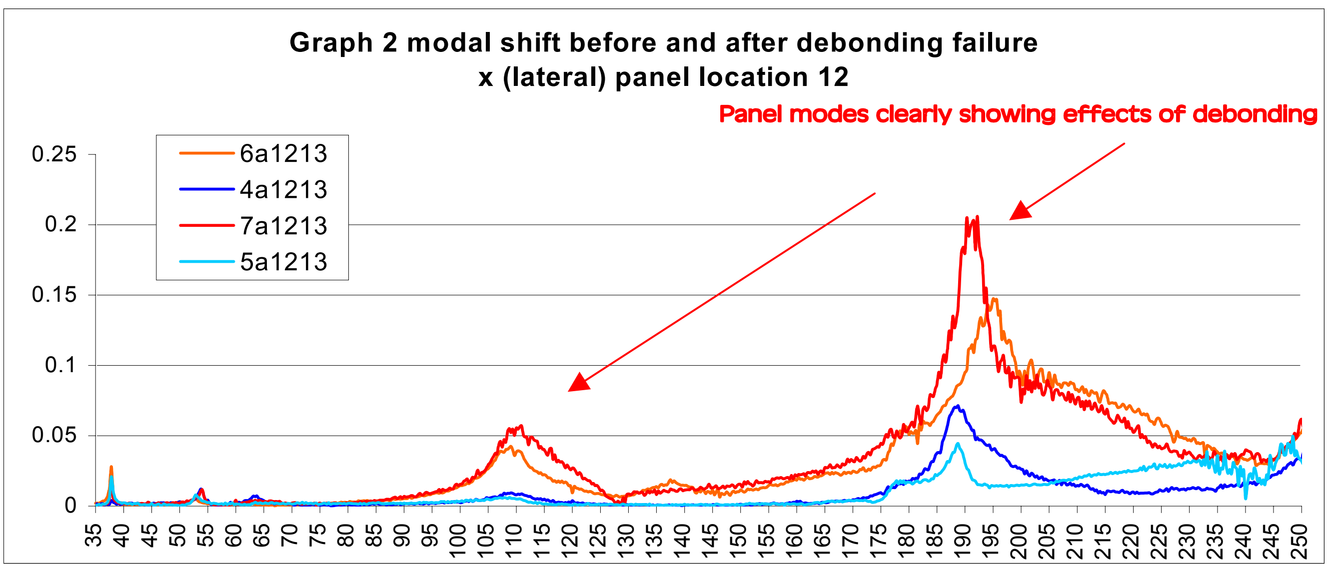
Data and Discussion: Graphs 1-9 vertical scale is the magnitude of the Frequency Response Function (FRF) and the horizontal scale is the Frequency in hz.





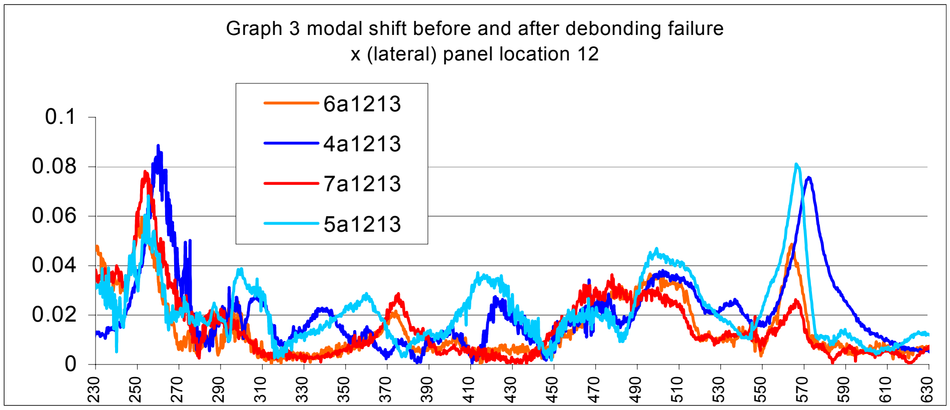
Comparison of first bending and Torsion modes before and after final loading to Failure is shown in graphs 1 through 3. Tests 4a and 5a (with clevis bracket) are prior to loading to failure. Tests 6a (with clevis bracket) and Test 7a are following the de-bonding failure.
The de-bonding failures appeared clearly by the Panel modes at higher frequencies in the lateral (x) directions as

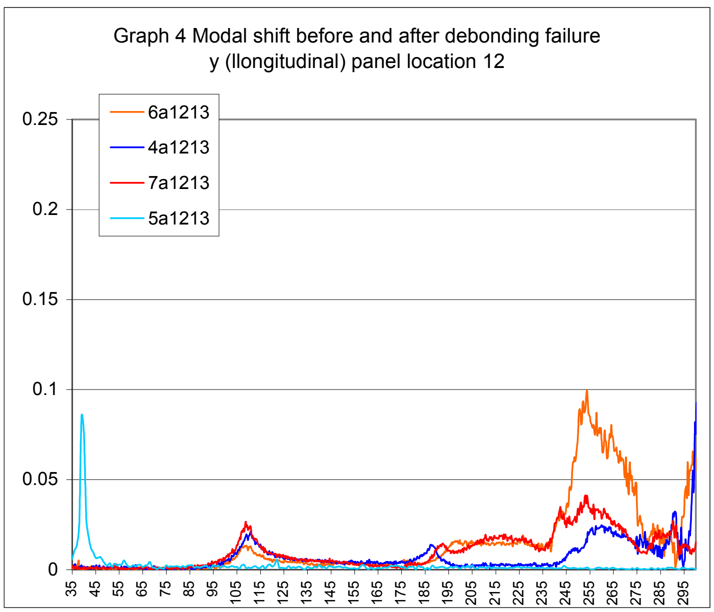
The detail above shows that you can not tell from the First bending mode that a de-bonding failure has occurred since The spars and ribs have not failed. The sensitivity of this mode to the addition of the 2-lb bracket is easily seen by the lowering of the bending frequency by about ¾ hertz.



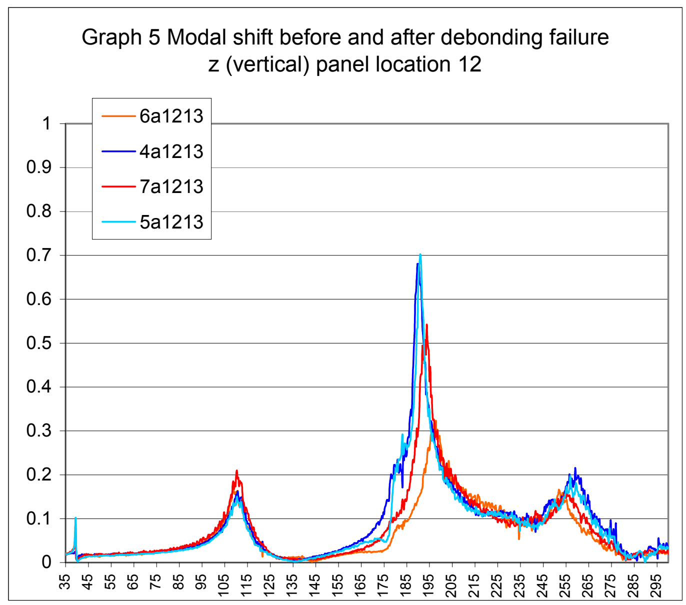
Shown in Graph 2. The very different magnitudes around 110 and 190 hz indicate a decrease in stiffness is this plane due to the carbon skin de-bonding.

At even higher frequencies the effects of de-bonding on magnitude is much less clear as shown in Graph 3 below.



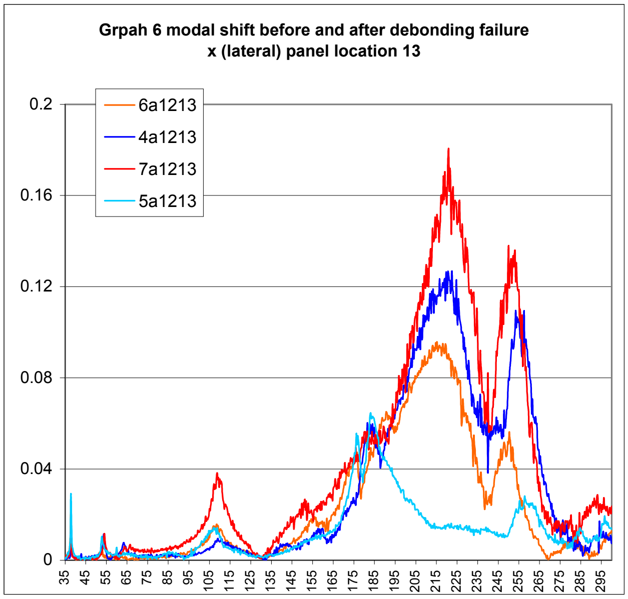


Graph 4 shows that In the y longitudinal direction the de-bonding failure and stiffness change are less apparent but still detectable.

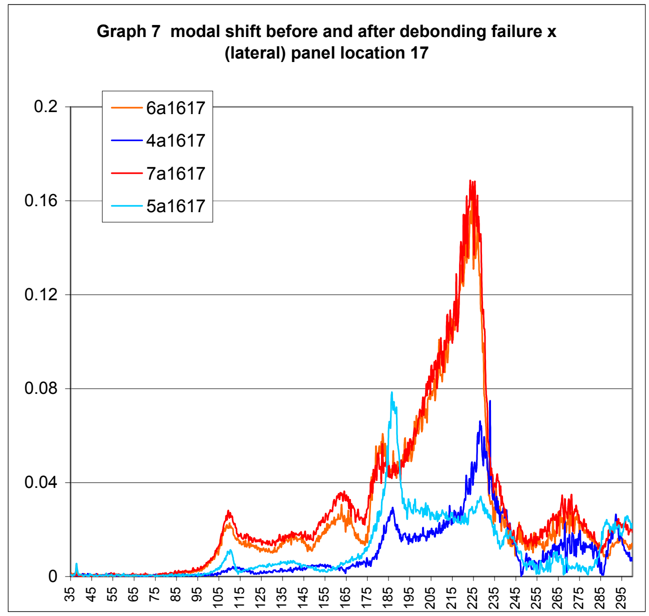


Graph 5 shows comparisons on the pre to post de-bonding failures in the vertical (z) direction show little difference. From this subset of the reviewed data it is apparent that to detect this type of failure where the main support system is still functional (load carrying) monitoring must be done in the correct direction and frequency range, else failures could go undetected.

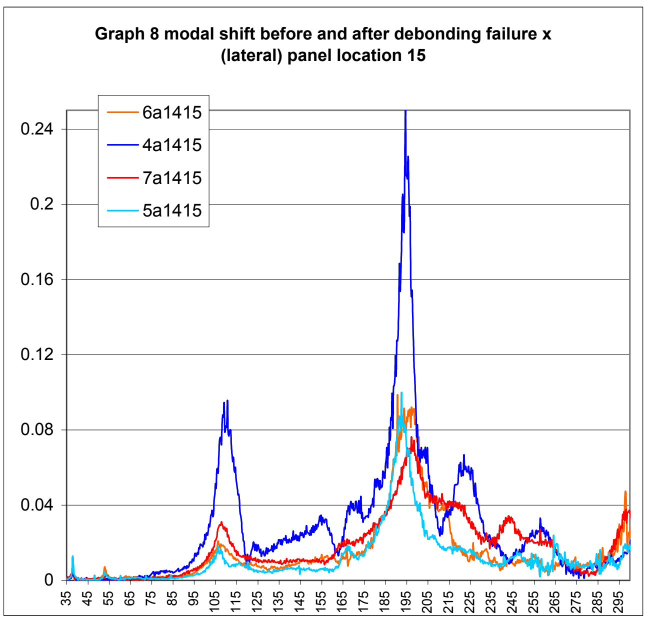
Response location is not critical if consideration is given to the underlying support structure. The panel centers which are as far away from the ribs and spars as possible provide the best indicators for the skin de-bonds. The Panels are where the failures are manifested with reduced lateral stiffness. Any of the Panel's locations 12 –17 provide similar indications of the failures (Graphs 6-9).



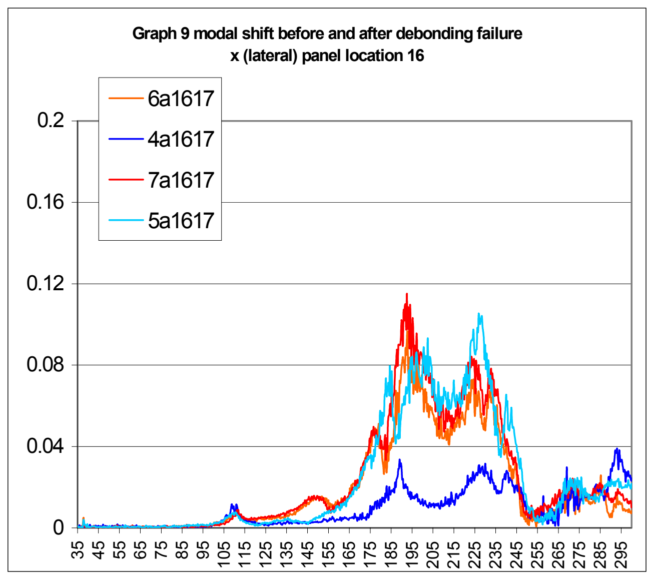
Graph 6 shows panel location 13 near free end



Graph 7 is panel location 17 near the mounted end



Graph 8 shows panel location 15 near the mid-section



Graph 9 is panel location 16 near the mounted end

Photo 8 Carbon skin de-bond on the Main Spar side

Photo 9 Close-up carbon skin de-bond

Shown above in Photos 8 and 9 is the skin de-bonding in the main Spar side also near location 15 on the panels caused from the load to failure test.

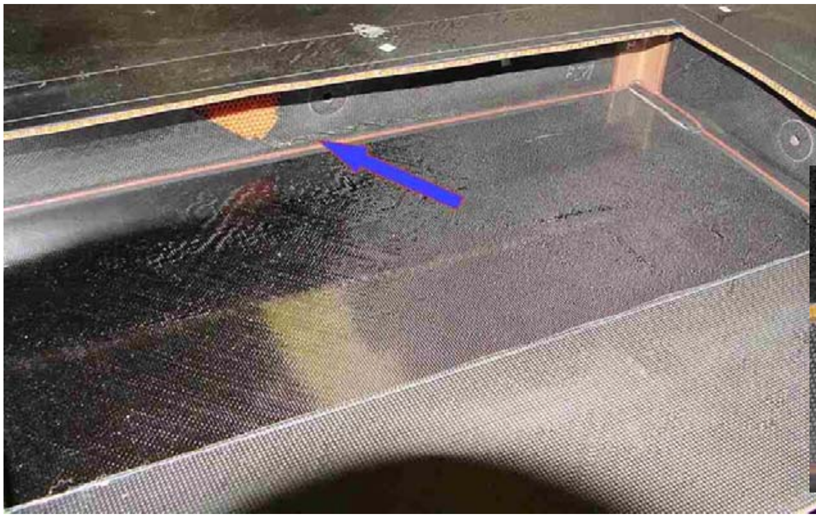


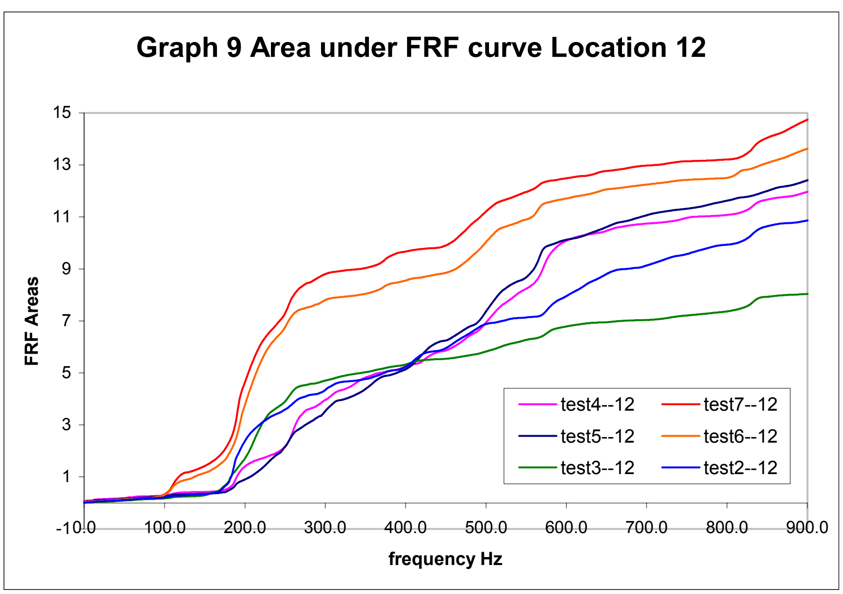
Photo 10 cutaway section near top surface skin de-bond

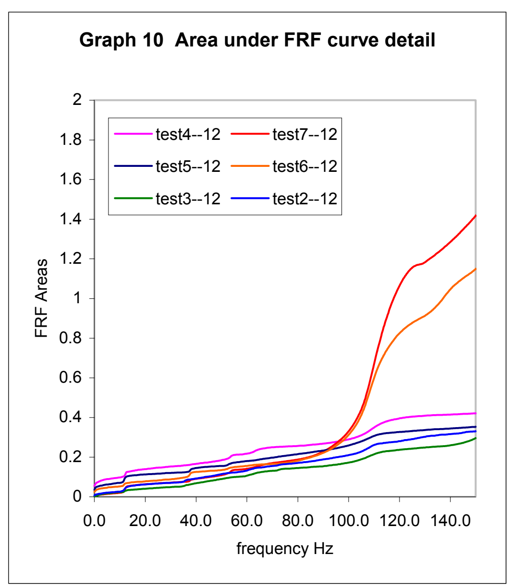
Photo 11 Close-up near top surface skin de-bond

Photos 10 and 11 Show the skin de-bond in the top of the main spar also near panel location 15 after loading to failure

The Graphs above indicate that Skin de-bonds were detectable but the problem remains of the time and knowledge that is required to gather and carefully analyze the data form the Modal testing. Therefore, a quicker evaluation that could be programmed in software would be useful. By plotting the running total areas under the FRF magnitude curves from the FRF data sets, a fail/no fail numeric evaluation is possible.

Graphs 9 and 10 below show the jump in area for the Post failure modal test 6 and 7 when compared to the pre – Failure test 2-5. The frequency where the areas diverge is around 110 hz corresponding to the lowest panel modes. Frequencies below 100 hz which include the bending and torsional modes of the support structure provide no indication as expected.





While this data is only For test bending beam specimen, a similar approach could be applied for other composite sections.

Conclusion:

The use of modal vibration testing and analysis to detect composite box beam failures is viable for such failures such as de-bonding of the carbon skin if important considerations are followed:

- The response accelerometers must be located on responsive areas of the structure (to the type of failure to be measured)
- The Axis (plane) of the measurements must correspond to the failed plane,
- The frequency range must envelop the failed structure frequency range.

These requirements necessitate timely and costly analysis by skilled professionals. Flagging significant changes in the area under the FRF curves however, can be programmed with software and provide autonomous fail/not failed criteria if Sufficient knowledge about the monitored assembly is known. This means that an identical assembly should be lab tested by producing known faults and observing the failure chactertiics with modal testing backed up by Finite Element modeling. An FRF area change limit can then be programmed in software along with real-time FRF area calculations for an autonomous warning system for operational usage.

The Future:

Macro Fiber Piezo-Composite Sensors Designed , fabricated and characterized [1] at NASA Langley Research Center could be embedded in the composite assemblies to provide a in-use failure indication when combined with microprocessor based real time analysis programming . The self-generated sensor voltages would be digitized and , transformed to the frequency domain with the continuous area under the FRF magnitude curves checked against a preset value for flagging an composite f ailure. In addition the application of wireless low power transmission technology would enable practical real world monitoring, Further research at Langley is expected to be on development of such a practical engineered system with a low installed and maintenance cost, making composite sections in critical areas practical and safer.

Photo 11

Typical Macro Fiber Piezo-Composite Sensor /Actuator

References:

[1] Rudy J. Werlink , Robert G. Bryant, Dennis Manos *Macro Fiber Piezocomposite Actuator Poling Study* TM-2002-211434 NASA Langley Research Center Hampton, Virginia, February 2002.