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Composite Impact Dynamics Research at NASA LaRC -- A Review

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INTRODUCTION

The Landing and Impact Dynamics Branch of NASA Langley Research Center has been involved in impact dynamics research (Fig. 1) since the early 1970's. For the first ten years, the emphasis of the research was on metal aircraft structures in both the General Aviation Crash Dynamics Program (Refs. 1-13) and the Controlled Impact Demonstration (CID) Program, a transport aircraft program culminating in the controlled crash test of a Boeing 720 aircraft in 1984 (Refs. 14-16). Subsequent to the transport work, the emphasis has been on composite structures with efforts directed at understanding the behavior, responses, failure mechanisms, and general loads associated with the composite material systems under crash type loadings. Considerable work has been conducted to address the energy absorption characteristics (Refs. 17-20) and it indicates that composites can absorb as much if not considerably more energy than comparable aluminum structures. However, due to their brittle nature, attention must be given to proper geometry and designs to take advantage of the good energy absorbing properties while providing desired structural integrity. Achieving the desired new designs often requires an understanding of how more conventional designs behave under crash type loadings.

The purpose of this paper is to present a review of the composite impact dynamics research being conducted at NASA Langley Research Center. Examples are presented of experimental and analytical data to illustrate the activities in the four program elements of the composite research.

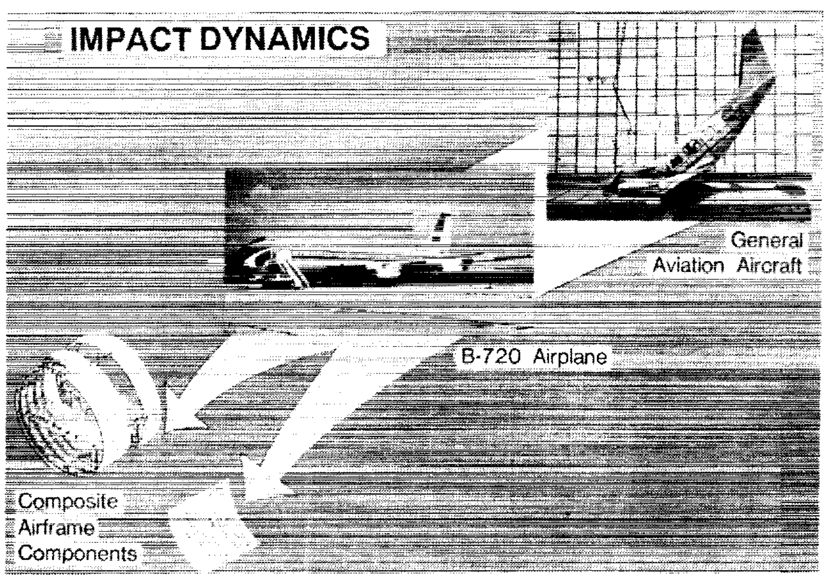

Figure 1

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IMPACT DYNAMICS RESEARCH FACILITY

The research was conducted by personnel at the Langley Impact Dynamics Research Facility (Fig. 2) using test equipment located at the installation. The Impact Dynamics Research Facility (IDRF), originally the Lunar Landing Facility used by the astronauts during the Apollo Program for simulation of lunar landings, has been modified to allow crash tests of full-scale aircraft under controlled conditions. The aircraft are swung by cables from an A-frame structure which is approximately 400 ft long and 230 ft high. The impact runway can be modified to simulate other ground crash environments, such as packed dirt, to meet a specific test requirement.

Each aircraft is suspended by cables from two pivot points 217 ft above the ground and allowed to swing pendulum-style into the ground. The swing cables are separated from the aircraft by pyrotechnics just prior to impact. The length of the swing cables determines the aircraft impact angle (from 0 degrees (level) to approximately 60 degrees). Impact velocities can be obtained up to 65 mph (governed by the pullback height). Variations of aircraft pitch, roll, and yaw can be varied by changes in the aircraft suspension harness attached to the swing cables. Data from onboard instrumentation are transmitted through an umbilical cable hard wired to the control room at the base of the A-frame. Photographic data are obtained by onboard, ground-mounted, and A-frame mounted cameras. Maximum allowable weight of the aircraft is 30,000 lb. Reference 21 provides complete details of the facility and test techniques for full-scale aircraft testing.

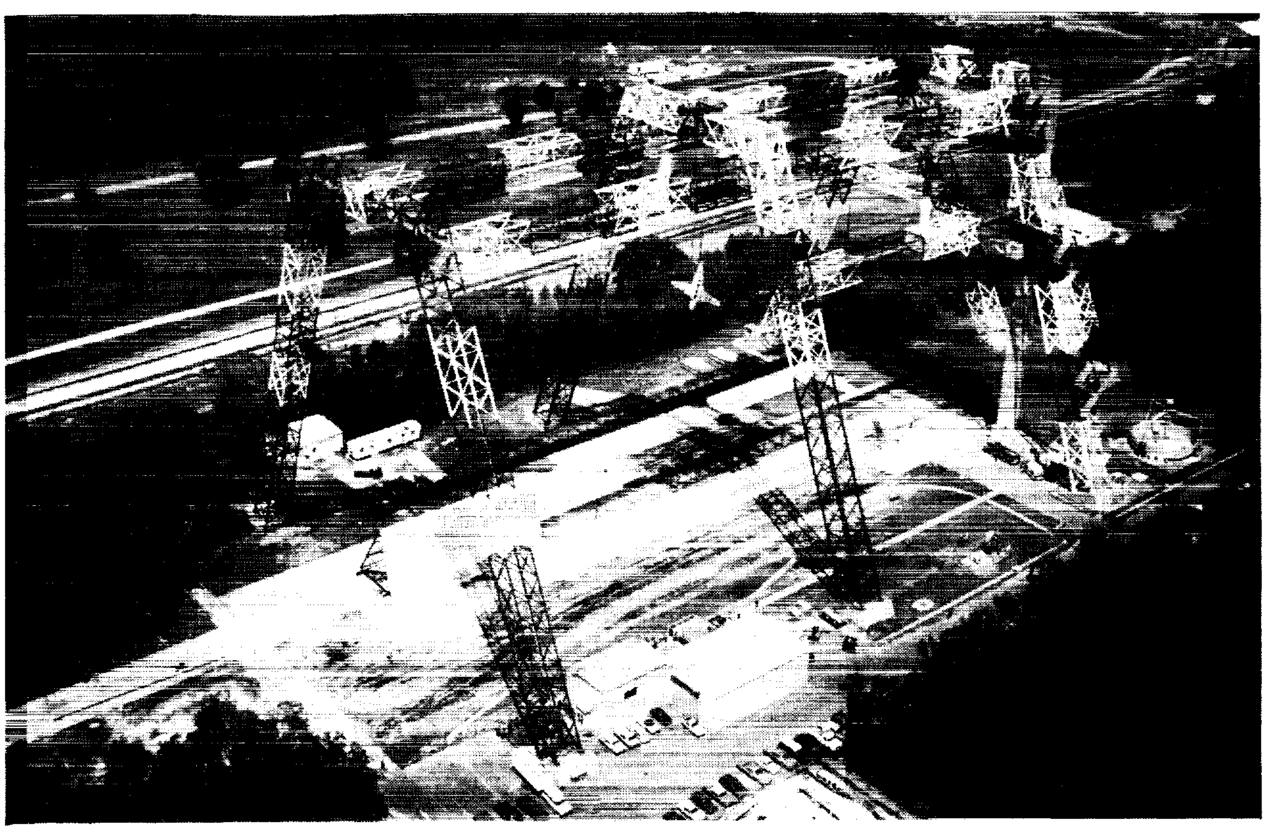


Figure 2

COMPOSITE IMPACT DYNAMICS RESEARCH PROGRAM ELEMENTS

The program elements of the Composite Impact Dynamics Research Program are illustrated in Fig. 3. Currently, efforts in crash dynamics research are in four areas: (1) development of a data base to understand the behavior, responses, failure mechanisms, and general loads associated with both conventional and innovative concepts using composite material systems under crash type loadings; (2) analytical studies/development relative to composite structures; (3) studies in scaling of composite structures under static and dynamic loads; and (4) full-scale tests of metal and composite structures to verify performance of structural concepts.

The overall goal of the research efforts is to gain a fundamental understanding of composite crash behavior and to formulate improved structural designs to meet performance, integrity, and energy absorption requirements. Examples of experimental testing relative to each program element will be highlighted in the paper. Analytical examples associated with the program elements will be presented at the appropriate time within the discussions of the experimental tests rather than in separate sections.

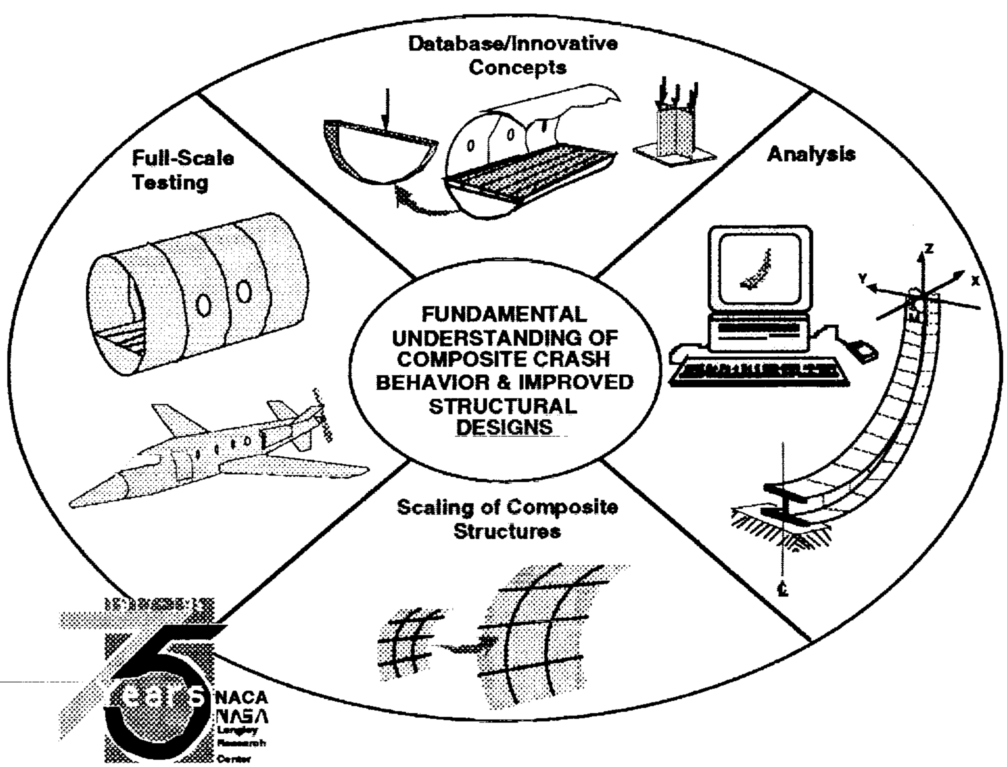


Figure 3

TYPICAL TRANSPORT FUSELAGE STRUCTURE

If one examines a typical transport fuselage structure as shown in Fig. 4, it becomes readily apparent that frames are one of the most important components used in the construction. As a consequence, the initial efforts in our studies were evaluation of individual frame components. The approach of studying simple structural elements and then moving to combinations of these elements in more complex substructures has been taken in the development of a data base on the dynamic response and behavior of composite aircraft structures. With this building block approach, more complex subfloor structures fabricated from the simpler components for both static and dynamic testing are discussed later in the paper.

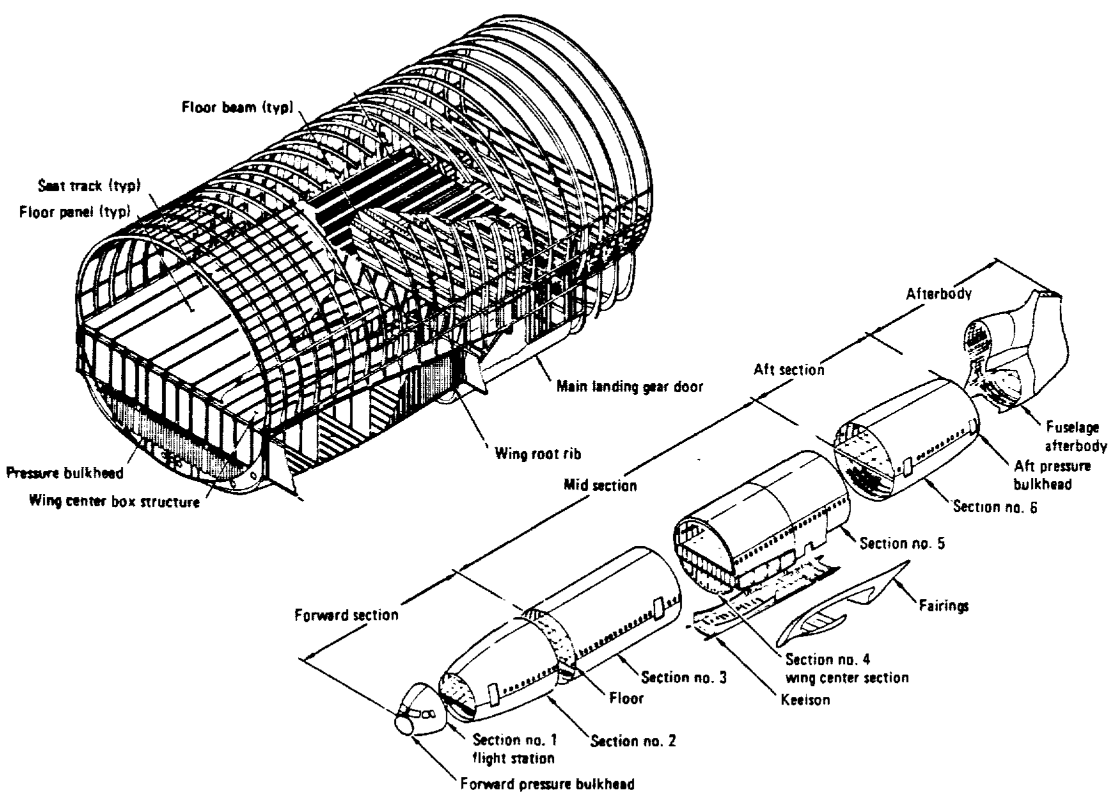


Figure 4

COMPOSITE FUSELAGE FRAME CONCEPTS

Various cross-sectional shapes for fuselage frames are used in metal aircraft and are often proposed for composite structures. Figure 5 presents sketches illustrating four of the more common geometries, I-, J-, C- and Z-cross sectional shapes. Several circular frames using these shapes were fabricated for testing. To add out-of-plane stability to the frames (with the exception of the Z-section frames), 3-1/2 inch wide skin material was added enhancing the ease of testing of both symmetrical and other nonsymmetrical sections. The skin, a [±45/0/90] $_{2S}$ lay-up sixteen ply (.08 inches) thick, was cocured with the 6 foot diameter frames. The frames were constructed in two different heights, 1-1/2 inches and 3/4 inches, to investigate the effect of frame height on behavior and responses.

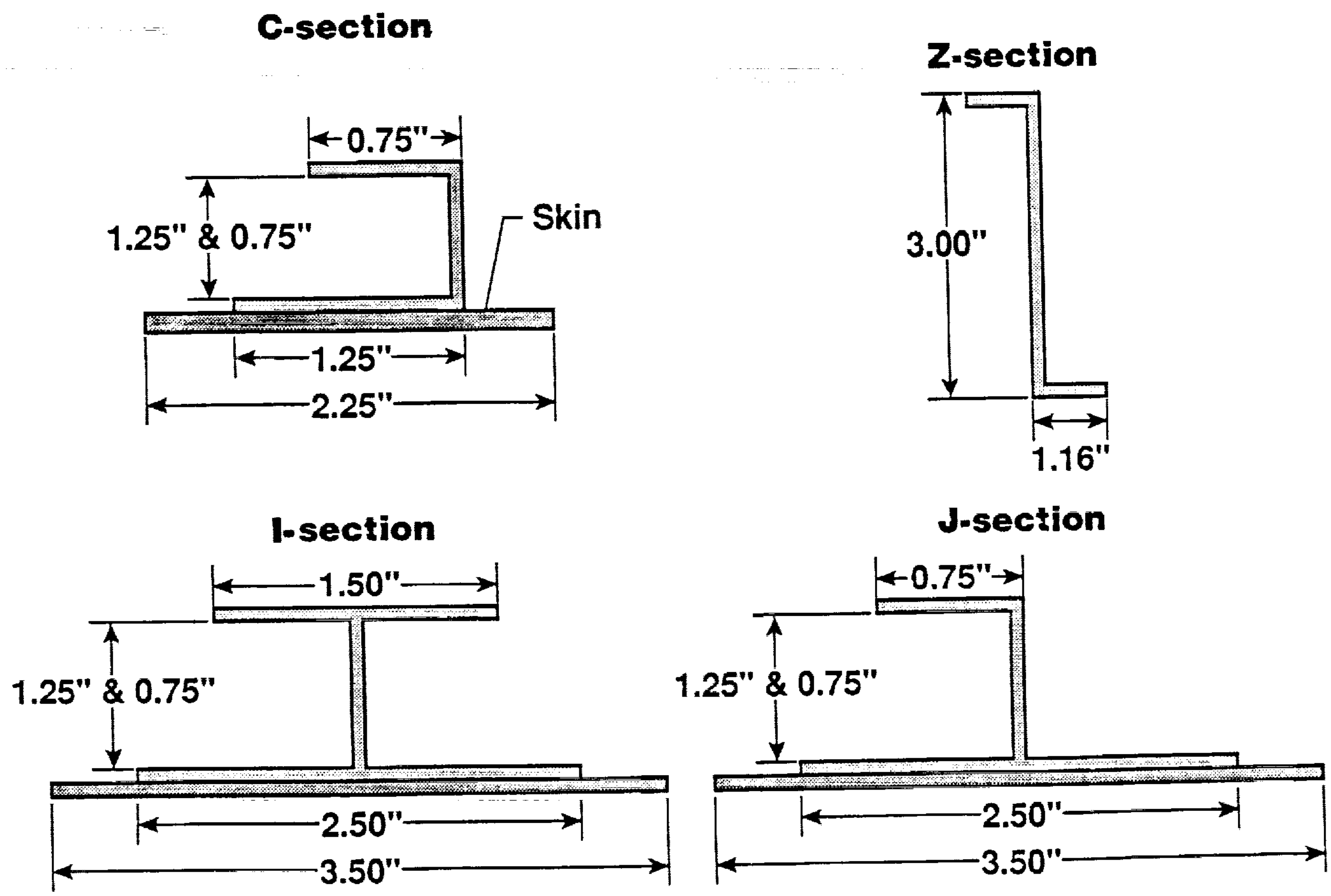


Figure 5

DYNAMIC LOADING BEHAVIOR OF COMPOSITE Z-FRAME

One of the first geometries to be studied under static and dynamic loadings was the Z-cross section. The six-foot diameter frames were constructed of 280-5HA/3502, a five harness satin weave graphite fabric. The height of the frame was 3 inches with a total width of 2.25 inches and about 0.08 inches thick. Lay-up of the frames was quasi- isotropic. Initial tests were with 360 degree frames made from four 90 degree segments joined with splice plates. Additional tests were conducted with half frames since the top half of the complete frames were undamaged in the tests.

Figure 6 presents results from the dynamic studies of the response of the Z-frames under approximately a 100 lbm floor loading at an impact velocity of 20 fps. As noted in the figure, the splice plates joining the segments of the frame are 45 degrees up the circumference from the point of impact. Also, complete failures (fractures) of the Z-section frames occurred at the bottom and approximately 60 degrees from the bottom. Potentially, it appears that the presence of the splice plates may have influenced the locations by moving the top failure points up a few degrees to about the 60 degree locations.

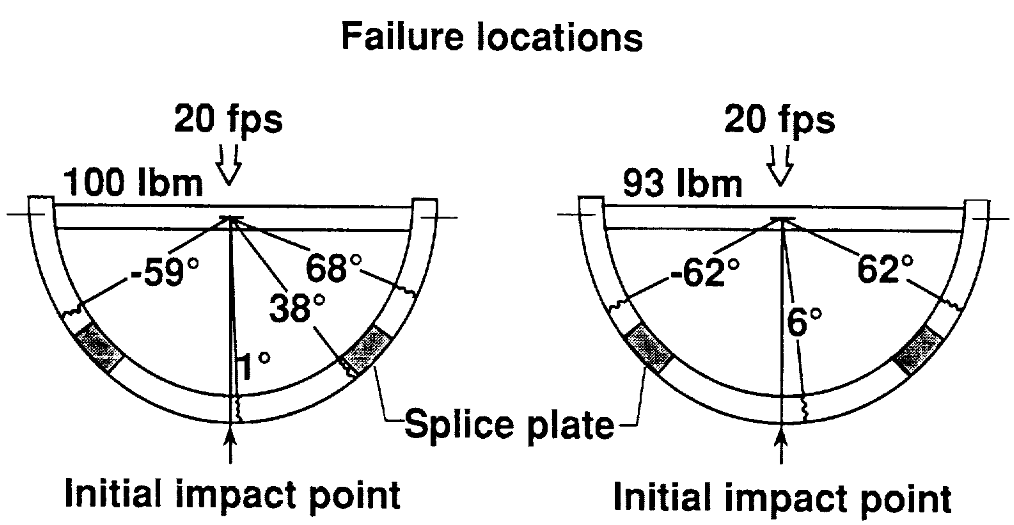


Figure 6

COMPOSITE FUSELAGE FRAME TEST

As a result of earlier tests with the Z cross-section circular frame, a 3.5 inch wide skin material was added to the I-, J-, and C- frame concepts to increase the torsional stiffness of the cross-sections and limit out-of-plane rotations and deformations. The skin, a $[\pm45/0/90]_{2S}$ lay-up sixteen ply (.08 inches) thick, was cocured with the 6 foot diameter frames. Data in Fig. 7 are for a lay-up of the frame of $[\pm45/45/90/0_3]_S$. Both the skin and the frame were fabricated with AS4/5208 graphite-epoxy material.

Figure 7 shows a typical set-up of a composite fuselage I-frame in a 120 000-lbf loading machine prior to a quasi-static test. The purpose of the study was to evaluate the effect of floor placement on the structural response and strength of the circular fuselage frames constructed of graphite-epoxy composite material. A steel I-beam was attached horizontally across the composite frame at the diameter position to simulate the floor. The horizontal floor positions were designated by the included angle measured between the ends of the floor attachments about the center of curvature of the frame. For example, the frame with the floor at the diameter is designated the 180° floor since the arc is 180° between the attachment points.

A vertical compressive load was applied to the composite fuselage frame through the simulated floor beam and the lower platen of the load machine. Special clamps (see lower right of Fig. 7) were used to bolt the I-beam to the composite frame and a 170° F melting point metal was poured into the small gap between the clamp and the frame to eliminate possible motion in the joint. As shown at the bottom left and top right of the figure, additional tests were conducted where the floor location was moved to produce 120° and 90° arcs. In each test the specimen was loaded at a rate of 500 lbf/minute up to a maximum of 1000 lbf.

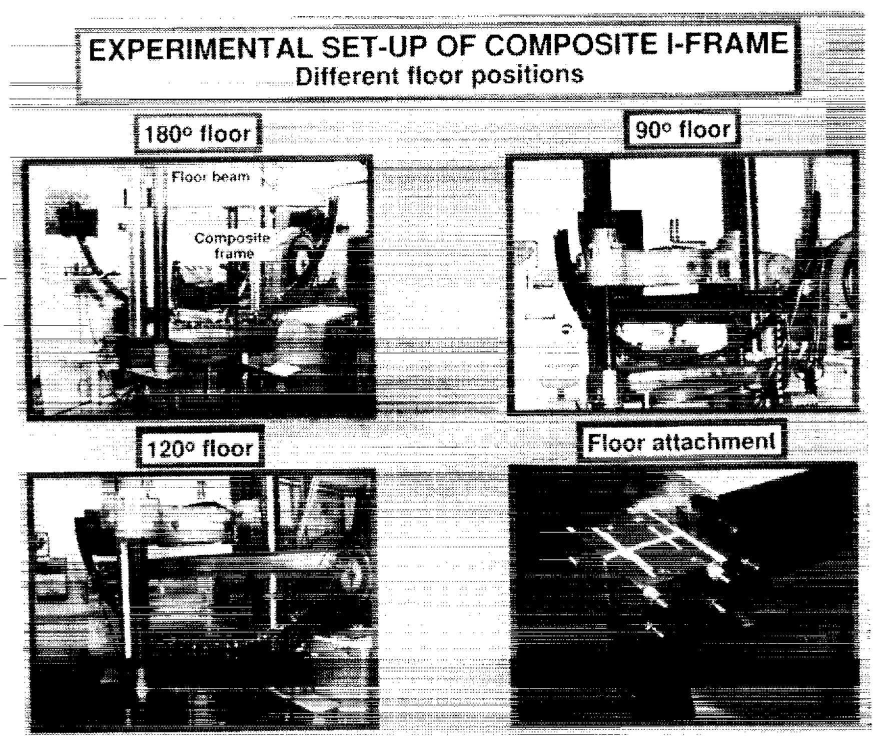


Figure 7

FINITE ELEMENT MODEL OF I-FRAME

To gain an understanding of the physics of behavior, the experimental research of structures under crash loadings is generally accompanied by analytical prediction or correlation studies whenever feasible. Various finite element codes with capabilities for handling dynamic, large displacement, non-linear response problems of metal and composite structures were used as tools in the research efforts. The analytical results presented in this paper were generated with a nonlinear finite element computer code called DYCAST (DYnamic Crash Analysis of STructures (Ref. 22) developed by Grumman Aerospace Corporation with principal support from NASA and the FAA.

Using the beam element from the DYCAST element library, the I-beam model illustrated in Fig. 8 was formulated. The combination of outer skin and I-frame were modeled with only I-beam elements. Since the skin lay-up provided less stiffness than the lay-up of the I-frame, the skin width was reduced by the ratio of the computed stiffnesses of the skin to the I-frame. As a result, the 3.5 inch skin width was reduced to approximately the same 2.5 inch width as the bottom flange of the I-frame itself. Thus, the resulting model consisted of straight ISEC elements with the bottom flange and skin combined to be 0.16 inches in thickness with only the material properties of the I-frame being used in the model. Symmetry was utilized and thirty-nine I-beam elements were used for the beam model. Both a force and a moment loading was applied to the top end of the frame. The static analytical load was increased linearly to a maximum load of 1000 lbf in 50 pound increments. The analytical results of the model are compared to the experimental data in the following section.

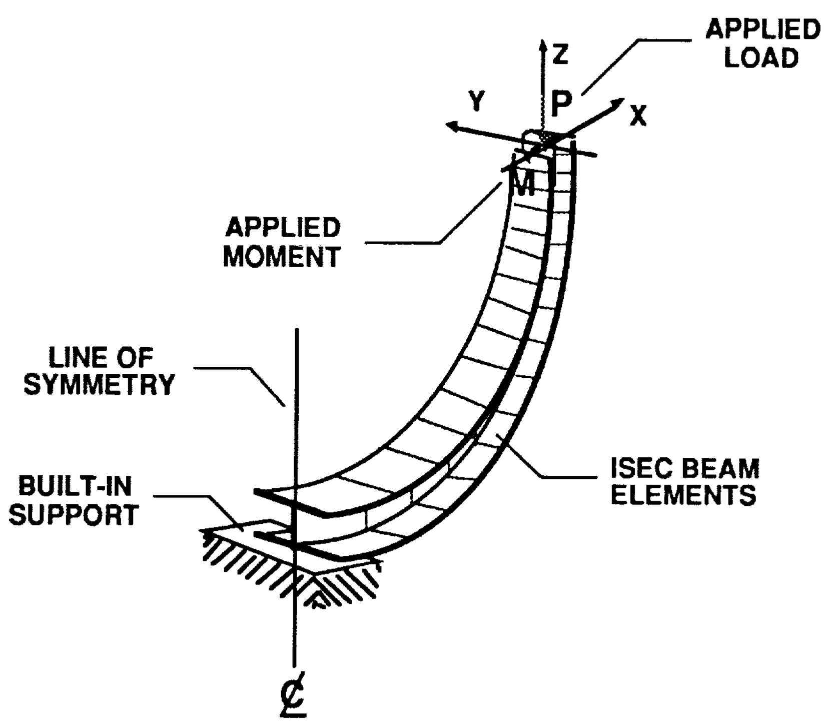


Figure 8

EXPERIMENTAL AND ANALYTICAL STRAIN COMPARISON FOR STATIC LOADING OF THE COMPOSITE I-FRAME

Comparisons of the experimental and analytical strain results on the I-frame were made. For example, Fig. 9 presents typical comparisons of the analytical distributions of strain with the experimental distributions for the 180°, the 120°, and the 90° floor positions, respectively, for both the skin and inner flange. Evaluation of the experimental and analytical strain distribution on the frame for three floor positions indicates a number of important points which include: (a) maximum strains were at the 0° or ground contact location with two secondary maximums occurring at symmetric locations between ± 55°-60° from the bottom contact area; (b) the predicted outer skin strain distribution exhibits the same "sea gull" shape as measured in the experiment; and (c) similar inverted circumferential strain distributions were noted for the inner flange of the frame as occurred in the experiment. The agreement in the magnitudes of the analytical and experimental strains and the shape of the distributions are considered excellent.

The effects on the response of the composite frame from changing the floor position in the composite frame were: (a) to alter the magnitude of the strain (moment) but not the common, general "sea gull" shape of the distribution under vertical loading; (b) to constrain the general "sea gull" shaped strain distribution to occur in the frame segment below the floor attachment locations; and (c) to increase the effective global structural stiffness of the frame as arc length of the frame was decreased.

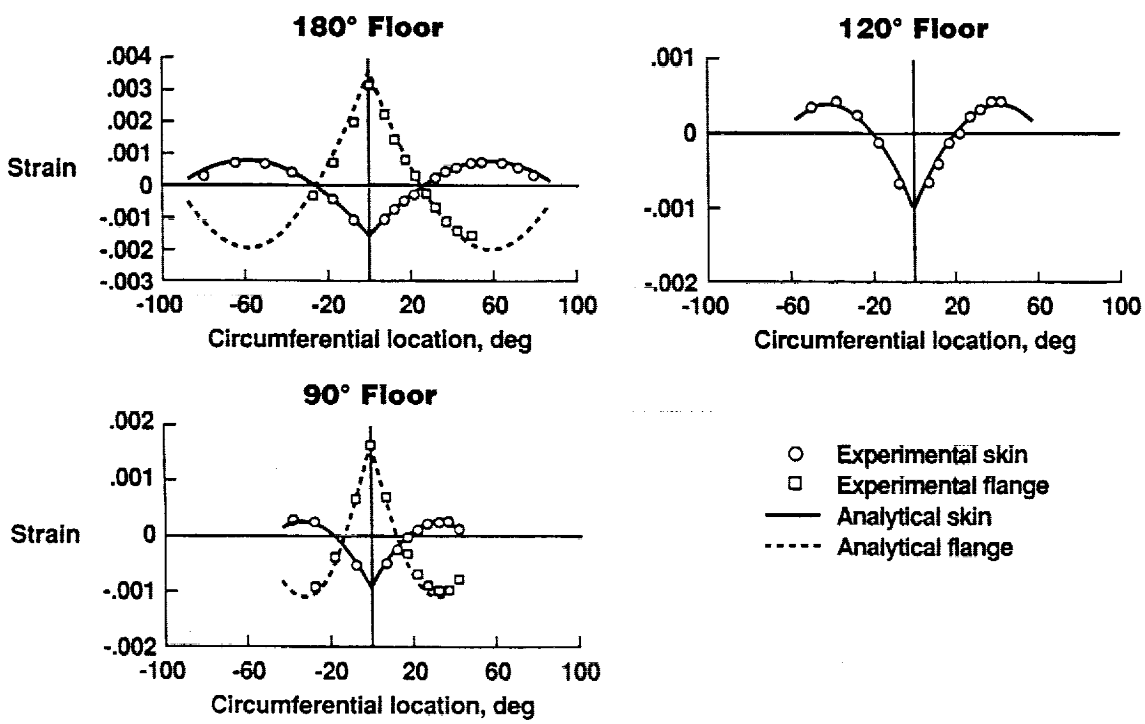


Figure 9

COMPOSITE SUBFLOORS WITH AND WITHOUT SKIN

As indicated previously, the approach of studying simple structural elements and then moving to combinations of these elements in more complex substructures has been taken in the development of a data base on the dynamic response and behavior of composite aircraft structures. The approach parallels the one used during the general aviation and transport aircraft programs. Consequently, three composite subfloor structures were fabricated following the initial investigation of the Z-frames discussed above.

Figure 10 is a photograph of the skeleton and skinned subfloor specimens constructed with three of the single Z-section frames similar to those that were studied earlier. Pultruded J-stringers attached the three frames through metal clips and secondary bonding methods. Aluminum floor beams tied the top diameter of the frames together to form the lower half of the subfloor. Notches in the frames allowed the stringers to pass through the frames. Two subfloors without skin (called skeleton subfloors) were fabricated. A third specimen (called skinned subfloor) had a ±45 lay-up skin bonded and riveted to the frames to form the lower fuselage type structure. Both static and dynamic tests were conducted with the subfloors.

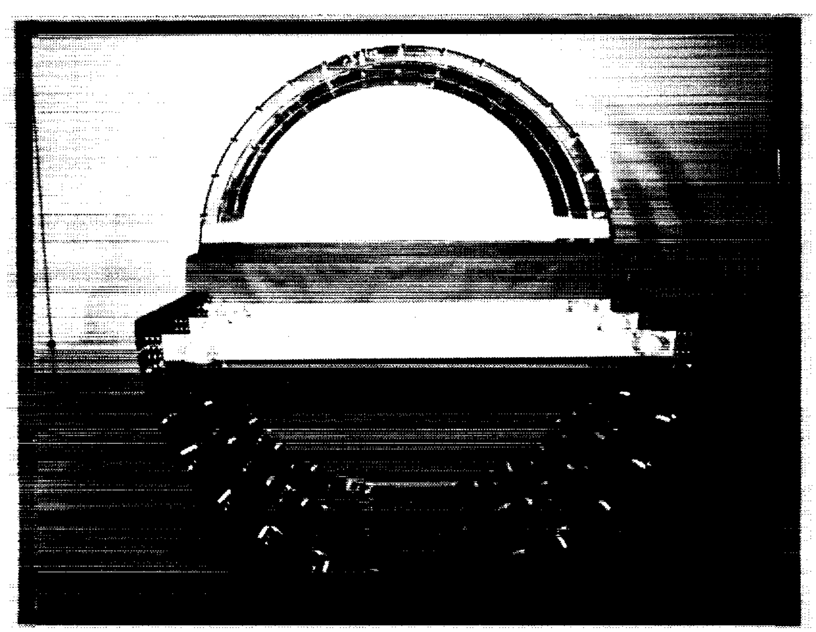


Figure 10

COMPOSITE SUBFLOOR BEHAVIOR--SPECIMEN WITHOUT SKIN

For the three composite subfloor specimens used for impact studies, two static and two dynamic tests were conducted on the subfloors. With the skeleton subfloor, a static and a dynamic test to destruction was conducted. With the skinned subfloor, a non-destructive static test followed by a dynamic test to failure was conducted.

Figure 11 shows the locations of fractures of the skeleton subfloor after an impact test onto a concrete surface at 20 feet per second. In the dynamic test of the skeleton subfloor, fractures were produced at notches in the frames. The locations, shown in Fig. 11, were also near the point of impact (about 11 degrees because of the splice plate) and at two other locations up the circumference of the frames (45 degrees and 78 degrees) and involved all three frames for a total of 15 fractures. The impact energy exceeded the energy absorbed by the local fractures and the floor bottomed out in the impact.

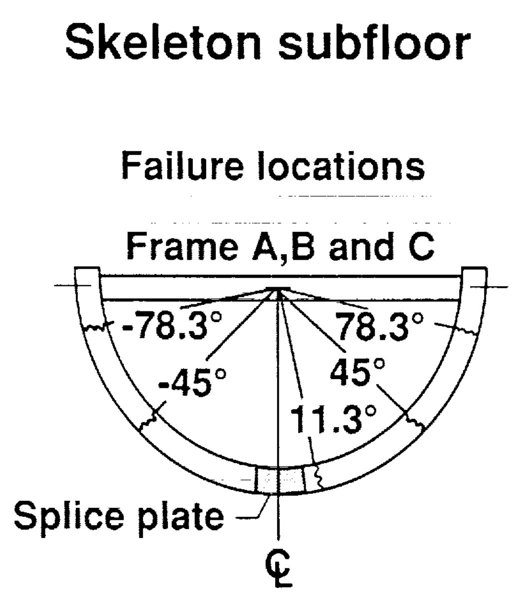


Figure 11

COMPOSITE SUBFLOOR BEHAVIOR--SPECIMEN WITH SKIN

Figure 12 presents impact results for the skinned subfloor after an impact of 20 feet per second. Points of failure of the frames in this specimen are indicated in the figure. Again the points of failure are at/near the impact point (within 12 degrees) and circumferentially at about 56 degrees up both sides of the frame on the middle and back frame and 45, 12 and 22.5 degrees on the front frame. It was observed that the subfloor impacted first on the front area which possibly explains the 12 and 22.5 degree fractures being different from the other locations. Again all three frames were involved in the failures. Some delamination of the frames from the skin was evident but the skin remained intact.

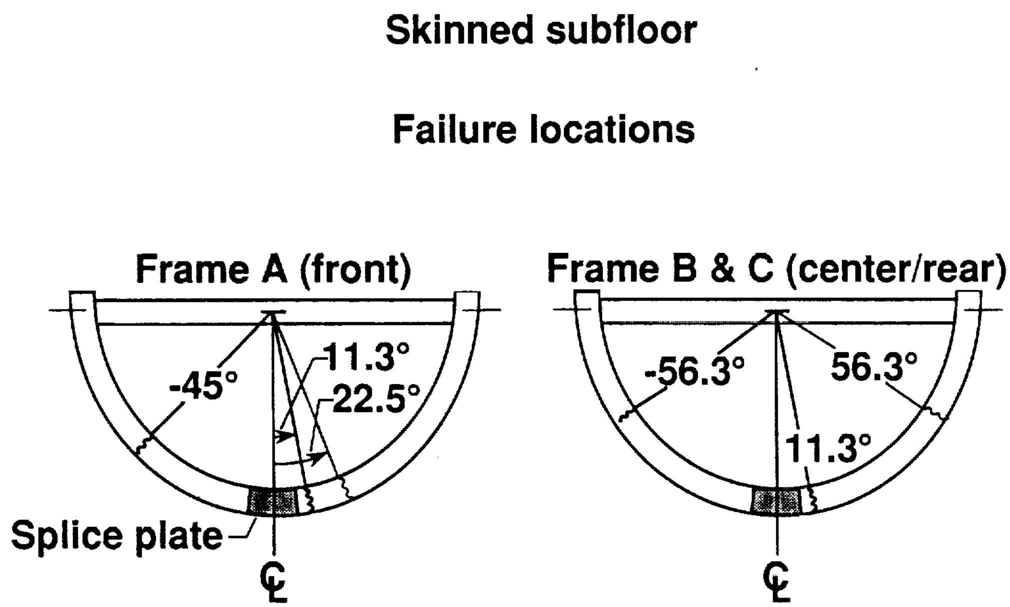


Figure 12

COMPARISON OF FRAME BEHAVIOR WITH SUBFLOORS

The determination of the effect of the floor location on the structural response of fuselage frames has aided in the understanding and prediction of full-scale subfloor or fuselage response to crash loading. For example, Fig. 13 shows a comparison of the normalized experimental dynamic strain distribution on the flange of the skeleton subfloor and the skin location corresponding to the flange position of the skinned composite subfloor specimens with the analytical I-frame strain. The results from the simple frame show a strong similarity to the response of the more complex subfloor structures. The structures share in common the generally circular or cylindrical shape, the vertical loading situations, and under vertical loads have strain (moment) distributions which have maximums at the point of loading and at approximately ±45° to ±60°, depending on boundary conditions, around the circumference from the ground contact point. Analytical results show the same distribution with maximums corresponding to the experimental locations. Failures of the subfloor structures were noted between these same 45° to 60° circumferential locations in the dynamic tests (see Ref. 23).

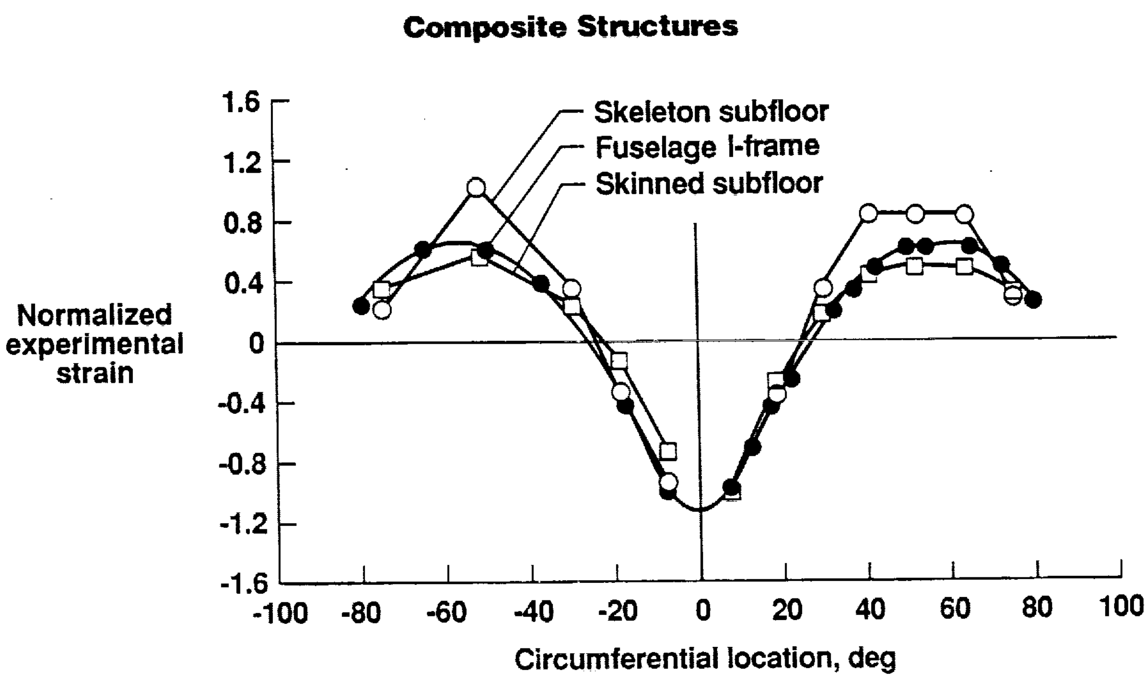


Figure 13

METAL TRANSPORT FUSELAGE FAILURE BEHAVIOR

To relate previous metal transport failure behavior to the current studies and observations with composite structures, data from tests of metal aircraft sections to support transport aircraft research efforts are included. NASA Langley Research Center conducted drop tests of two 12-foot long fuselage sections cut from an out-of-service Boeing 707 transport aircraft to measure structural, seat and occupant responses to vertical crash loads, and to provide data for nonlinear finite element modeling. One section was from a location forward of the aircraft wings and one was from aft of the wing location (Refs. 24 and 25). The sections were loaded with seats, anthropomorphic dummies, data acquisition system pallet, power pallet, and camera batteries to test not only structural, seat, and occupant responses but also to test equipment to be used in the full-scale transport crash conducted later.

Structural damage locations of the transport aircraft structures resulting from the 20 fps drop tests are shown in Fig. 14. The damage to the transport sections was confined to the lower fuselage below the floor level. Under the vertical impact of 20 fps, all of the frames ruptured near the bottom impact point. Plastic hinges formed in each frame along both sides of the fuselage at about 50 degrees up the circumference from the bottom contact point. The upward movement of the lower fuselage was approximately 22-23 inches at the forward end and 18-19 inches at the rear for the section taken from forward of the wing location, whereas in the section from aft of the wing location the crushing was about 14 inches forward and 18 inches in the rear. Although the aircraft structures are metal and the failures discussed above involve plastic deformations with some tearing of the metal rather than brittle fractures, the general observed failure pattern and locations for the transport fuselage sections are noted to be quite similar to the results of the composite frames and subfloors discussed herein.

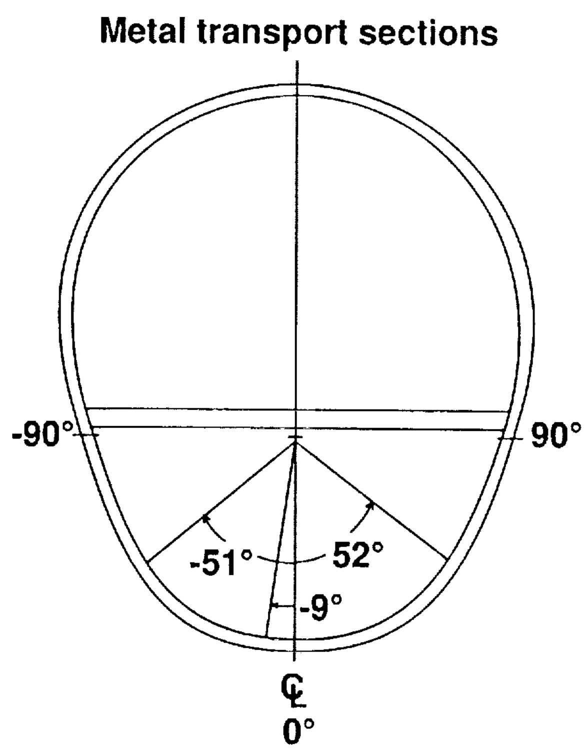


Figure 14

SUMMARY OF OBSERVED FAILURE BEHAVIOR

The response behavior determined during the studies of full-scale aircraft sections, fuselage frames, and subfloors are summarized in Fig. 15. The figure shows normalized moment distribution on a representative frame of the various specimens and the failure locations which were noted from static or dynamic tests. The visual impression is quite striking among the various specimens. It is suggested that from the results of simpler frames to the more complex subfloors and full-scale sections, a strong similarity is evident in the failure behavior of the structures. The structures share in common the generally circular or cylindrical shape, the normal loading situations, and what appears to be a similar pattern of failure behavior. Analytical models of frame structures under vertical loads have moment distributions which have maximums at the point of loading and at approximately 45 to 60 degrees (depending on boundary conditions) around the circumference from the ground contact point. Failures of the structures were noted at these same locations. Such observations can help dynamists gain a better understanding of what to expect from such structures in crash loading situations, can guide designers of new structures to better account for the vertical crash loads, and allow better energy absorption to be included in the new designs. Additionally, the observations can help analysts better model the aircraft structures for predicting the failure responses and behavior under crash situations.

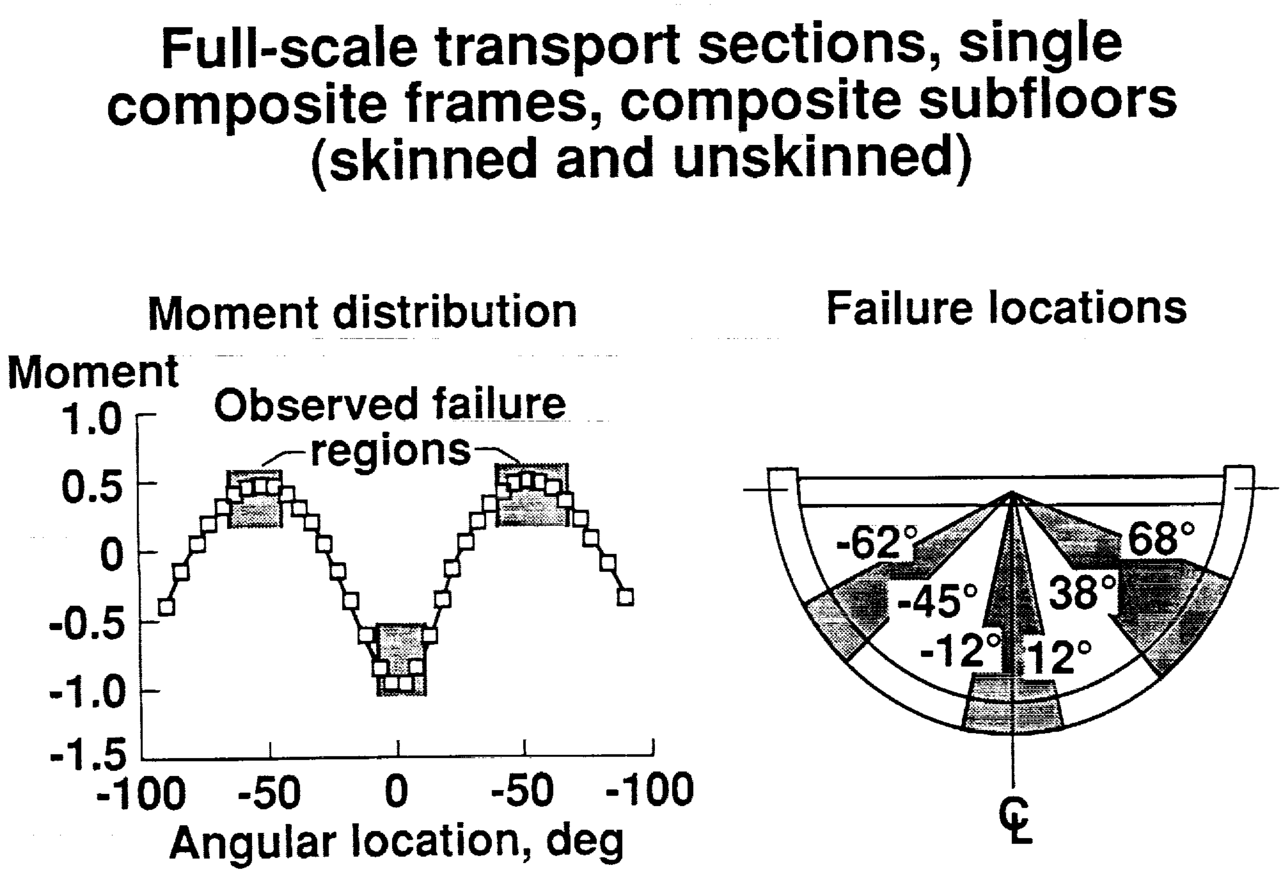


Figure 15

FULL-SCALE COMPOSITE AIRCRAFT CONCEPT

As was illustrated in Fig. 3, one of the four program elements of the Composite Impact Dynamics Research in the Landing and Impact Dynamics Branch of the Structural Dynamics Division at NASA Langley Research Center is full-scale testing of aircraft structures under crash loading conditions.

Two full-scale composite general aviation aircraft structures, two complete wing sets, and landing gears have been obtained for testing. As shown in Fig. 16, the full-scale test aircraft is an 8 place airplane with twin Pratt and Whitney engines (650 hp each) powering a pusher type propeller. The gross takeoff weight is approximately 7200 lbm with empty weight being approximately 4000 lbm. Overall length of the plane is 38 feet with a wing span of 39 feet. The structure of the test aircraft represents a composite skin/frame type fuselage construction concept with the exception of the interior floor structure which consists of aluminum beams on which the seat rails and seats are mounted. Design support testing is underway to replace the existing floor structure in one of the two aircraft with an energy absorbing concept constructed with composite materials. The retrofit approach is, of course, necessitated and as a consequence, special tests have been undertaken to develop the replacement floor concept to assure that structural behavior and failure loads and modes of failure are achieved in the concept prior to inclusion in the aircraft.

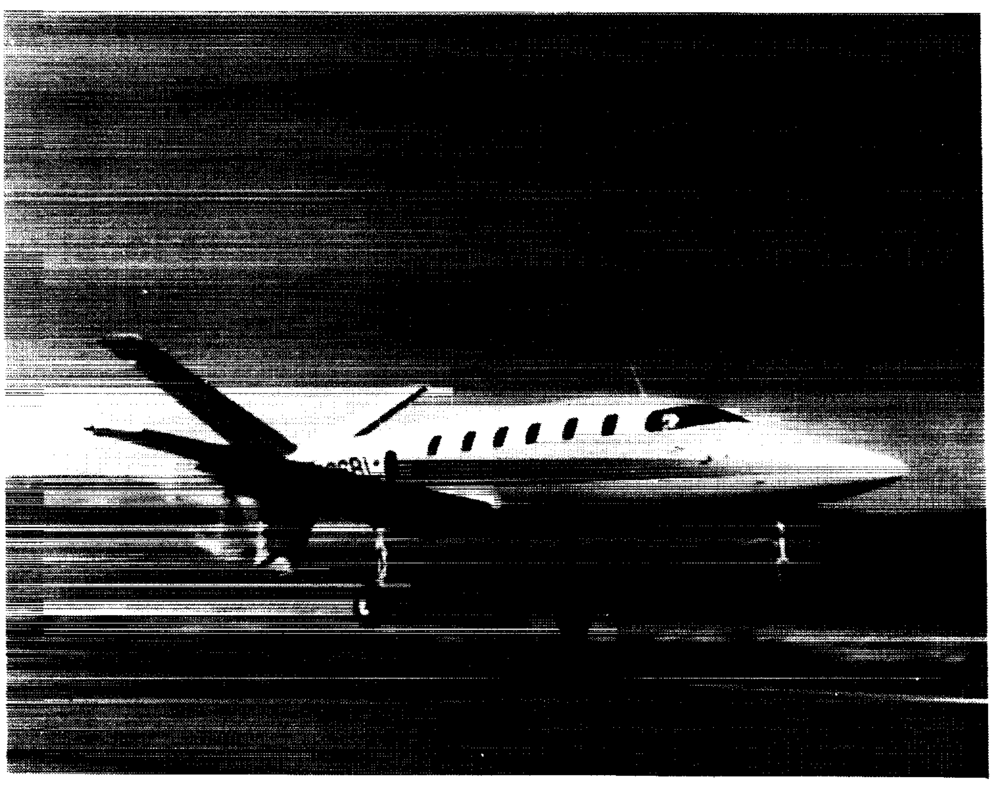


Figure 16

ENERGY ABSORBING BEAM DESIGN FOR COMPOSITE AIRCRAFT SUBFLOOR

A preliminary experimental program is being conducted to study the behavior of energy absorbing composite spars for aircraft subfloors. The study, which is a part of a wider full scale aircraft test program, examines the efficiency of replacement floor structures for an existing all composite fuselage aircraft shown in the lower right of Fig. 17. The efforts are a continuation of previous research (Refs. 7 and 13) dealing with crashworthy metal aircraft subfloor structures. A typical section of the composite fuselage with the original subfloor structure is shown in the center of the figure. As shown in the center figure, the four spars that support the seat rails are aluminum, whereas the rest of the subfloor structure is graphite composite. Static tests of such a subfloor section have shown that the existing structure is too stiff and too strong for cushioning loads resulting from crash speeds in the neighborhood of 30 fps, as recommended by the Part 23 of the Federal Aviation Regulations. Thus, the objective of this study is to design and test a retrofit subfloor structure that would provide the desired cushioning (less than 20 g of occupant load) at crush speeds of approximately 30 fps. In particular, the four aluminum sparsj are to be replaced by composite sine wave beams. The sine wave composite beam concept, shown at upper left, has been examined previously with encouraging results (Ref. 19).

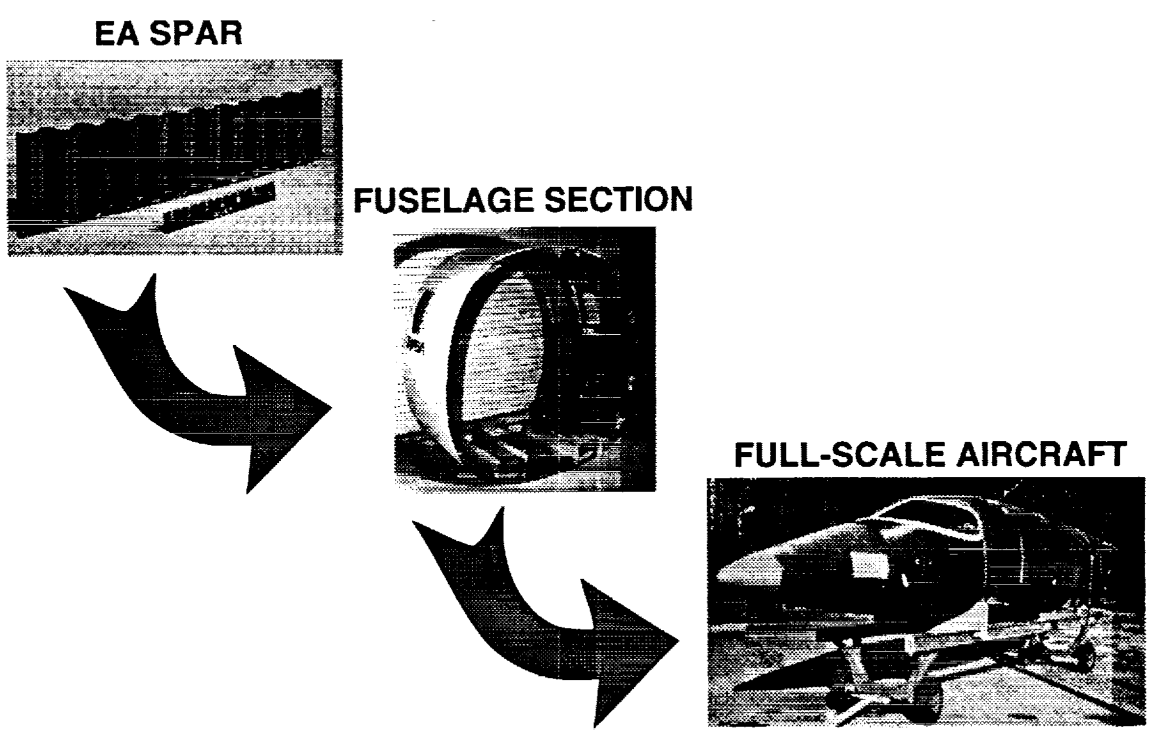


Figure 17

PROPOSED COMPOSITE SUBFLOOR STRUCTURE

A schematic of the proposed subfloor structure is shown in Fig. 18. An ideal beam design should contain two flanges for improved stiffness which, in addition, would offer two crush initiators, one in juncture between the web and the flange. However, because of the double curvature of the fuselage and the retrofit nature of the problem, the beams under construction are limited to a single flange on the top which contains the single crush initiator.

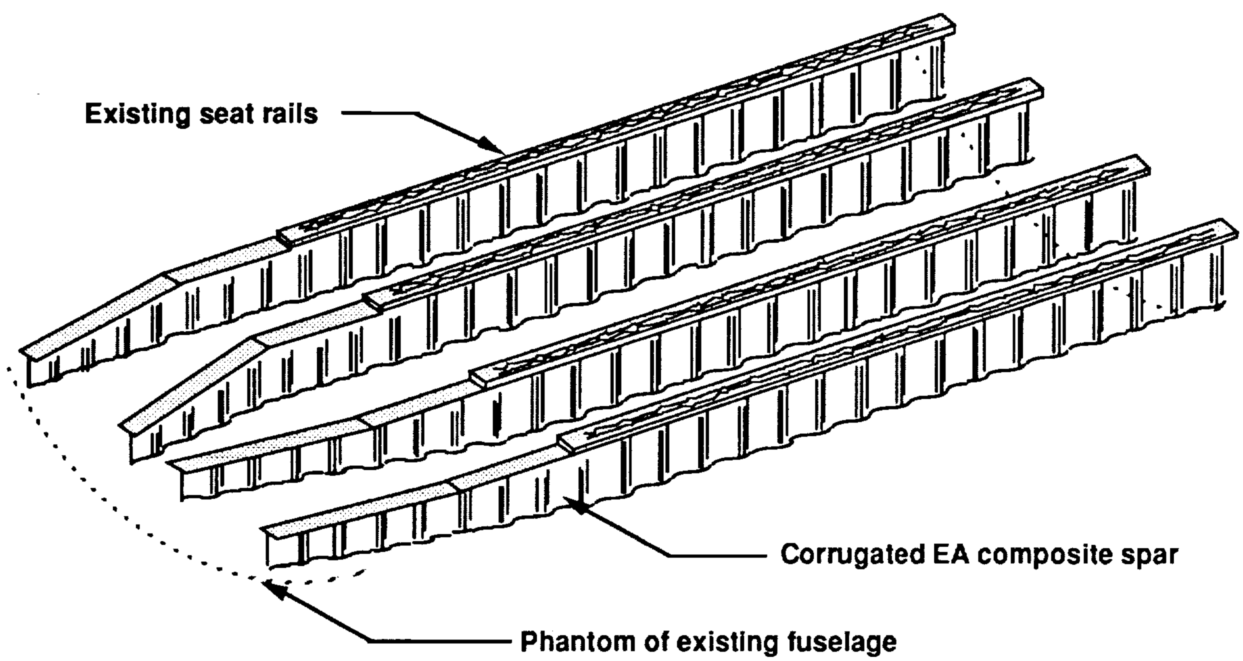
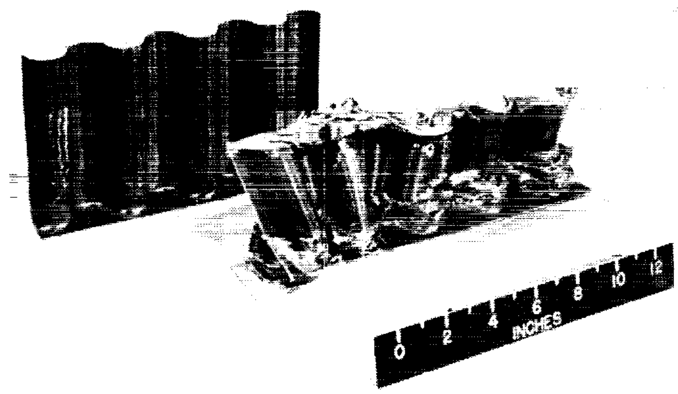


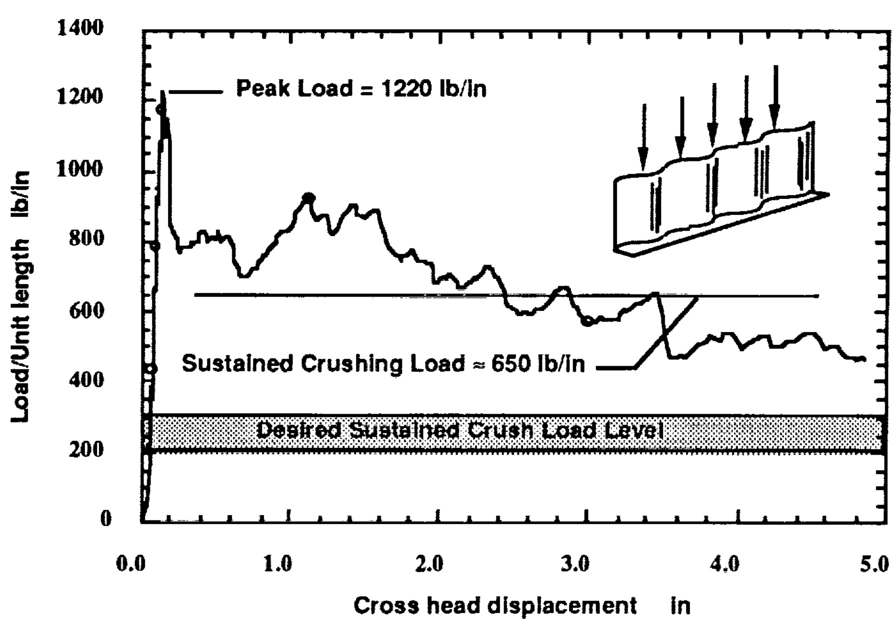
Figure 18

STATIC RESPONSE OF INITIAL COMPOSITE SPAR DESIGN

A preliminary spar beam with an inverted "L" cross section has been fabricated according to the suggested geometric details in Ref. 19. The stacking sequence and composite systems were selected such that the stiffness of the beam transverse to the longitudinal axis was half that of the aluminum spars. Both graphite and aramid reinforced epoxy have been used in the stacking sequence; $(\pm 30°\text{gr.}/\pm 45°\text{kevlar fabric}/0°_2\ \text{gr.})_S$. A series of static and dynamic tests have been conducted to evaluate the overall performance of the spar design. A typical virgin and crushed spar section is shown in Fig. 19 (a) and a load/displacement plot of a quasi-statically loaded composite spar is shown in Fig. 19 (b). Note that, while the section crushed progressively, thus absorbing a large amount of energy, the ultimate load and the sustained crushing loads of approximately 1200 and 650 lb/in respectively are far too high as opposed to 200 - 300 lb/in of desired load.



(a)

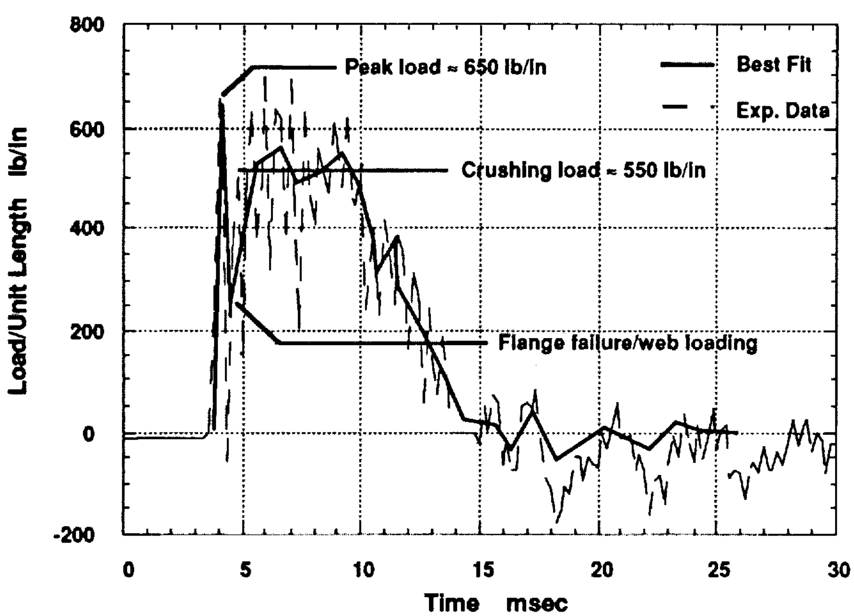


(b)
Figure 19

DYNAMIC RESPONSE OF INITIAL COMPOSITE SPAR DESIGN

A series of dynamic tests were carried out on a 14 ft drop tower shown in Fig. 20 (a) to simulate a 30 fps mass drop equal to the corresponding weight of the seat and the occupant which would load the spar section. The mass for a 12" long spar section was 184 lbm. A typical load/time plot from a dynamic test is shown in Fig. 20 (b). Note that, while the value of the sustained crushing load was comparable to the one obtained in the corresponding static tests, the ultimate load was much lower. The dynamic results, shown in Fig. 20 (b), indicate that there was no dynamic rate effect up to the 30 fps impact velocity of the test as compared to the static data and that the loads from the dynamic test of the preliminary beam design is also much too high for a human occupant.

(a)



(b)
Figure 20

CONSIDERATIONS FOR NEW COMPOSITE SPAR DESIGN

The previous results together with the experience gained in the fabrication of the first spars influenced the design of a new set of spars and tests. Some of the factors and constraints that controlled the design of the new spars included:

(1) simplification of the fabrication process - eliminate unidirectional prepreg, complex angles, and hybridization
(2) elimination, if possible, of the graphite reinforced material to improve ductility under dynamic loads- to ensure survivability of the flange under normal landing loads
(3) improvement of the web bending stiffness to ensure adequate longitudinal spar bending stiffness following the loss of the flange
(4) reduction of the ultimate and the sustained crushing loads to less than 300 lb/in (and greater than 200 lb/in) - to improve cushioning
(5) symmetric loading - apply load symmetrically from the flange to the web to improve global stability of the spar-web and improve stroke efficiency

It was found that more design goals could be met with a sandwich construction and that some of the additional complications associated with the fabrication of the sandwich spars could also be offset with the simplification of the skin lay-up. Thus, a "T" section illustrated at the bottom of Fig. 21 was chosen instead of the original "L" section. A number of sandwich spar sections are being fabricated with fabric kevlar webs and hybrid flanges. A full test matrix is shown above the sketch of the "T" section. Static and dynamic testing of these sections will commence after specimens have been instrumented.

Flange Cover Stacking Sequence	Web Stacking Sequence	Flange Core* Thickness Density	Web Core* Thickness Density	Flange Width in	Specimen Length in	Specimen Height in	No. of Specimens	Type of Test
±45f/0/±45f¤ Carbon	±45f Kevlar	1/4 in 3lb/ft^3	1/4 in 3lb/ft^3	3	12	8	2	Static Dynamic
±45f/0/±45f* Carbon	±45f Kevlar	/	1/4 in 3lb/ft^3	3	12	8	2	Static Dynamic
±45f/0/±45f* Carbon	(±45f)s Kevlar	1/4 in 3lb/ft^3	1/4 in 3lb/ft^3	3	12	8	2	Static Dynamic
±45f/0/±45f* Carbon	(±45f)s Kevlar	/	1/4 in 3lb/ft^3	3	12	8	2	Static Dynamic
±45f/0/±45f* Carbon	±45f Kevlar	/	1/2 in 3lb/ft^3	3	12	8	2	Static Dynamic
±45f/0/±45f* Carbon	(±45f)s Kevlar	/	1/2 in 3lb/ft^3	3	12	8	2	Static Dynamic

* - LAST-A-FOAM
¤ - Fabric

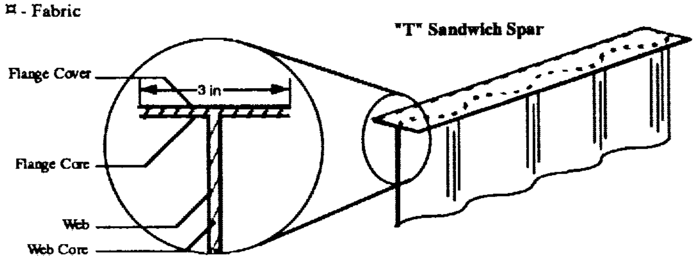


Figure 21

FINITE ELEMENT MODELING OF THE COMPOSITE AIRCRAFT

As part of the overall program, along with the full-scale testing of the composite aircraft concept, finite element predictions of the behavior and loads of the aircraft under crash conditions will be conducted. The computer program KRASH (Ref. 26), which is a three-dimensional, hybrid, finite element modeling technique, will be used to predict response and loads of the test aircraft under selected impact parameters using the KRASH finite element aircraft model (or variation thereof) shown in Fig. 22. As depicted in the figure, KRASH represents the structure of the aircraft as a combination of masses, beams, rigid connections, and external springs.

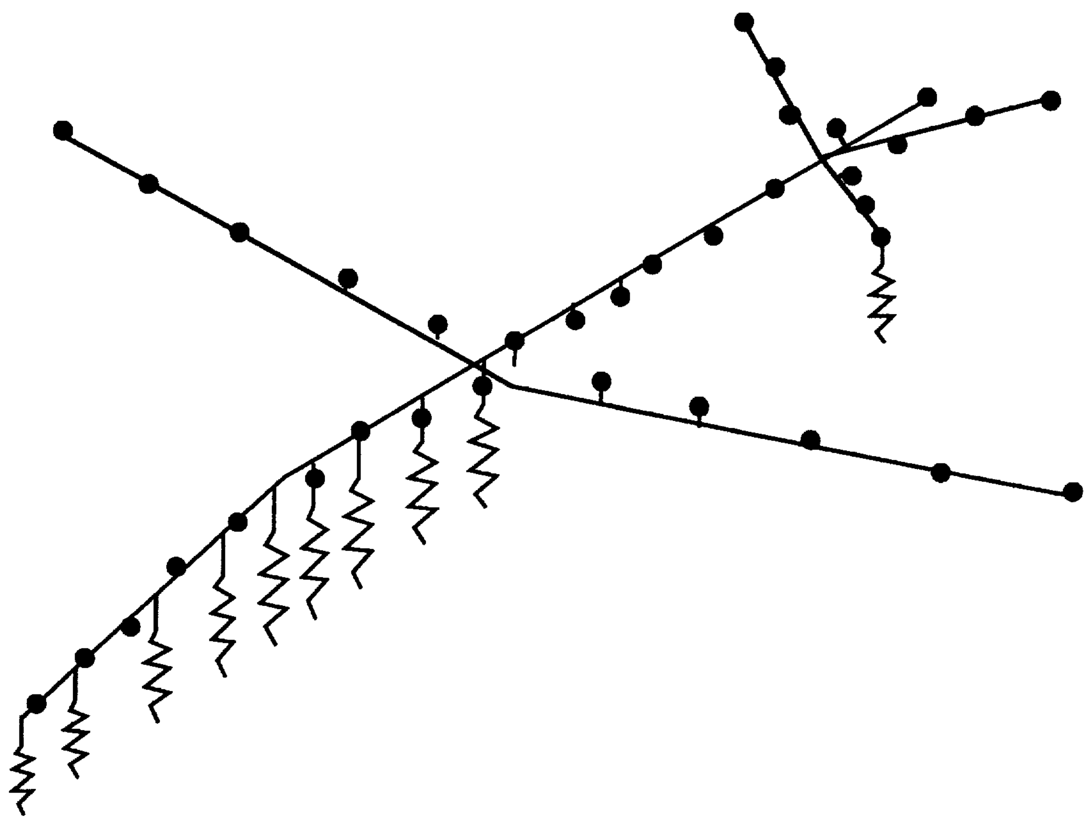


Figure 22

SCALING OF COMPOSITE STRUCTURES

The final research area in Composite Impact Dynamics Research is the scaling of composite structures. One activity is discussed herein, but Refs. 27 to 31 deal with other studies.

Figure 23 shows some typical results of a study to isolate the factors responsible for scale effects in the tensile strength of angle ply graphite/epoxy composite laminates. Two generic ±45° lay-ups were studied, one with blocked plies and one with distributed plies with stacking sequences containing between 8 and 32 plies. The 8-ply laminate consisted of off-axis plies arranged in a $(+45°/-45°)_{2S}$ sequence, and was denoted the baseline or model stacking sequence. A high modulus, high strength brittle graphite-epoxy system (AS4/3502) was used to fabricate six "scaled-up" laminates with the following stacking sequences: $(+45°_n/-45°_n)_{2S}$ (blocked plies), and $(+45°/-45°)_{2nS}$ (distributed plies), where n = 2, 3, and 4. Tensile coupon specimens having four scaled sizes were constructed including full scale size, 3/4, 2/4, and 1/4, corresponding to n equal 4, 3, 2, and 1, respectively. Angle ply laminates are commonly used for damage tolerance in the surface of composite laminates where the load bearing plies are shielded against impact and fatigue loads. It is therefore important to understand the effect of specimen thickness and stacking sequence on the stress-strain response, the ultimate strength, and the mode of failure for this class of laminates.

Results in Fig. 23 indicate that for increasing specimen size: (1) strength decreased for blocked ply laminates; (2) strength increased for distributed ply laminates; and (3) strength of distributed laminates is greater than blocked laminates for a given specimen size. The significance of these findings, beyond the scaling of structures issues, is that ASTM standard tests for determination of the in-plane shear stiffness and strength are based on ±45° angle ply testing, although exact specifications for the laminate stacking sequence are not stated. Results of this research show that the values of strength can vary tremendously depending on whether the laminate stacking sequence contains blocked or distributed plies. Also, the size of the laminate, especially the number of plies, is important. Recommendations to improve the standard testing practices have been made to the ASTM so that a meaningful shear strength value, independent of specimen size, can be determined from tensile tests on ±45° laminates.

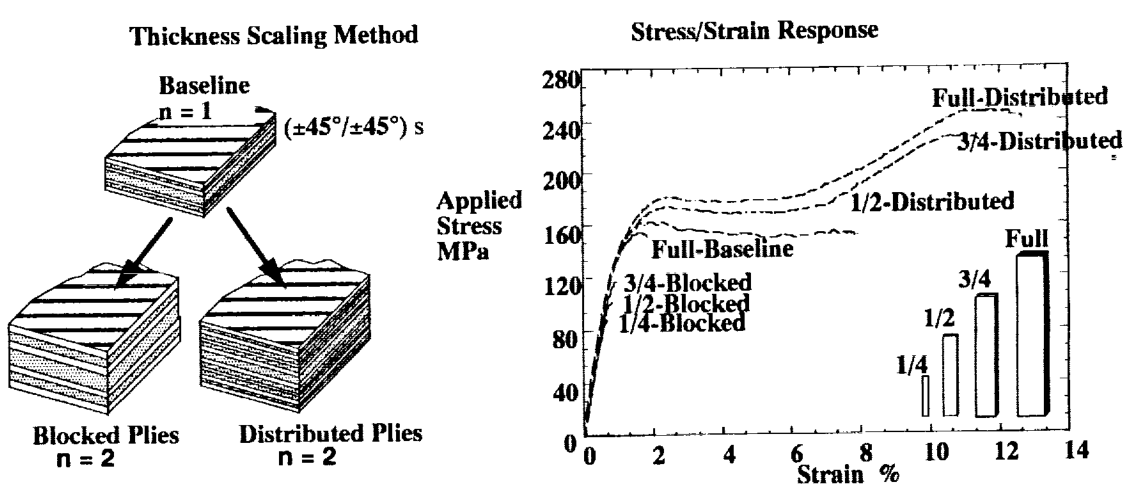


Figure 23

CONCLUDING REMARKS

The Composite Impact Dynamics Research Program at NASA LaRC focuses on generating a database for understanding composite structural behavior under crash loads, examines conventional and innovative metal and composite structures for meeting performance, integrity, and energy absorption requirements, analyzes/enhances analysis tools for composite applications, studies scaling effects in composite structures under static and dynamics loads, and conducts full-scale structures to verify performance of structural concepts.

Typical examples of research in each of the program elements were presented to illustrate the research effort. Experimental and analytical results were presented which showed the effect of floor placement on the structural response of circular fuselage frames constructed of graphite-epoxy composite material. The results from the simple frame showed a strong similarity to the response of more complex subfloor structures. The structures share in common the generally circular or cylindrical shape, the vertical loading situations, and under vertical loads have strain (moment) distributions which have maximums at the point of loading and at approximately $\pm 45^{\circ}$ to $\pm 60^{\circ}$, depending on boundary conditions, around the circumference from the ground contact point. Analytical results show the same distribution with maximums corresponding to the experimental locations. Failures of the subfloor structures were noted between these same 45° to 60° circumferential locations in the dynamic tests.

A design support test program to develop a composite energy absorbing floor structure to replace metal floor in a composite aircraft concept was outlined and preliminary results presented. A preliminary spar beam with an inverted "L" cross-section has been fabricated. A series of static and dynamic tests have been conducted to evaluate the overall performance of the spar design. A typical spar section crushed progressively absorbing a large amount of energy, but the ultimate load and the sustained crushing loads were far too high for human survivability. In the dynamic tests, the value of the sustained crushing load was found to be comparable to the one obtained in the corresponding static tests; however, the loads from dynamic tests of the preliminary beam design were also much too high for a human occupant. New design goals were established which should be met with sandwich type spar construction.

Scaling results were presented which have had wide-spread influence on standard test practices for material properties. ASTM standard tests for determination of the in-plane shear stiffness and strength are based on $\pm 45^{\circ}$ angle ply testing, although exact specifications for the laminate stacking sequence are not stated. Results of this research have shown that the values of strength can vary tremendously depending on whether the laminate stacking sequence contains blocked or distributed plies. In addition, size of the laminate, especially the number of plies, is important. Recommendations to improve the standard testing practices have been made to the ASTM so that a meaningful shear strength value, independent of specimen size, can be determined from tensile tests on $\pm 45^{\circ}$ laminates.

The Composite Impact Dynamics Research Program will contribute to the technology necessary for the development of improved composite structural aircraft concepts for energy absorption and enhanced passenger protection under crash loads.

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