The Combination of Mechanical System Simulation and Finite Element Analysis Software to Model Structural Failure in an Aircraft Accident Investigation

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ABSTRACT

The MSC/NASTRAN finite element analysis (FEA) code is used in conjunction with the ADAMS mechanical system simulation (MSS) program to simulate the structural behavior of the airframe of a commercial aircraft maneuvered beyond its design limits during a landing. FEA-generated linear elastic structures are subjected to non-linear boundary conditions in the MSS analysis to simulate the initial structural failure of the airframe. The analysis results are compared with crash site evidence and subsequent, independent engineering analysis.

Background

Recently a wide-body, commercial cargo liner crashed on landing at a major airport on the East Coast. The occupants of the aircraft escaped with minor injuries, although the aircraft and its cargo were destroyed. The accident site was meticulously examined by the National Transportation Safety Board, the FAA, and other parties to the investigation (including the aircraft operator, the aircraft manufacturer, the pilot's union, and the airport authority). Examination of the digital flight data recorder (DFDR) indicated that the aircraft had been subjected to loads which substantially exceeded the FAR requirements to which the aircraft had been designed, but questions arose as to where the structural breakup initiated. From the positioning of debris on the runway, it appeared that a fitting in the right side of the aircraft body had failed first, but re-examination of the structure, even when subjected to loads substantially above design ultimate, could not explain this failure mode. Considering the possibility that the structure had been subjected to some unanticipated dynamic load condition, the decision was made to model the event with the ADAMS MSS code.

Brief ADAMS Program Description

The ADAMS (Automatic Dynamic Analysis of Mechanical Systems) computer code is a product of Mechanical Dynamics, Inc. (MDI), of Ann Arbor, Michigan. ADAMS is a Mechanical System Simulation (MSS) program designed to solve the equations of motion for complex mechanical systems undergoing, generally, large motion. Based on the principles of Lagrangian Dynamics, ADAMS numerically erects and solves the system equations as functions of time. These equations are usually both algebraic and differential as well as highly nonlinear. The basic formulation employed by ADAMS is given in reference [1], p. 436. Lagrange's equations are given as

$$1) \qquad \mathbf{F}_j = \frac{d}{dt}\left(\frac{\partial \mathrm{L}}{\partial \dot{\mathbf{q}}_j}\right) - \frac{\partial \mathrm{L}}{\partial \mathbf{q}_j} + \sum_{i=1}^{m} \frac{\partial \mathrm{F}_i}{\partial \mathbf{q}_j} \, l_i - \mathbf{Q}_j = 0 \qquad \text{for } \mathrm{j} = 1, \ldots \mathrm{n}$$

$$2) \qquad \mathbf{F}_i = 0$$

ADAMS employs the Craig-Bampton [2] approach to implement modal flexibility. Initially, any point at which the component is 'touched' by other components in the system is declared a "hardpoint" and a (static) superelement [3] analysis is performed to condense the structure down to degrees of freedom at these locations. Next, these points are fixed in space and a modal analysis is performed. The total behavior of the system at any point in the simulation is considered to be expressible by the appropriate combination of these two sets of modes. Prior to implementing the flexible body, the mixed structural system is 'orthonormalized' to bring all retained modes into the modal regime. The additional equations occur as force expressions in the system equations, with modal displacements and velocities introduced as additional solution variables.

Modeling Approach

The decision was made to employ the actual aircraft DFDR information to guide the aircraft model through the landing event in order to bring it into contact with the runway for the purpose of determining the landing gear/airframe force-displacement histories. The wing and landing gear structures would be represented in ADAMS as beam structures. In addition to this, estimated airframe aerodynamic and simplified engine thrust data were to be incorporated to permit a qualitative estimate of the post-impact behavior of the subject aircraft, if practicable. A three-phase approach was initially taken to model the landing event. Initially, all of the structural components were modeled as rigid. This relatively simple model permitted the kinematic systems to be quickly verified. Next, the landing gear components were rendered flexible using the ADAMS BEAM (Euler-Timoshenko formulation) element. Finally, the wings were modeled using, again, beams, and the aerodynamic loads were distributed along them based on classical, linear aerodynamic assumptions.

Model Description

The ADAMS model is initially constructed in aircraft coordinates. At the start of the analysis, it is dynamically positioned at the runway threshold and subsequently moved in a controlled fashion down the runway (ref. fig. 1).

Mass properties for the various components were initially supplied by the manufacturer. As the system was modified to incorporate flexible components in the right main landing gear and wing, the (rigid) airframe and (rigid) landing gear mass properties were modified or removed to account for the flexible structural mass being added. The mass, inertia values, and CG location of the (residual) rigid airframe were adjusted to bring the assembled system properties into agreement with those initially provided by the manufacturer.

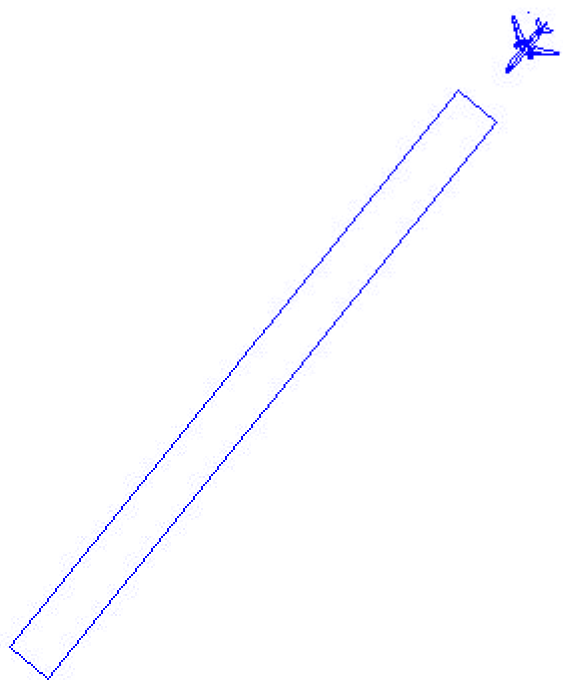

Fig. 1 Positioned Model Configuration

The airframe (ref. fig. 2) consists of a rigid fuselage and empennage to which is attached a flexible wing structure composed of lumped masses and constant-section beams, with beam properties reduced from root to tip to account for wing taper. The engine, main landing gear trunnion, and side brace support structure (referred to as the ‘trap panel’) are each connected by rigid offsets to their closest respective wing beam node. Thus, the loading due to each is reacted at a different position along the wing beam and will be influenced by wing flexure. It must be emphasized that the local stiffness properties at these offset locations are, at best, approximate.

Based on existing MSC/NASTRAN models, augmented, where necessary, by information from landing gear assembly and part drawings, the principal structural components of the landing gear were rendered flexible (ref. fig. 3)

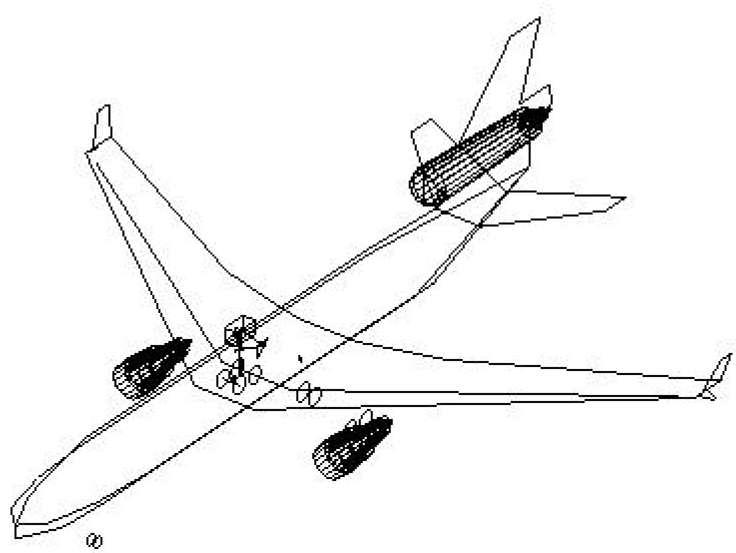

Fig. 2 Airframe – Flexible Wing – Rigid Fuselage & Empennage

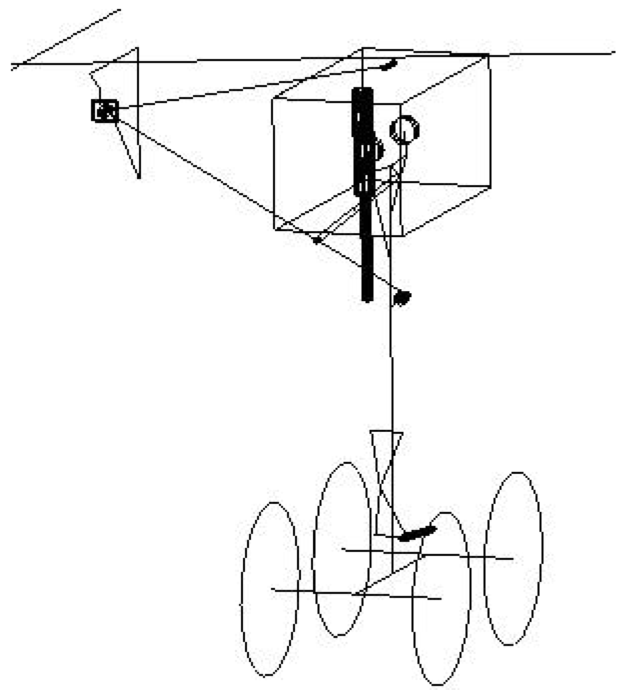

Fig. 3 Right Main landing Gear – ADAMS BEAM Structure

Fig. 4 gives a closeup of the suspect "trap panel" fitting. The two lines projecting to the right

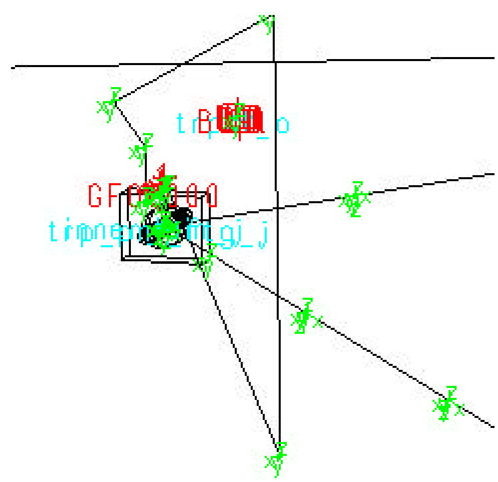

Fig. 4 Trap Panel Fitting Attachment

represent the (inboard) ends of the fixed side brace (upper line) and folding brace (lower line) respectively. These two braces are constrained by a common pin in the center of the trap panel fitting. Lateral loading on the landing gear is reacted by forces entering the folding side brace, coming into the pin, and then being re-transmitted outboard to primary structure near the gear trunnion support. The only loads intended to be reacted at the fitting are vertical. The fitting is secured to the trap panel (the closed polygon in the figure) by two 1-in diameter bolts, the outboard one of which is a snug fit. The two bolts are modeled as a stiff, 6-component spring, located at the bolt pattern centroid.

It should be noted that this (broken) outboard bolt was one of the first pieces of debris located near the right main gear tire skid marks at the crash site.

A “flight path controller” (FPC) was constructed from ADAMS constraint functions to force the model CG to follow the path determined by the aircraft flight data recorder (DFDR). The actual landing simulation is preceded by a positioning phase in which the aircraft model is moved from co-aligned aircraft/Earth coordinates to alignment with runway coordinates. After this initial positioning, the X-, Y-, Z-, and Yaw-, Pitch-, and Roll position of the aircraft is forced onto the model until the second, and critical main wheel contact, after which the aircraft is released from the FPC. In the actual event, the DFDR recorders "dropped out" for a more than a second after the second contact, presumably because of impact effects.

The elevation and heading of the runway are adjustable via constraint functions to allow adjustment of the ground contact point with respect to the DFDR information.

The position and attitude of the aircraft is dictated by the FPC up to the limit of DFDR signal. However, it is possible to release the FPC at any simulation point, after which the path of the aircraft will be determined by the forces acting upon it. Aero forces were added to permit a qualitative estimate of the aircraft behavior after the failure of the DFDR signal. Also, the wing structure in the model is flexible, so that aero loading will effect the wing deflection and, hence, the landing gear strut loads due to the FPC-imposed motion. The aero forces are distributed to 8 points on each (flexible) wing, to 4 points on each (rigid) horizontal stabilizer, and to one point on the (rigid) vertical fin. The lift, drag, and pitching moments for each aero station are computed as an area percentage of the surface as a whole using the classical factors of area, aspect ratio, and mean aerodynamic chord. No attempt is made to vary the downwash distribution along the spans.

The tire forces are applied directly to the axle centers at the wheels. The tires are not modeled as independent parts, thus spin up/down effects are not included. The vertical forces are applied as a bi-linear spring with a stiffness of 18 kip/in up to 10 in. of tire crush, after which the stiffness rises to 201 kip/in, simulating wheel rim contact through the tire carcass to the runway. The tire side force is applied as three components: yaw, tilt (camber), and rim force. The yaw force is input using a manufacturer-supplied spline of force as a function of normal force, slip angle, and velocity. The camber force is a linear function of normal force and camber angle, and the rim force is a function of normal force and scrubbing velocity.

The axial strut force is a nonlinear function of displacement and stroking velocity. It is applied using three manufacturer-supplied spline functions. The piston/cylinder bearing loads are applied using constraints to the piston and cylinder concentric at 2 reference locations. The forces generated by the concentric constraints are used to input a stroking friction force employing a constant coefficient of friction multiplying the instantaneous, radial bearing loads and biased to oppose the stroking velocity.

The manufacturer supplied estimated attachment spring rates for the trap panel and trunnion support structures. These springs are input between the rigid offsets from the wing beams and their respective parts.

Simple, constant-value thrust forces, based on DFDR data, were applied along all three engine centerlines.

Fig. 5 below shows the ADAMS model of the subject aircraft at simulation time t=6.0 sec. The right main landing gear loads are at or near the maximum (over)load condition. The large (downward) force vector represents the reaction force applied by the fuselage trap panel to the trap panel fitting and is consistent with the intended design function of the structure. The large (upward) vertical forces represent the (four) right main tire forces, and the curved vectors represent the overturning moments applied to the outboard tires.

The results from this analysis indicated that the right main landing gear of the subject aircraft had been exposed to loads that were clearly beyond the aircraft's design envelope. As a result of this overload condition, the landing gear support/attachment and wing structure had failed, although the exact failure sequence could not be determined at this point of the accident investigation. This first phase ADAMS analysis indicated that the loads in the trap panel fitting were insufficient to cause the bolt failure evidenced at the crash scene. Examination of the wreckage indicated slight plastic bending of the front outboard axle, which gave a good estimate of the tire loads to which it had been subjected. The tire loads predicted by the analysis were higher than this by approximately 60%. Loads of this magnitude would have failed the bogie structure completely. Speculation arose that, either the failure had initiated elsewhere in the system, or that the coarse beam representation of the airframe was insufficient to model the actual support conditions of the landing gear attachment. In an effort to resolve this issue, the decision was made to replace the airframe model from the initial analysis with a more comprehensive structural representation employing an MSC/NASTRAN-derived ADAMS FLEX_BODY element.

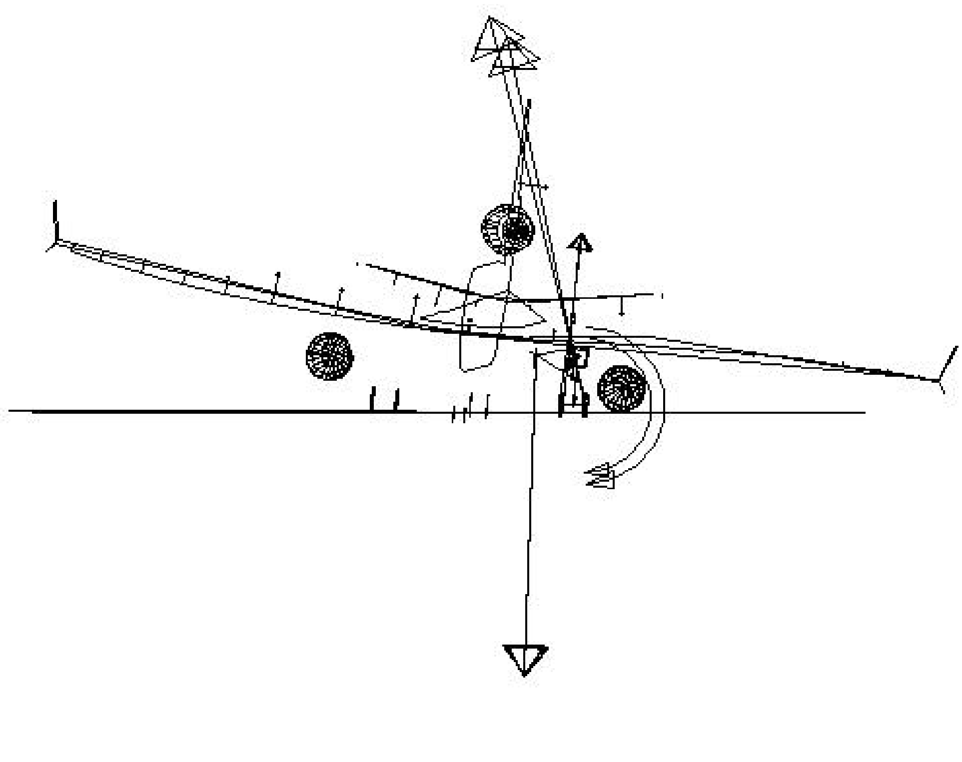

Fig.5 ADAMS Model of Aircraft During the (Critical) 2nd Impact

Enhanced Airframe Model

The MSC/NASTRAN model of the airframe was created from a combination of several different FEA models. Shear panel/rod structures originally analyzed using the manufacturer's FEA program were combined with MSC/NASTRAN beam element models used for flutter analysis to yield the conglomerate analytical model shown in fig. 6 below. This effort required some reconfiguring of the FEA structures since the manufacturer's FEA code permitted the use of isolated mid-side nodes in shear panels, a feature which has no direct MSC/NASTRAN equivalent.

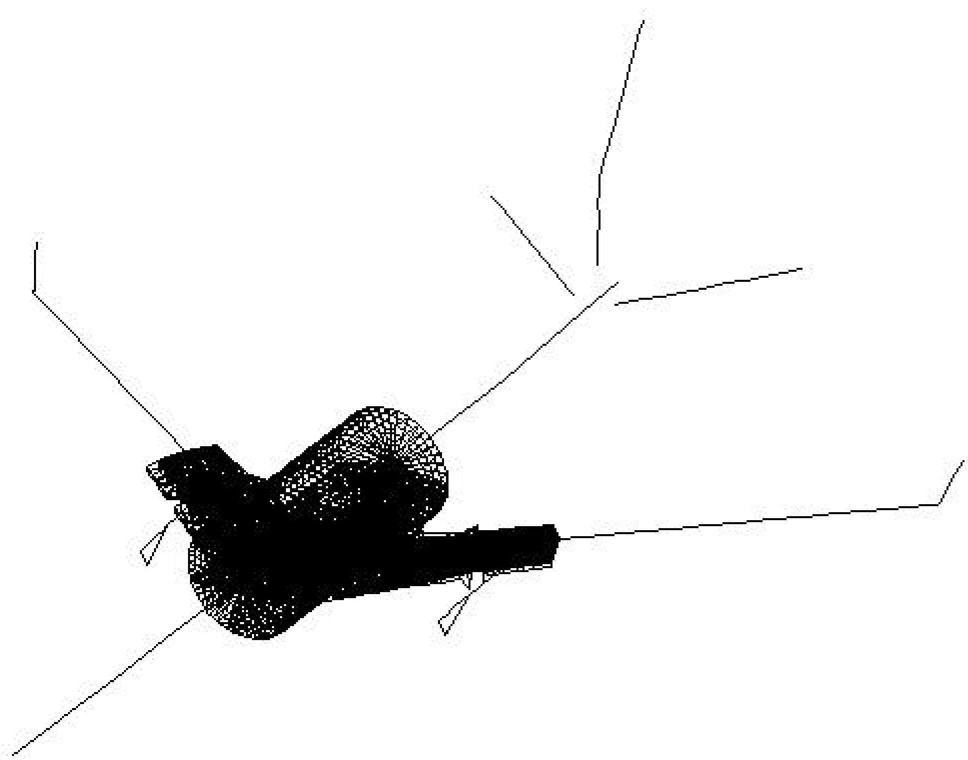

Fig. 6 MSC/NASTRAN Airframe Model

The center fuselage/center and inner wing structures are modeled using (predominantly) shell and beam elements. The forward fuselage, aft fuselage and empennage, and the outer wings are modeled using beam elements, as are the engine pylons. Where necessary, concentrated mass elements are employed to maintain proper mass integrity.

After conversion from the manufacturer's FEA code to MSC/NASTRAN, modal analyses were performed to validate the converted structural models against accepted results for the structure.

ADAMS Modal Flexibility – Overview

The ADAMS method for rendering system components flexible is based on the Craig-Bampton approach [2]. Any point (e.g., GRID point or node) in a structural component which is 'touched' by something else in the system is considered a 'hardpoint' and is declared a master node in an FEA static, superelement condensation. In addition to those degrees of freedom associated with the selected hardpoints, additional, modal degrees of freedom are added from an eigenvalue analysis performed with the hardpoints globally fixed. The possible structural deformations for the flexible component in question consist of a combination of the geometrically orthogonal Cartesian coordinates at the 'hardpoints' with the normalized, modal, eigencoordinates. The method restricts the behavior of the structure to the linear regime. Specifically, the deflections internal to the modal structure must be small (linear geometrical assumption) and the stresses resulting from these deformations must be non-plastic (linear material assumption). Within these restrictions, the linear structural system can be incorporated into a mechanical system simulation (MSS) undergoing gross, nonlinear motion. It is also assumed that 2nd order effects such as centripetally-induced stress-stiffening are absent. Special methods can be employed with ADAMS flexible bodies to remove the linear geometrical and stress-stiffening restrictions, but they were not deemed necessary for this analysis. Details on the general implementation of flexible components in ADAMS are given in [4].

The end result of the conversion process in this case is a structure that is condensed down from 50,000+ Cartesian coordinates to approximately 120 orthogonal coordinates.

Special ADAMS/FLEX Application

In the general case, the elastic content of the flexible structure does not, and usually cannot, change during the simulation. In this instance, however, it was desired to investigate potential failure sequences. It is known from the crash site investigation that the rear spar web on the subject aircraft failed between the right main gear trunnion (or bulkhead) rib and the wing root (between wing stations 264 and 118). This failure occurred either prior to, or as a consequence of, the collapse of the right main landing gear. To represent this behavior, the airframe model was modified by, in effect, cutting the shear web (ref. fig. 7) in the described location vertically just inboard of the trunnion rib and horizontally from the trunnion rib to the root at the web attachment to the upper and lower spar caps. The 'cuts' introduced at these locations were then closed in the ADAMS analysis using very stiff springs across the (coincident) grids defining the element boundaries. This represents a somewhat unique use of the Craig-Bampton method in that system eigenmodes are combined with external restraints to modify the system behavior. In effect, two different, linear structures, one 'uncut' and the other 'cut', are obtained from a single, linear, self-modifying structure.

The fuse spring must be recursively triggered, i.e., it must turn off for all subsequent time, once the critical force level has been reached. To do this, a user-specified differential variable (an ADAMS DIFF function) is specified which is forced to have a zero value until the triggering force level is exceeded. The value of the differential variable is ramped from zero to one over a user-specified load tolerance. This DIFF variable is used in an ADAMS STEP function to multiply the fuse spring expression, which 'closes the loop' between the fuse trigger and the force expression. As long as the DIFF value remains zero, the STEP multiplier has a value of one. As soon as the DIFF becomes greater then zero, the STEP multiplier goes to zero.

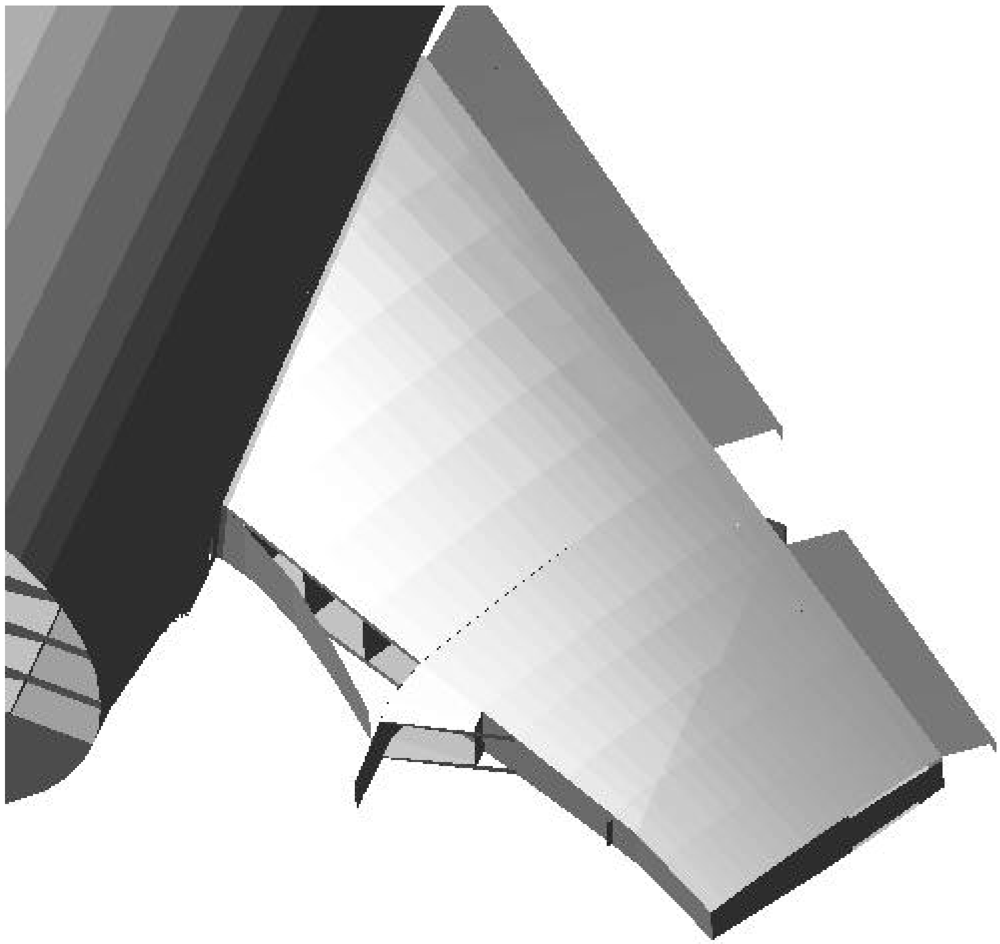

Fig. 7 Released Spar Web

A similar artifice was also employed to fuse the trap panel fitting to the airframe trapezoidal panel support structure. Use of the CAD database for the aircraft indicated that, should the fixed side brace pivot about its pin and reach a 'prying angle' of more than 9.1 degrees, its clevis would bind on the trap panel fitting and induce unintended bending loads in the brace. Examination of the wreckage showed that such binding had, indeed, occurred. Further, the fixed side brace had been plastically deformed in bending by a load applied upward and backward. Classical beam analysis indicated what load was necessary to cause the onset of plastic deformation, and this load was used to fuse the fitting.

Prior to employing the model in analysis, its modal characteristics were validated against accepted characteristics for the aircraft.

Analysis Results – Enhanced Model

Analysis of the enhanced model indicated that the second impact on the right main gear resulted in an overload sufficient cause the strut to bottom. The corresponding loads in the rear spar web of the wing were sufficiently above the ultimate design limit to cause failure of the web structure. Subsequent to the failure, the right wing twists substantially nose-down under the imposed loads.

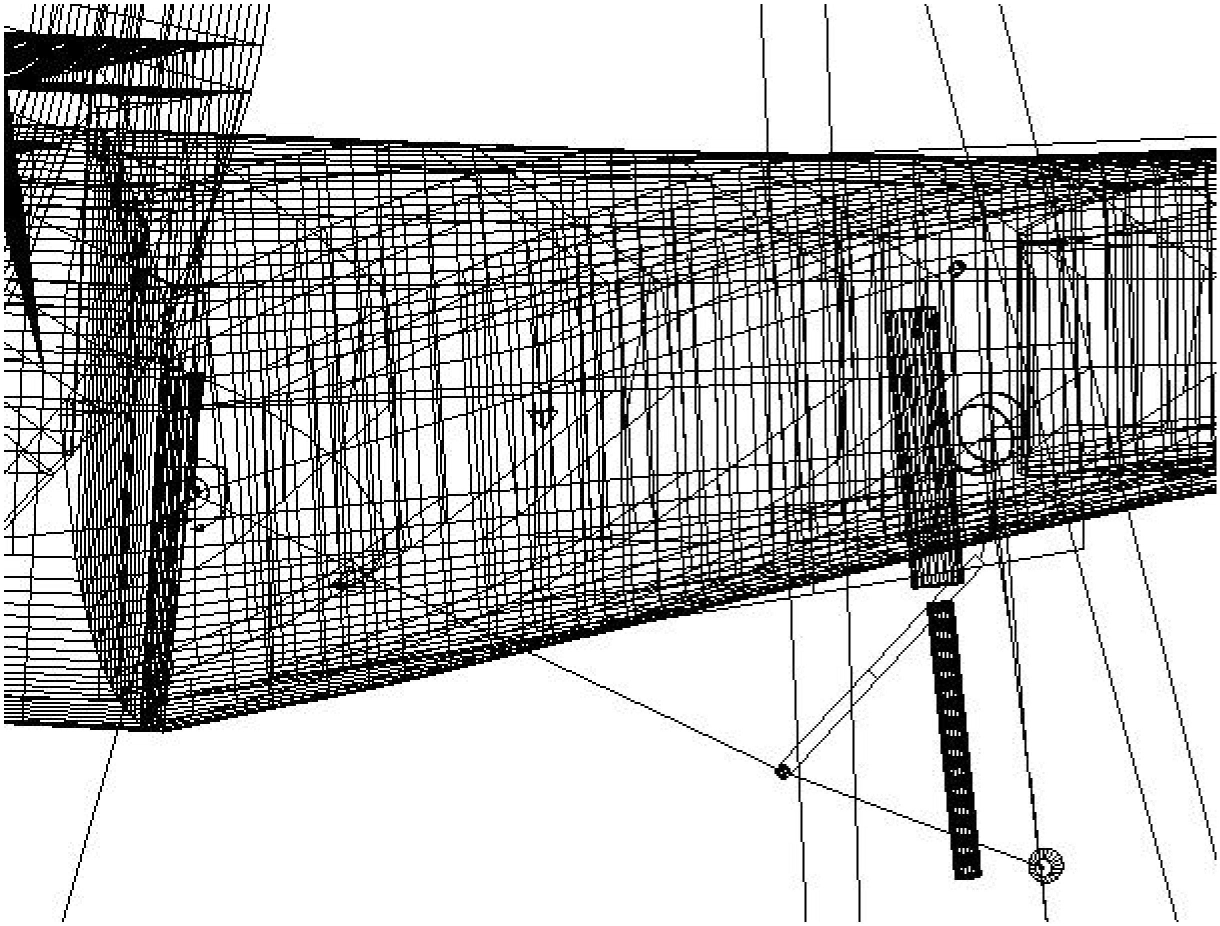

Fig. 8 Spar Web Fracture

In fig. 8 the bottom edge of the failed (hence unloaded) spar web can be seen protruding below the wing surface, which has twisted nose-down. In effect, the wing torque box has become an exaggerated C-section, with the shear center ahead of the front spar. This twisting causes the right wing to 'dump' most of its lift and results in a sudden and substantial outboard motion of the right main gear bogie, resulting from the fixed and folding landing gear side braces pivoting about their (common) attachment at the trap panel fitting (ref. fig. 9). These results are consistent with the DFDR data showing that, at this impact, the aircraft continued to roll to the right, in spite of control inputs commanding a roll to the left. Fig. 10 shows a trace of the paths of the inner and outer front tires, starting at the initial contact of the forward outboard tire and ending just after the fracture. These traces correlate very closely with the skid marks for the right main gear documented at the crash site, inclusive of the outboard jink when the spar web fractures. Further, the right front tire loads were very close to those required for the plastic axle deformation seen at the crash site.

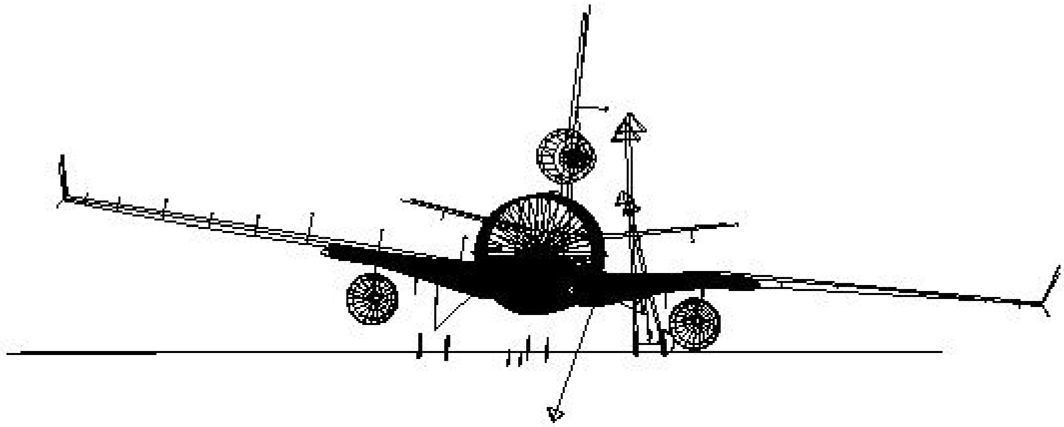

Fig. 9 Aerodynamic Unloading and Lateral Gear Displacement due to Fracture

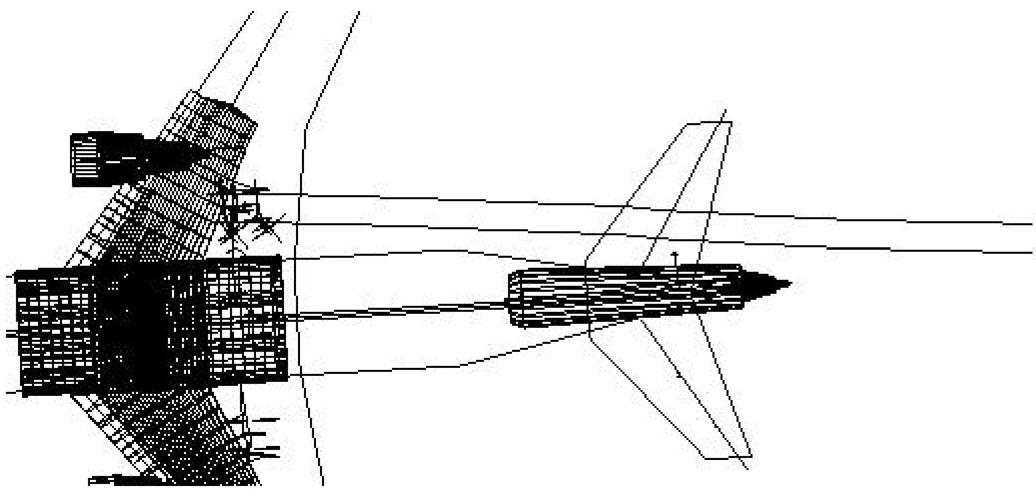

Fig. 10 Initial Tire Traces After Touchdown

The analysis also indicated that, even with a failed spar web, the wing twist was not sufficient to reach the prying angle in the trap panel fitting necessary to fail it. This implied that the actual wing failure was more extensive, and modifications to the failure paths in the model would be needed.

Multiple Substructures – 3-Piece Airframe Model

To permit a more extensive breakup, the airframe structure was broken into 3 components. Figs. 11, 12, and 13 show the airframe, right wing, and pylon substructures, respectively. As in the one-

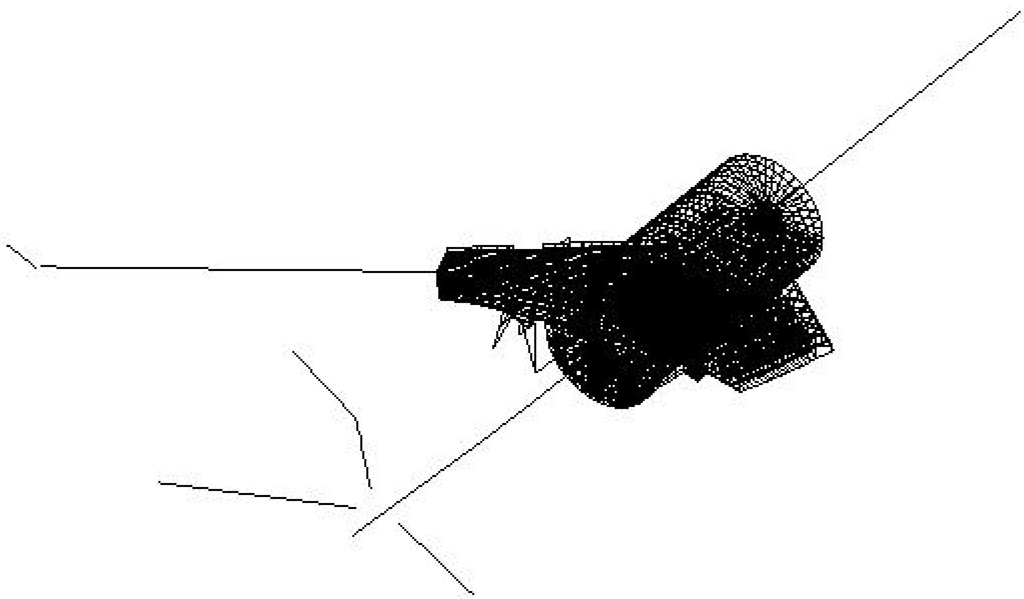

Fig. 11 Primary Airframe Substructure

Fig. 12 Right Wing Substructure

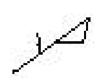

Fig. 13 Right Engine Pylon Substructure

piece airframe model, the spar web of the right wing was cut to simulate a failure from the trunnion rib to the root. In addition, the right wing structure was modeled to separate at the trunnion rib, with the rib remaining with the wing. This failure path was closed by stiff springs, which were fused to fail at the same time as the rear spar web. The right pylon was attached to the wing at four connection points, and the rear connection could be fused to fail should the right nacelle contact the ground and develop a sufficient load, permitting the engine/nacelle assembly to pivot about the front connections and allow the wing to drop to the ground. To reduce modeling turnaround time, the ADAMS RESTART feature was used to begin the analysis at time t=5.0 sec, i.e., just prior to the critical, second touchdown. Fig. 14 shows the state of the model at this point. Subsequently, the model was restarted and the simulation extended to time t=6.68 sec.

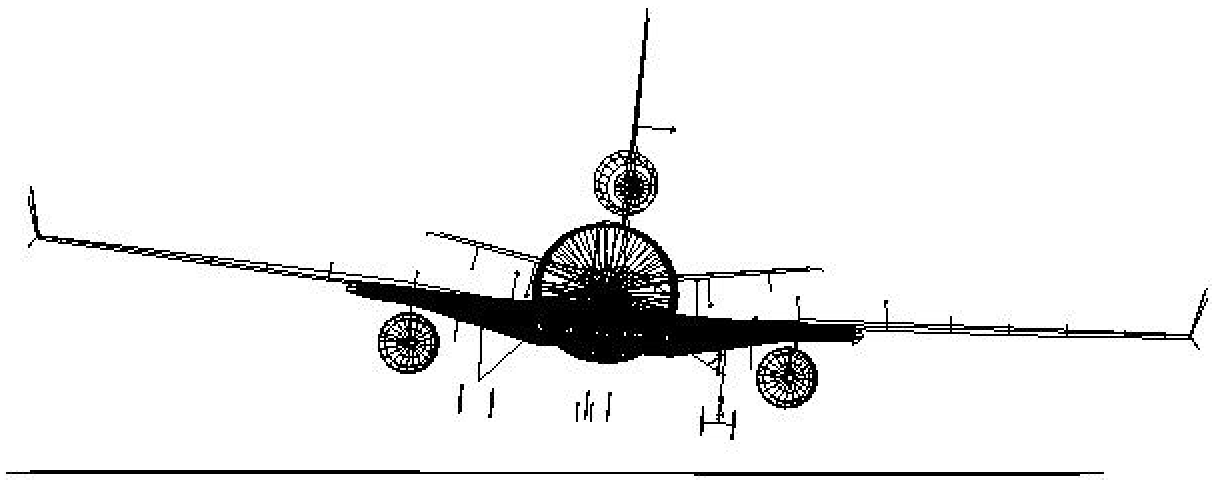

Fig. 14 Model State at Time t=5.0 sec

(maximum). The second impact of the right main gear initiated at 5.8+ simulation seconds. Analysis indicated that, again, the right main gear was impacted with sufficient force to bottom the strut, and that structural failure due to the overload occurred shortly thereafter. As with the one-piece model, when the wing structure fractures, the right main gear bogie moves rapidly outboard. Fig. 15 shows the model at simulation time t=6.4 sec, by which point the right wing has separated completely. At this point in the simulation, the right main gear trunnion rib has moved vertically and aft by a sufficient amount to break the trap panel fitting loose from its attachment to the airframe. It is clear that the fixed side brace is bent upward and aft in the model, which is consistent with the damage to the component as found at the crash site.

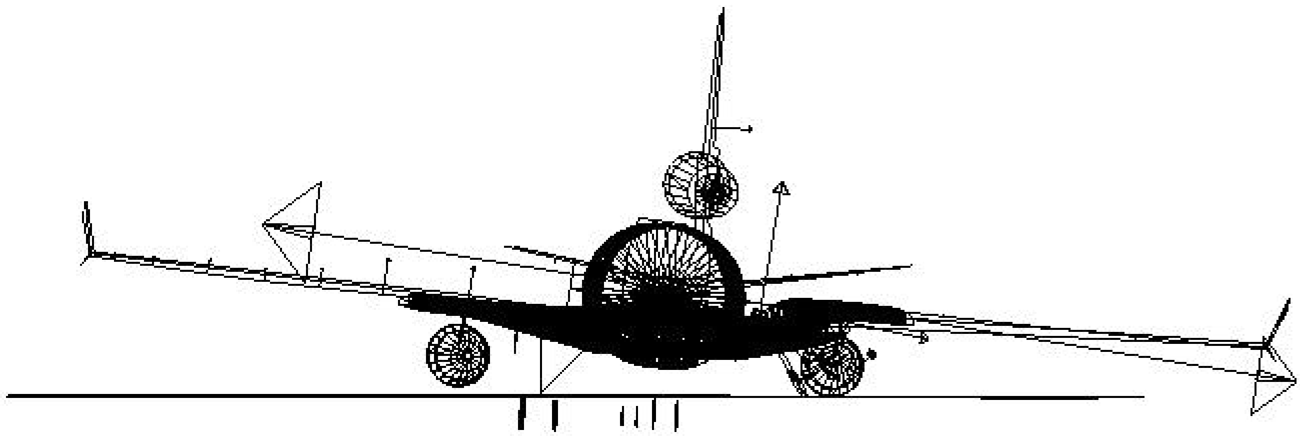

Fig. 15 Model at Time t=6.4 Sec.

Conclusions

The use of MSC/NASTRAN in conjunction with ADAMS proved very useful in determining the nature and sequence of the structural failures that occurred in the landing incident described here. The ADAMS model incorporating the MSC/NASTRAN-derived flexible structures yielded results that closely correlated with, both, evidence at the crash scene and general structural/mechanical behavior predicted by other software used in the certification of the subject aircraft or taken from actual structural tests. The analysis was able to determine that, due to loads well beyond the ultimate limits to which the aircraft had been designed, the failure of the structure had originated at the rear spar web of the wing and not, as originally thought, at the inboard trap panel fitting. The use of non-linear, external forces at the boundaries of linear elastic structures was successful in producing different, linear states from a single linear model.

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