NASA Technical Memorandum 101543

STRUCTURAL DYNAMICS DIVISION RESEARCH AND TECHNOLOGY ACCOMPLISHMENTS FOR FY 1988 AND PLANS FOR FY 1989

(NASA-TM-101543) STRUCTURAL DYNAMICS DIVISION RESEARCH AND TECHNOLOGY ACCOMPLISHMENTS FOR FY 1988 AND PLANS FOR FY 1989 (NASA) 187 p CSCL 20D
N89-15895
Unclas
G3/02 0189698

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January 1989

National Aeronautics and
Space Administration

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STRUCTURAL DYNAMICS DIVISION
RESEARCH AND TECHNOLOGY ACCOMPLISHMENTS FOR FY 1988 AND PLANS FOR FY 1989

SUMMARY

The purpose of this paper is to present the Structural Dynamics Division's research accomplishments for FY 1988 and research plans for FY 1989. The work under each branch (technical area) is described in terms of highlights of accomplishments during the past year and highlights of plans for the current year as they relate to five year plans and the objectives for each technical area. This information will be useful in program coordination with other government organizations, universities, and industry in areas of mutual interest.

ORGANIZATION

The Langley Research Center is organized into directorates as shown in figure 1. Directorates are subdivided into divisions and offices. The Structural Dynamics Division of the Structures Directorate consists of five branches as shown on figure 2. This figure lists the key people in the division which consists of 66 NASA civil servants and 12 members of the Army Aerostructures Directorate, USAARTA, Army Aviation Systems Command co-located at the Langley Research Center. Recent changes in key positions include the selection of Mr. Irving Abel as Chief, the appointment of Dr. Larry D. Pinson as Assistant Chief, the selection of Mr. Rodney H. Ricketts as Head of the Configuration Aeroelasticity Branch, the appointment of Mr. Huey D. Carden as Assistant Head of the Landing and Impact Dynamics Branch, and the selection of Dr. Thomas E. Noll as Head of the Aeroservoelasticity Branch. Each branch represents a technical area and disciplines under the technical areas are shown in the figure.

The division conducts analytical and experimental research in the five technical areas to meet technology requirements for advanced aerospace vehicles. The research focuses on the long range thrusts shown in figure 3. The Configuration Aeroelasticity Branch (CAB), Unsteady Aerodynamics Branch (UAB), and Aeroservoelasticity Branch (ASEB) all work in the area of the prediction and control of aeroelastic stability and response of aircraft and rotorcraft. The Landing and Impact Dynamics Branch (LIDB) conducts research on the crash dynamics of aircraft structures and on the technology for improving the safety and handling performance of aircraft during ground operations. The Spacecraft Dynamics Branch (SDB) conducts research on the prediction and control of the structural dynamic response of complex space structures.

FUNCTIONAL STATEMENT

The Division conducts analytical and experimental research in the areas of aeroelasticity, aeroservoelasticity, unsteady aerodynamics, impact and landing dynamics, and spacecraft dynamics to meet technology requirements for advanced atmospheric and space flight vehicles. Develops analytical and computational methods for predicting and controlling aeroelastic instabilities, deformations, vibrations, and dynamic response. Investigates interaction of structure with aerodynamics and control systems, landing dynamics, impact dynamics, and resulting structural response. Evaluates structural configurations embodying new material systems and/or advanced design concepts for general application and for specific classes of new aerospace vehicles. Uses a broad spectrum of test facilities to validate analytical and computational methods and advanced configuration and control concepts. Develops research techniques to demonstrate safety from aeroelastic instabilities for new airplanes, helicopters, and space launch vehicles. Test facilities include the Transonic Dynamics Tunnel, the General Rotor Aeroelastic Laboratory, the Impact Dynamics Research Facility, the Aircraft Landing Dynamics Facility, and the Structural Dynamics Research Laboratory.

FACILITIES

The Structural Dynamics Division has four major facilities available to support its research as shown in figure 4.

The Transonic Dynamics Tunnel (TDT) is a maximum Mach 1.2 continuous flow, variable-pressure wind tunnel with a 16-foot-square test section which uses air or a heavy gas (R-12) as the test medium. The maximum Reynolds number obtainable is approximately 10 million per foot in heavy gas and 3 million per foot in air. This unique "National" facility is used almost exclusively for testing of aeroelastic phenomena. Semi-span, side-wall mounted models and full-span sting mounted or cable-mounted models are used for aeroelastic studies of fixed wing aircraft. In addition, the Aeroelastic Rotor Experimental System (ARES) test stand is used in the tunnel to study the aeroelastic characteristics of rotors. The General Rotor Aeroelastic Laboratory, located in an adjacent building, is used to setup the ARES test stand in preparation for entry into the TDT and for rotorcraft studies in hover. A new TDT Data Acquisition System became operational in FY 1988.

The Aircraft Landing Dynamics Facility (ALDF) is capable of testing various types of landing gear systems at velocities up to 220 knots on a variety of runway surfaces under all types of simulated weather conditions. The ALDF consists of a 2800 ft. long rail system, a 2.0 million pound thrust propulsion system, a test carriage, and an arrestment system. Test articles can be subjected to vertical loads up to 65,000 lbs or sink rates of 20 ft/sec on a wide variety of runway surface conditions. The facility provides for testing at speeds and sizes pertinent to large transport aircraft, fighter aircraft, and the Space Shuttle Orbiter.

The Impact Dynamics Research Facility is capable of crash testing full-scale general aviation aircraft and helicopters under controlled conditions. The facility is a 220 feet high, 400 feet long gantry structure which is the former Lunar Landing Facility. General aviation aircraft and helicopters weighing up to 20,000 lbs can be tested up to 60 mph using a free-swinging pendulum approach or up to 100 mph with rocket assist. Attitudes can be adjusted for desired pitch, roll, and yaw parameters. Impact surfaces can be concrete or dirt. High speed motion pictures and 90 data channels are available to record the crash event. A vertical test apparatus is attached to one leg of the facility for drop-testing structural components. The facility is used to support in-house research and other agency programs (Army, Air Force, FAA).

The Structural Dynamics Research Laboratory (SDRL) is designed for conducting research on the dynamic and control response of spacecraft structures. The facilities consist of the 16 meter Thermal Vacuum Chamber, Main Backstop Area, and Large Component Test Room. These facilities provide a variety of environmental simulation capabilities, including acceleration, vacuum and thermal radiation. The Chamber has a 55-foot diameter, hemispherical dome with a 64-foot high peak, flat floor and option for a large centrifuge or a rotating platform. Access is by an airlock door and an 18-foot by 20-foot test specimen door. A vacuum level of 10 torr can be achieved within 120 minutes and, with diffusion pumps, 10^{-4} torr vacuum can be achieved within 160 minutes. A temperature variation of 100°F can be obtained in the chamber by using 250-sq. ft. of portable radiant heaters and liquid nitrogen cooled-plates. The Backstop Area is dominated by the 38-foot-high back-stop of I-beam construction. Test areas around this fixture are 15 x 35 x 38 feet high and a tower 12 x 12 x 95 feet high. The Large Component Test Room is an open room with full environmental control system and nominal dimensions of 75 x 84 x 798 feet high. There are various size hoists and accessible platforms for suspension systems, instrumentation, and viewing. Closed circuit television is available for monitoring research studies. Test articles can be excited by several types of actuators and small shakers. State-of-the-art capability is available for signal conditioning and processing including GenRad 2515 digital signal processing systems and a VAX 11-780/EAI 2000 hybrid computer system for simulation and on-line test control. A number of improvements are presently being made that will benefit controls/structures interaction research.

FY 1988 ACCOMPLISHMENTS

Configuration Aeroelasticity Branch

The Configuration Aeroelasticity Branch conducts research (figure 5) to develop the aeroelastic understanding and prediction capabilities needed to apply new aerodynamic and structural concepts to future flight vehicles and to determine and solve the aeroelastic problems of current designs; to evaluate the aeroelastic characteristics of new rotor systems; and to determine, analytically and experimentally, effective means for predicting and reducing helicopter vibrations. This work is more clearly identified in figure 6 which shows the five-year plan of the three disciplines and their expected results.

The Configuration Aeroelasticity FY 1988 accomplishments listed below are highlighted in figures 7 through 25.

Aircraft Aeroelasticity:

- Effects of Span Reduction on Flutter of Arrow Wing SST Configurations Determined in TDT
- Flutter Characteristics of Supersonic Cruise Configurations Determined in TDT
- Effects on F-16 of New Composite Leading Edge Flaps and New Air Defense Pylons Studied in TDT
- Flutter Characteristics of Wing Tip Geometry Studied in the Vigyan Wind Tunnel
- Flutter Characteristics and Boundaries Defined for Highly Swept Delta Wings
- Temperature Effects Integrated Into Finite Element Models
- Transonic Flutter Characteristics of Advanced Composite A-6 Replacement Wing Determined in TDT
- Milstar Radome Tested for Panel Flutter in Transonic Dynamics Tunnel
- Unusual Torsion Instability Excited By Spoiler Surfaces
- Atlas-Centaur Large Payload Fairing Model Indicates Flight Vehicle Will Be Free of Aeroelastic Problems
- Supercritical Wing Tested for Flutter on PAPA in the TDT
- Laser Light Sheet Flow Visualization System Developed for TDT
- PAPA - A New Wind-Tunnel Mount System for Flutter Research

Rotorcraft Aeroelasticity:

- Passive Blade Twist Control Improves Performance of Tilt-Rotor Vehicles
- TDT Oscillating Flow Field Measured for Helicopter Rotor Gust Studies
- Motorized Pitch Link Developed for Improved Blade Tracking in TDT
- Forward Flight Rotor Tracking Characteristics Investigated in TDT
- TDT Tests Evaluate Performance Characteristics of Advanced Design Helicopter Rotor Blades

Rotorcraft Structural Dynamics:

- Optimization Approach for Helicopter Vibration Reduction Demonstrated
- Ground Vibration Test of Helicopter Airframe Identifies Important Contributors to Vibratory Response

Unsteady Aerodynamics Branch

The Unsteady Aerodynamics Branch conducts research (figure 26) to produce, apply and validate through experiments a set of analytical methods for predicting steady and unsteady aerodynamic loads and aeroelastic characteristics of flight vehicles--with continued emphasis on the transonic range and emerging emphasis on high angle maneuvering conditions. Considerations of dynamic vortex-structure interactions, dynamic loads and buffet are becoming major areas of interest. This work is more clearly identified in figure 27 which shows the five year plan of the branch for theory development, experiments, and design methods.

A major accomplishment of the year was the release to U.S. industry of the CAP-TSD (Computational Aeroelasticity Program - Transonic Small Disturbance) code. The code has been extensively modified to operate on CRAY X-MP computers with Solid State Disc storage units, and the applications given in the following figures give an indication of its ability to treat a number of situations. The other major accomplishment has been the development of higher equation level methods--unsteady Euler equations solver implemented for dynamic unstructured grids.

The Unsteady Aerodynamics FY1988 accomplishments listed below are highlighted in figures 28 through 37.

Methods Development:

- Solid State Disc Version of CAP-TSD Developed
- Strip Boundary Layer Capability Improves Accuracy of CAP-TSD Results
- Supersonic Far-Field Boundary Conditions Improve Accuracy and Efficiency for Aeroelastic Calculations
- Unstructured Dynamic Grid Method Developed for Aeroelastic Analysis with High Level CFD Codes
- Unsteady Euler Algorithm Developed Based Upon Dynamic Unstructured Grid Methodology

Applications:

- Wing Torsional Flutter Features Studied with CAP-TSD
- Rigorous Calibration of Flutter Analysis Methods for Slender Delta Wings
- Supersonic Flutter of F-20 Horizontal Tail Model Accurately Predicted
- TSD Potential Code Predicts P-80 Aileron Buzz
- Analysis of NTF Arc-Sector/Fixed Fairing Indicates Hump Mode Flutter

Aeroservoelasticity Branch

The Aeroservoelasticity Branch (figure 38) conducts research to develop methodologies for the modeling, the analysis, and the synthesis of multifunctional active control systems; conceives, recommends and performs experiments to validate the methodologies; and provides technical support to major NASA flight projects. The scope of this work is more explicitly identified in figure 39 which shows the five year plan of the three major thrusts and their expected results.

The Aeroservoelasticity FY 1988 accomplishments listed below are highlighted in figures 40 through 49.

Analysis and Modeling:

- Minimum-State Approximations of Unsteady Aerodynamics Permits Large Order Reduction of Aeroservoelastic Equations
- Overlap Between SDG and PSD Gust Analysis Methods Established

- Matched Filter and Random Process Theories Provide Efficient Options for Determining Maximized Time-Correlated Gust Loads
- Correction Factor Methodologies Developed to Improve Predictions of Unsteady Aerodynamics

Control Law Synthesis:
- Design of Digital Multi-Input/Multi-Output FSS Obtained for AFW Model
- Stability Robustness Improved Using Singular Value Constraints
- Analytical Sensitivities Improve Integrated Structure/Control Law Methodology

Applications and Validations:
- AFW Flutter Boundary Lowered by Addition of "Tip Missile"
- AFW Control System Hardware Layout
- Analysis Establishes Guidelines for Avoiding Aeroelastic Instabilities of X-Wing Aircraft

Landing and Impact Dynamics Branch

The Landing and Impact Dynamics Branch conducts research (figure 50) utilizing two major facilities, the Aircraft Landing Dynamics Facility (ALDF) and the Impact Dynamics Research Facility (IDRF). The landing dynamics group conducts research to advance technology for safe, economical all-weather aircraft ground operations including the development of new landing gear systems. The group coordinates in-house research, grants, and contracts with U.S. tire industry to achieve the technology required. The impact dynamics group conducts research to obtain a better understanding of response characteristics of generic composite aircraft components subjected to crash loading conditions and to develop/enhance analytical tools capable of predicting response of composite structures. In-house research, grants and contracts are also utilized to achieve the technology to better understand impact dynamics and to develop better structural concepts capable of providing energy absorption and reduced crash loads. The work of the Landing and Impact Dynamics Branch is more clearly identified in figure 51 which shows the five-year plan of the disciplines in both landing and impact dynamics along with their expected results.

The Landing and Impact Dynamics Branch FY 1988 accomplishments listed below are highlighted in figures 52 to 62.

Landing Dynamics:
- 30 x 11.5 - 14.5, Type Radial Aircraft Tire Program
- Corduroy Texture Identified as Solution to Spin-Up Wear Damage Problem At the Kennedy Space Center Shuttle Landing Facility
- Orbiter Flat-Tire Landing Model Is Developed
- Scale Model Test Guide Shuttle Net Arrestment System Development
- F-106B Airplane Active Control Landing Gear Program Nose Gear Drop Tests
- Runway Friction Workshop

Impact Dynamics:

- Static Response of Composite Fuselage Floor Sections
- Model Development For Analysis of Thin-Walled Beams
- Energy Absorbing Characteristics of Composite Subfloor Intersections
- Scaling Effects in the Large Deformation Bending Response of Composite Beams
- Interim Transportation Overpack Container (ITOC) Experiments and Analysis

Spacecraft Dynamics Branch

The Spacecraft Dynamics Branch conducts research (figure 63) on the dynamics and control of advanced spacecraft. Analytical methods are developed and verified to advance the state-of-the-art in large flexible complex space structures such as the U. S. Space Station, earth observation platforms and large area antennas. During the past year a large flexible space truss was analyzed and tested in the laboratory yielding good correlation with analysis and establishing present limitations in test and analysis of very low frequency space structures with closely spaced modes. The hardware for researching multibody slewing was assembled this year and analytical simulations were performed. Also, a multidisciplinary analytical research tool for linear coupled structure and controls simulation was completed and a second research tool for nonlinear coupled kinematic and structural deformation code saw the release of its alpha version. In addition, hybrid scaling laws were developed and verified analytically to allow the construction of a sub-scale space station model which will be used to verify dynamics analysis of a complex space structural system. The design and fabrication of this system is now underway. Finally, a major activity involving the advancement of ground test methods for control/structure interaction was initiated along with laboratory upgrading necessary to accomplish this goal. The scope of this work is more explicitly identified in figure 64 which shows the five year plan of the three major thrusts and their expected results.

The Spacecraft Dynamics Branch FY 1988 accomplishments listed below are highlighted in figures 65 through 72.

Ground Test Methods:

- Results of 20 Meter Mini-Mast Ground Test Program
- Hybrid Scaling Laws Developed and Validated for Dynamically Scaled Space Station Model

Optimal Spacecraft Performance:

- Structural Tailoring Objective Identified for Minimization of Controller Energy in Active Structures
- Completion of Integrated Multidisciplinary Research Tool (IMAT)
- Effect of Active Truss-Bay Location on Vibration Suppression

Multibody Dynamics:
- Nonlinear Analysis Corrects Conventional Slewing Predictions
- IMAT Contributions to Space Station Level 2 November Reference Data Book
- Terminal Control of Multibody Maneuver

PUBLICATIONS

The FY 88 accomplishments of the Structural Dynamics Division resulted in a number of publications. The publications are listed below by organization in the categories of journal publications, formal NASA reports, conference presentations, contractor reports, tech briefs, and patents.

Division Office

Journal Publications:

1. Abel, I.: Filling the Expertise Gap. Aerospace America, Vol. 26, No. 8, August 1988, p. 16-17.

Formal NASA Reports:

2. Dixon, S. E.; and Gardner, J. E.: Loads and Aeroelasticity Division Research and Technology Accomplishments for FY 1987 and Plans for FY 1988. NASA TM-100534, January 1988.

3. Pinson, L. D.: Recent Advances in Structural Dynamics of Large Space Structures. NASA TM-100513, October 1987.

Conference Presentations:

4. Abel, I.; and Noll, T. E.: Research and Applications in Aeroservoelasticity at the NASA Langley Research Center. Presented at the 16th Congress of the International Council of the Aeronautical Sciences (ICAS), August 28-September 2, 1988, Jerusalem, Israel. ICAS Paper No. 88-5.7.1.

5. Doggett, R. V., Jr.; and Cazier, F. W., Jr.: Aircraft Aeroelasticity and Structural Dynamics Research at the NASA Langley Research Center - Some Illustrative Results. Presented at the 16th Congress of the International Council of the Aeronautical Sciences (ICAS), August 28 - September 2, 1988, Jerusalem, Israel. Paper No. US-22, Session 5.7.2. Also available as NASA TM-100627.

6. Pinson, L. D.: Recent Advances in Structural Dynamics of Large Space Structures. Presented at the Thirty-Eighth International Astronautical Federation (IAF) Congress, October 10-17, 1987, Brighton, England. Paper No. IAF-87-51. Also available as NASA TM-100513.

7. Sliwa, S. M.; and Abel, I.: Overview of Dynamics Integration Research in Progress at the Langley Research Center. Presented at the Second NASA /Air Force Symposium on Recent Experiences in Multidisciplinary Analysis and Optimization, September 28-30, 1988, Hampton, Virginia.

Configuration Aeroelasticity Branch

Journal Publications:

8. Cole, S. R.: Divergence Study of a High-Aspect-Ratio, Forward Swept Wing. Journal of Aircraft, Vol. 25, No. 5, May 1988, p. 478-480.

Formal NASA Reports:

9. Durham, M. H.; Cole, S. R.; Cazier, F. W., Jr.; Keller, D. F.; Parker, E. C.; Wilkie, W. K.; and Doggett, R. V., Jr.: Parametric Flutter Studies of an Arrow-Wing Configuration - Some Early Results. NASA TM-100608, June 1988.

Conference Presentations:

10. Bohlmann, J. D.; Eckstrom, C. V.; Weisshaar, T. A.: Aeroelastic Tailoring for Oblique Wing Lateral Trim. Presented at the AIAA, ASME, et al., 29th Structures, Structural Dynamics and Materials Conference, April 18-20, 1988, Williamsburg, Virginia. AIAA Paper No. 88-2263-CP.

11. Eckstrom, C. V.; Seidel, D. A.; and Sandford, M. C.: Unsteady Pressure and Structural Response Measurements on an Elastic Supercritical Wing. Presented at the AIAA, ASME, et al., 29th Structures, Structural Dynamics and Materials Conference, April 18-20, 1988, Williamsburg, Virginia. AIAA Paper No. 88-2277-CP. Also available as NASA TM-100591.

12. Lake, R. C.; and Nixon, M. W.: A Preliminary Investigation of Finite-Element Modeling for Composite Rotor Blades. Presented at the University of Maryland and the American Helicopter Society Second International Conference on Rotorcraft Basic Research, February 16-18, 1988, College Park, Maryland. In Proceedings. Also available as NASA TM-100559 and AVSCOM TM 88-B-001.

13. McMasters, J. J.; Roberts, W. H.; Payne, F. M.; Sandford, M. C.; and Durham, M.: Recent Air-Freon Tests of a Transport Airplane in High Lift Configurations. Presented at the AIAA 15th Aerodynamic Testing Conference, May 18-20, 1988, San Diego, CA. AIAA Paper No. 88-2034.

14. Murthy, T. Sreekanta: Optimization of Helicopter Airframe Structures for Vibration Reduction-Considerations, Formulations and Applications. Presented at the AIAA Aircraft Design, Systems and Operations Meeting, September 7-9, 1988 in Atlanta, Georgia. AIAA Paper No. 88-4422.

15. Murthy, T. S.; Kvaternik, R. G.: Studies on Large Scale Optimization of Helicopter Airframes for Vibration Reduction. Presented at the Second NASA/Air Force Symposium on Recent Experiences in Multidisciplinary Analysis and Optimization, September 28-30, 1988, Hampton, Virginia.

16. Nixon, M. W.: Improvements to Tilt Rotor Performance Through Passive Blade Twist Control. Presented at 1988 Army Science Conference, June 21-24, 1988, West Point, NY. Also available as NASA TM-100583 AVSCOM TM 88-B-010.

Tech Briefs:

17. Farmer, M. G.: Hanging Windmills From Cables. NASA Tech Brief LAR-13434.

Unsteady Aerodynamics Branch

Journal Publications:

18. Batina, J. T.: Efficient Algorithm for Solution of the Unsteady Transonic Small-Disturbance Equation. Journal of Aircraft, Vol. 25, No. 7, July 1988, p. 598-605.

19. Gallman, J. W.; Batina, J.T.; and Yang, T. Y.: Computational Transonic Flutter Boundary Tracking Procedure. Journal of Aircraft, Vol. 25, No. 3, March 1988, p. 263-270.

20. Hafez, M. M.; Whitlow, W., Jr.; and Osher, S. J.: Improved Finite-Difference Schemes for Transonic Potential Flow Calculations. AIAA Journal, Vol. 25, No. 11, November 1987, p. 1456-1462.

21. Howlett, J. T.: Efficient Self-Consistent Viscous-Inviscid Solutions for Unsteady Transonic Flow. Journal of Aircraft, Vol. 24, No. 11, November 1987, p. 737-744.

Formal NASA Reports:

22. Batina, J. T.; Bennett, R. M.; Seidel, D. A.; Cunningham, H. J.; and Bland, S. R.: Recent Advances in Transonic Computational Aeroelasticity. NASA TM-100663, September 1988.

23. Whitlow, W., Jr.: Computational Unsteady Aerodynamics for Aeroelastic Analysis. NASA TM-100523, December 1987.

Conference Presentations:

24. Batina, J. T.: Unsteady Transonic Algorithm Improvements for Realistic Aircraft Applications. Presented at the AIAA 26th Aerospace Sciences Meeting, January 11-14, 1988, Reno, NV. AIAA 88-0105. Also available as NASA TM-100516.

25. Batina, J. T.: Unsteady Transonic Small-Disturbance Theory Including Entropy and Vorticity Effects. Presented at the AIAA, ASME, et al., 29th Structures, Structural Dynamics and Materials Conference, April 18-20, 1988, Williamsburg, Virginia. AIAA Paper No. 88-2278-CP. Also available as NASA TM-100568.

26. Batina, J. T.; Seidel, D. A.; Bennett, R. M.; Cunningham, H. J.; and Bland, S. R.: Steady and Unsteady Transonic Small Disturbance Analysis of Realistic Aircraft Configurations. Presented at the Transonic Symposium: Theory, Application, and Experiment, April 19-21, 1988, Hampton, Virginia. NASA CP pending. Also available as NASA TM-100557.

27. Bennett, R. M.; Batina, J. T.; and Cunningham, H. J.: Wing Flutter Calculations With the CAP-TSD Unsteady Transonic Small Disturbance Program. Presented at the AIAA, ASME, et al., 29th Structures, Structural Dynamics and Materials Conference, April 18-20, 1988, Williamsburg, Virginia. AIAA Paper No. 88-2347-CP. Also available as NASA TM-100580.

28. Cunningham, H. J.; Batina, J. T.; and Bennett, R. M.: Modern Wing Flutter Analysis by Computational Fluid Dynamics Methods. Presented at the ASME Winter Annual Meeting, December 13-18, 1987, Boston, Massachusetts. ASME Paper No. 87-WA/Aero-9. Also available as NASA TM-100531.

29. Edwards, J. W.: Computational Unsteady Aerodynamics for Lifting Surfaces. Presented at the von Karman Institute for Fluid Dynamics Lecture Series on "Unsteady Aerodynamics," April 18-22, 1988, Brussels, Belgium.

30. Gibbons, M. D.; Soistmann, D. L.; and Bennett, R. M.: Presented at the Fourth National Aero-Space Plane Technology Symposium, February 17-19, 1988, Monterey, California. In NASP CP-4027, Vol. VI, p. 247-268.

31. Mohr, R. W.; Batina, J. T.; and Yang, H. T. Y.: Mach Number Effects on Transonic Aeroelastic Forces and Flutter Characteristics. Presented at the AIAA, ASME, et al., 29th Structures, Structural Dynamics and Materials Conference, April 18-20, 1988, Williamsburg, Virginia. AIAA 88-2304-CP. Also as NASA TM-100547.

32. Vinh, L-S.; Edwards, J. W.; Batina, J. T.; and Seidel, D. A.: Transonic Stability and Control of Aircraft Using CFD Methods. Presented at the AIAA Atmospheric Flight Mechanics Conference, August 15-17, 1988, Minneapolis, Minnesota. AIAA Paper No. 88-4374-CP.

33. Whitlow, W., Jr.: Application of Unsteady Aerodynamic Methods for Transonic Aeroelastic Analysis. Presented at the 16th Congress of the International Council of the Aeronautical Sciences (ICAS), August 28 - September 2, 1988, Jerusalem, Israel. ICAS No. 88-5.5.3. Also available as NASA TM-100665.

34. Whitlow, W., Jr.: Application of a Nonisentropic Full Potential Method to AGARD Standard Airfoils. Presented at the AIAA 26th Aerospace Sciences Meeting, January 11-14, 1988, Reno, NV. AIAA 88-0710. Also as NASA TM-100560.

Tech Briefs:

35. Seidel, D. A.; Batina, J. T.; and Whitlow, W., Jr.: XTRAN2L: A Program for Solving the General-Frequency Unsteady Two-Dimensional Transonic Small-Disturbance Equation (Version 1.2). NASA Tech Brief LAR-13899.

Patents:

36. Hess, R. W.; Davis, W. T.; and Davis, P. A.: Oscillation Pressure Device for Dynamic Calibration of Pressure Transducers. U. S. Patent 4,698,997. Issued October 13, 1987.

Aeroservoelasticity Branch

Journal Publications:

37. Zeiler, T. A.; and Weisshaar, T. A.: Integrated Aeroservoelastic Tailoring of Lifting Surfaces. Journal of Aircraft, Vol. 25, No. 1, pp. 76-83, January 1988.

Formal NASA Reports:

38. Tiffany, S. H.; and Adams, W. M., Jr.: Nonlinear Programming Extensions to Rational Function Approximation Methods for Unsteady Aerodynamic Forces. NASA TP-2776, July 1988.

Conference Presentations:

39. Gilbert, M. G.: Results of an Integrated Structure/Control Law Design Sensitivity Analysis. Presented at the Second NASA-Air Force Symposium on Recent Experiences in Multidisciplinary Analysis and Optimization, September 28-30, 1988, NASA/LaRC, Hampton, Virginia. NASA CP pending.

40. Mukhopadhyay, V.: Digital Active Control Law Synthesis for Aeroservoelastic Systems. Presented at the 1988 American Control Conference, June 15-17, 1988, Atlanta, Georgia.

41. Mukhopadhyay, V.; Pototzky, A. S.; and Noll, T. E.: Control Law Synthesis and Optimization Software for Large Order Aeroservoelastic Systems. Presented at the NASA Computational Aspects in the Control of Flexible Structures, July 12-14, 1988, Williamsburg, Virginia.

42. Mukhopadhyay, V.: Digital Robust Control Law Synthesis Using Constrained Optimization. Presented at the Second NASA-Air Force Symposium on Recent Experiences in Multidisciplinary Analysis and Optimization, September 28-30, 1988, NASA/LaRC, Hampton, Virginia. NASA CP pending.

43. Noll, T. E.; and Perry, B., III: The Active Flexible Wing Aeroservoelastic Wind-Tunnel Test Program - A Status Report. Presented at the NASA Computational Aspects in the Control of Flexible Structures, July 12-14, 1988, Williamsburg, Va.

44. Perry, B.; Buttrill, C. S.; Adams, W. M.; Noll T. E.; Tiffany, S. H.; and Mukhopadhyay, V.: The Active Flexible Wing Program - A Status Report. Presented at the TTCP HAG-6 Workshop on Active Controls and Structural Integrity, Royal Aerospace Establishment, September 28-29, 1988, Farnborough, United Kingdom.

45. Perry, B., III; Dunn, H. J.; and Sandford, M. C.: Control Law Parameterization for an Aeroelastic Wind-Tunnel Model Equipped With an Active Roll Control System and Comparison With Experiment. Presented at the AIAA, ASME, et al., 29th Structures, Structural Dynamics and Materials Conference, April 18-20, 1988, Williamsburg, Virginia. AIAA 88-2211-CP. Also available as NASA TM-100593.

46. Pototzky, A. S.; Spain, V.; Soistmann, D.; and Noll, T. E.: Application of Unsteady Aeroelastic Analysis Techniques on the National Aerospace Plane. Presented at the Fourth National Aerospace Plane Symposium, February, 1988, Monterey, California. Also available as NASA TM-100648

47. Pototzky, A. S.; Zeiler, T. A.; and Perry, B.: Matched Filter Theory and Random Process Approaches for Computing Maximum Dynamic Responses. Presented at the TTCP HAG-6 Workshop on Active Controls and Structural Integrity, Royal Aerospace Establishment, September 28-29, 1988, Farnborough, United Kingdom. Also available as NASA TM-100653.

48. Wieseman, C. D.: Methodology for Matching Experimental and Analytical Aerodynamic Data. Presented at the AIAA/ASME, et al., 29th Structures, Structural Dynamics and Materials Conference, April 18-20, 1988, Williamsburg, Virginia. AIAA Paper No. 88-2392-CP.

49. Zeiler, T. A.; and Buttrill, C. S.: Dynamic Analysis of an Unrestrained Rotating Structure Through Nonlinear Simulation. Presented at the AIAA, ASME, et al., 29th Structures, Structural Dynamics and Materials Conference, April 18-20, 1988, Williamsburg, Virginia. AIAA Paper No. 88-2232-CP.

50. Zeiler, T. A.; and Wieseman, C. D.: Aeroelastic Modeling for the FIT Team F/A-18 Simulation. Presented at the Second NASA-Air Force Symposium on Recent Experiences in Multidisciplinary Analysis and Optimization, September 28-30, 1988, NASA/LaRC, Hampton, Virginia. NASA CP pending.

Landing and Impact Dynamics Branch

Journal Publications:

51. Daugherty, R. H.; and Stubbs, S. M.: Cornering and Wear Behavior of the Space Shuttle Orbiter Main Gear Tire. SAE 1987 Transactions, Vol. 96, Section 6, 1988, p. 1361-1366.

52. Davis, P. A.; and Lopez, M. C.: Static Mechanical Properties of 30 x 11.5 - 14.5, Type VIII, Aircraft Tires of Bias-Ply and Radial-Belted Design. SAE 1987 Transactions, Vol. 96, Section 6, 1988, p. 1367-1377.

53. Howell, W. E.; and Fisher, B. D.: In-Flight Environmental Effects on Airplane Composite Vertical Fin Caps. SAE 1987 Transactions, Vol. 96, Section 6, 1988, p. 1119-1129.

Formal NASA Reports:

54. Daugherty, R. H.; Stubbs, S. M.; & Robinson, M. P.: Cornering Characteristics of Main-Gear Tire of the Space Shuttle Orbiter. NASA TP-2790, March 1988.

55. Davis, P. A.; Stubbs, S. M.; and Tanner, J. A.: Langley Aircraft Landing Dynamics Facility. NASA RP-1189, October 1987.

56. Davis, P. A.; and Lopez, M. C.: Static Mechanical Properties of 30 x 11.5 - 14.5 Type VIII Aircraft Tires of Bias-Ply and Radial-Belted Design. NASA TP-2810, May 1988.

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Spacecraft Dynamics Branch

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FY 1989 PLANS

The FY 1989 plans for the Structural Dynamics Division are broken out by each of the branches (technical areas) and selected highlights of proposed FY 1989 milestones are presented.

Configuration Aeroelasticity Branch

Figure 73 summarizes accomplishments planned for FY 1989 selected from the Branch's broad based research program on dynamic and aeroelastic phenomena of aircraft and rotorcraft.

A large portion of this work is associated with wind-tunnel tests in the Langley TDT with companion theoretical studies. Research studies are planned for both rotorcraft and aircraft. The rotorcraft studies will use the ARES (Aeroelastic Rotor Experimental System) model. Rotorcraft work will focus on applications of advanced aerodynamic and structural methodology to new rotor concepts. Aircraft studies will include flutter testing of the Advanced Tactical Fighter (ATF), a statically unstable cable mounted research model, and ground winds loads testing of the Atlas II launch vehicle. Research testing to evaluate several active control concepts on the Active Flexible Wing (AFW) is planned for the summer as part of a NASA/Rockwell cooperative program.

The work in finite-element modeling of helicopter structures to improve the prediction of vibration characteristics will continue. In-house studies and studies by the major airframe manufacturers will be pursued. In addition, the development of analysis tools for the design of composite rotor blades is a continuing effort with emphasis on the design and testing of extension-twist coupling of rotating tubes and blades.

Highlights of proposed FY 1989 research for the three technical areas of Aircraft Aeroelasticity, Rotorcraft Aeroelasticity, and Rotorcraft Structural Dynamics are shown in figures 74 through 76.

Aircraft Aeroelasticity:
- Aircraft Aeroelasticity

Rotorcraft Aeroelasticity:
- Rotorcraft Dynamics and Aeroelasticity

Rotorcraft Structural Dynamics:
- DAMVIBS Future Emphasis

Unsteady Aerodynamics Branch

For FY 1989 there will be continuing activity in developing computational methods to solve nonlinear, unsteady fluid flow equations for application to aeroelastic analysis (figure 77). There will be continued applications of the CAP-TSD code to aeroelastic response problems in order to further define its range of accuracy. The branch will continue to provide support for the code via a contract programmer, and documentation of the code will proceed. Flutter and aeroelastic response calculations using higher equation level codes will be made, and correlation with potential equation solutions will further delineate appropriate regions for the various flow solvers. This year will also see key developments in the field of vortex and viscous dominated flows and their interaction with aeroelastic response. Research in these areas are being carried on with both structured and unstructured grid methodologies.

Aeroservoelasticity Branch

There are several efforts planned for FY 1989 in the areas of Analysis and Modeling, Control Law Synthesis, and Applications and Validations (figure 78).

In the Analysis and Modeling area, the focus of our attention will be nonlinear transonic aerodynamics for ASE (AeroServoElasticity) applications. This activity will include analyses for the prediction of the passive (open loop) symmetric and antisymmetric flutter boundaries of the Active Flexible Wing (AFW) wind-tunnel model; initial studies to develop procedures for performing ASE analyses and evaluations at transonic speeds; and investigations to begin the integration of nonlinear, time dependent aerodynamics into an ASE design methodology. Studies to evaluate the effects of temperature distributions and thermal stresses and gradients on the aeroelastic characteristics of hypersonic vehicles will be conducted. Various active control concepts which offer the potential for improving the loss of aircraft stability due to thermal effects will also be investigated

In the Control Law Synthesis area, the methodology of using analytical sensitivity expressions will be applied to obtain a reduced-order controller for the AFW wind-tunnel model. The development of an integrated structure and control law design approach based on hierarchal multilevel decomposition and optimization techniques will also be initiated. This study will incorporate the analytical sensitivities of an optimized structural design and of optimum control law solutions. The effort to develop an aerodynamic correction factor methodology based on steady experimental or CFD pressures, total aircraft forces or local aerodynamic section properties for use in obtaining accurate unsteady force predictions will be completed.

In the Applications and Validations area, a cooperative effort between LaRC and Rockwell International will continue through the first series of wind-tunnel tests on the AFW model. The objective of the program is to obtain experimental data for validating multipoint and multifunction digital control law analysis and synthesis methodologies. Activities will also be initiated to provide preliminary aeroelastic and ASE evaluations of the Advanced Launched Systems (ALS) fly-back booster concepts. In the long term, this effort will involve the development of aeroelastic parametric trends, the synthesis of active systems for controlling aeroelastic response and wind-tunnel tests using both semispan and full span, actively controlled, aeroelastic models of the ALS fly-back booster.

Selected highlights of proposed FY 1989 research are listed below and are shown in figures 79 through 83.

Analysis and Modeling:

- Unsteady Time-Domain Aerodynamic Methods to Expand Existing ASE Analysis Capabilities

Control Law Synthesis:
- Reduced-Order Dynamic Controller Design by Multilevel Optimization
- Integrated Aeroservoelastic Design by Multilevel Optimization

Applications and Validations:
- Multi-Year Joint NASA/Rockwell Aeroservoelastic Wind-Tunnel Test Program Underway
- Reusable Flyback Booster Aeroservoelastic Characteristics To Be Determined

Landing and Impact Dynamics Branch

Figure 84 lists the areas of continuing activity in the landing and impact dynamics for FY 1989. The activities include continuation of the development of tire modeling strategies through both in-house efforts and University grants. Work will continue to develop a data base on the operational behavior characteristics of radial and H-type aircraft tires. Efforts will also be focused on research with the active control landing gear for the F-106B aircraft with the goal of demonstrating the potential of the active control gear during landing and ground operations. The ALDF part of the Runway Traction Program will begin in FY 89 as well as the continuation of support for the Heavy Rains Simulation tests using the ALDF with the test wing mounted on the carriage.

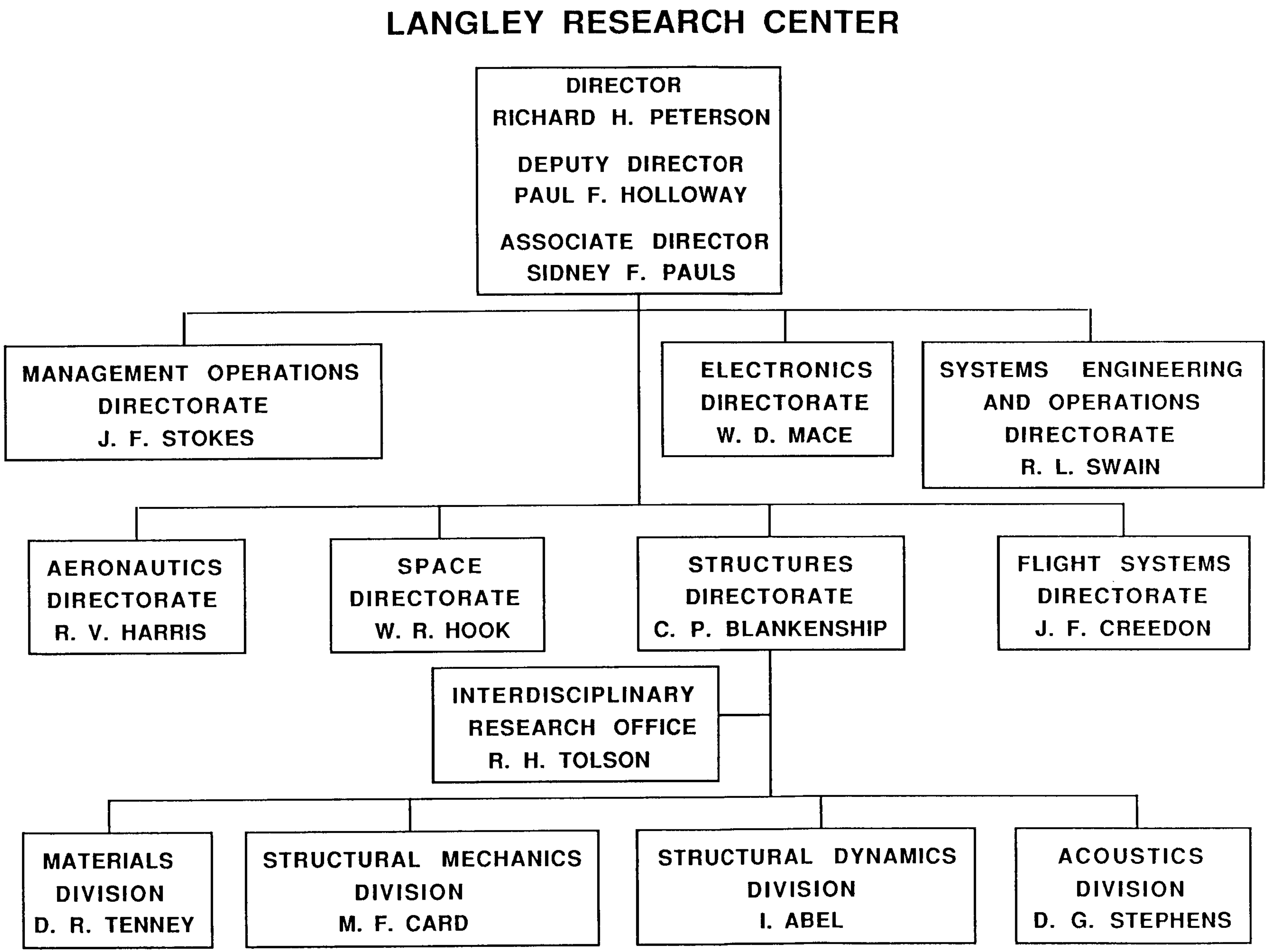
In the impact dynamics area, various static and dynamic tests will be conducted on concepts of composite frames, subfloors, and energy absorbing components for potential application in composite aircraft structures. Tests and analysis will be completed of scale model studies with composite beams under impact loads which will compliment the static data already completed. Additional efforts will continue to update the element library in the computer code DYCAST to enhance the composite structures analysis capability under crash loads. Additionally, evaluation of other computer codes such as DYNA3D and PAMCRASH for applications in composite structures analysis will be conducted on the branch's MicroVAX computer. Furthermore, initial tests utilizing the composite aircraft fuselage components should be underway in FY 1989.

Spacecraft Dynamics Branch

During FY 1989, research will continue on controlled multibody analysis and test (figure 85). A multibody maneuvering experiment will be completed and comparison made with analysis to improve analytical and experimental capability in this area. Developments will also be pursued to incorporate learning control into multibody dynamics. Version 1.0 of the 3-D Large Angle Transient DYNamics (LATDYN) research code for multibody space structures will be available. Applications of LATDYN to multibody control/structure interaction test articles will be performed. In addition, research on test methods will be aggressively pursued. Testing of a generic space station sub-scale model will be carried out and a hybrid model test will be delivered. Moreover, Principal Investigator activities involved in the planning and design for getting structural dynamics on-orbit space station data will be initiated and the hybrid space station sub-scale model will be utilized to verify data acquisition procedures. Finally, an initial phase, large scale, control/structure interaction test model will be installed in the SDRL and tests begun.

CONCLUDING REMARKS

This publication documents the FY 1988 accomplishments, research and technology highlights, and FY 1989 plans for the Structural Dynamics Division.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MATERIALS DIVISION D. R. TENNEY
STRUCTURAL MECHANICS DIVISION M. F. CARD
STRUCTURAL DYNAMICS DIVISION I. ABEL
ACOUSTICS DIVISION D. G. STEPHENS

Figure 1.

STRUCTURAL DYNAMICS DIVISION

IRVING ABEL, CHIEF
LARRY PINSON, ASSISTANT CHIEF
JAMES GARDNER, TECHNICAL ASST.
ROBERT DOGGETT, CHIEF SCIENTIST

5

CONFIGURATION AEROELASTICITY

RODNEY RICKETTS, HEAD
BILL CAZIER, ASST. HEAD

16 + 9

A/C AEROELASTICITY

R/C AEROELASTICITY

R/C STRUCTURAL DYN.

UNSTEADY AERODYNAMICS

JOHN EDWARDS, HEAD

10

METHODS DEVELOPMENT

APPLICATIONS

AEROSERVOELASTICITY

THOMAS NOLL, HEAD

8

ANALYSIS &MODELING

CONTROL LAW SYNTHESIS

APPLICATIONS/
VALIDATIONS

SPACECRAFT DYNAMICS

BRANTLEY HANKS, HEAD

17

GROUND TEST METHODS

OPTIMAL SPACECRAFT PERFORMANCE

MULTIBODY DYNAMICS

TOTAL NASA CS	-	66
TOTAL ARMY CS	-	12
TOTAL		78

LANDING & IMPACT DYNAMICS

JOHN TANNER, HEAD
HUEY CARDEN, ASST. HEAD

10 + 3

LANDING DYNAMICS

CRASH DYNAMICS

Figure 2.

STRUCTURAL DYNAMICS DIVISION

LONG RANGE THRUSTS

AERONAUTICS

- TRANSPORT AIRCRAFT
  - AEROELASTICITY
  - LANDING AND IMPACT DYNAMICS
- HIGH PERFORMANCE AIRCRAFT
  - AEROELASTICITY
- ROTORCRAFT
  - AEROELASTICITY

SPACE

- LARGE SPACE STRUCTURES
  - STRUCTURAL DYNAMICS

Figure 3.

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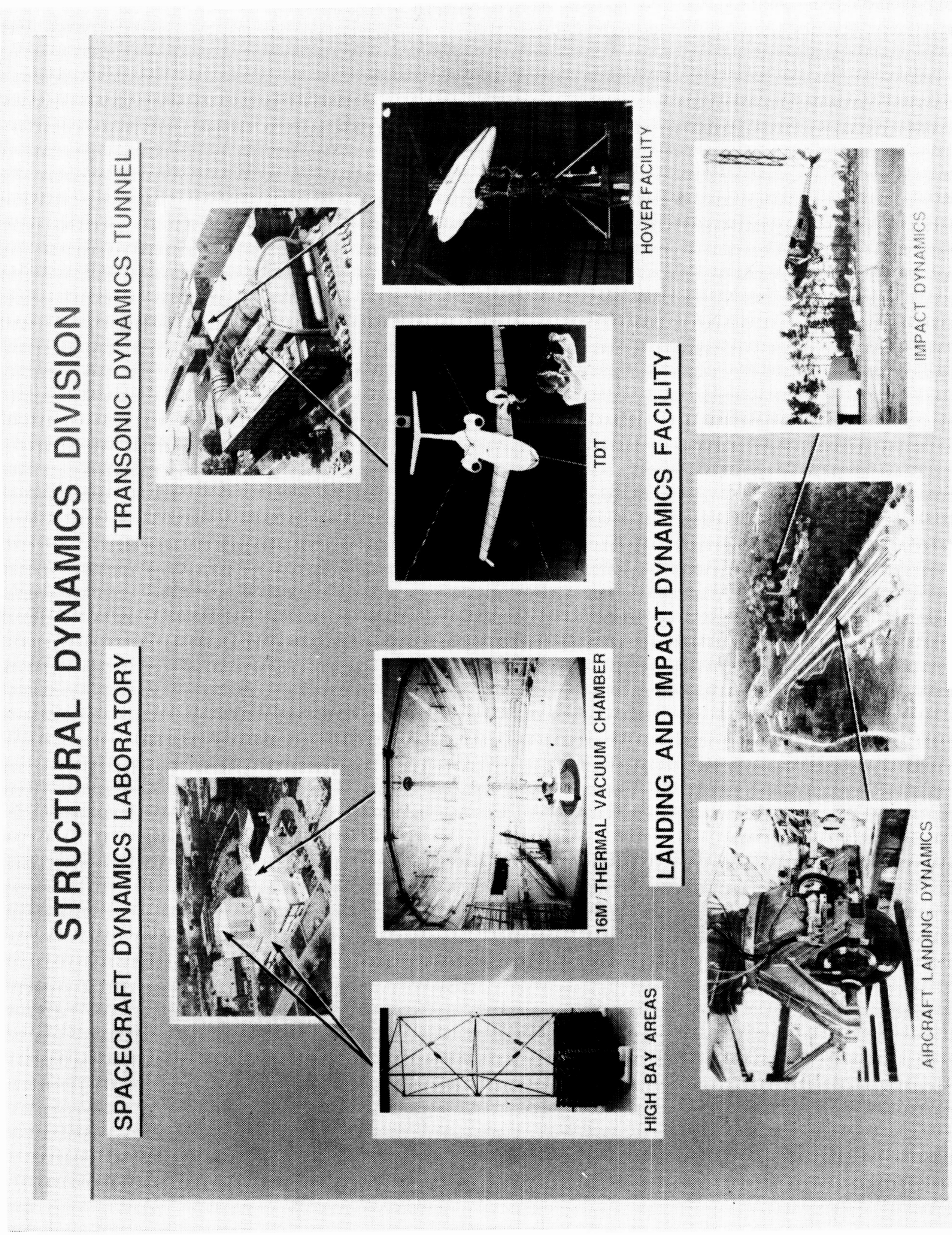

Figure 4.

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NASA
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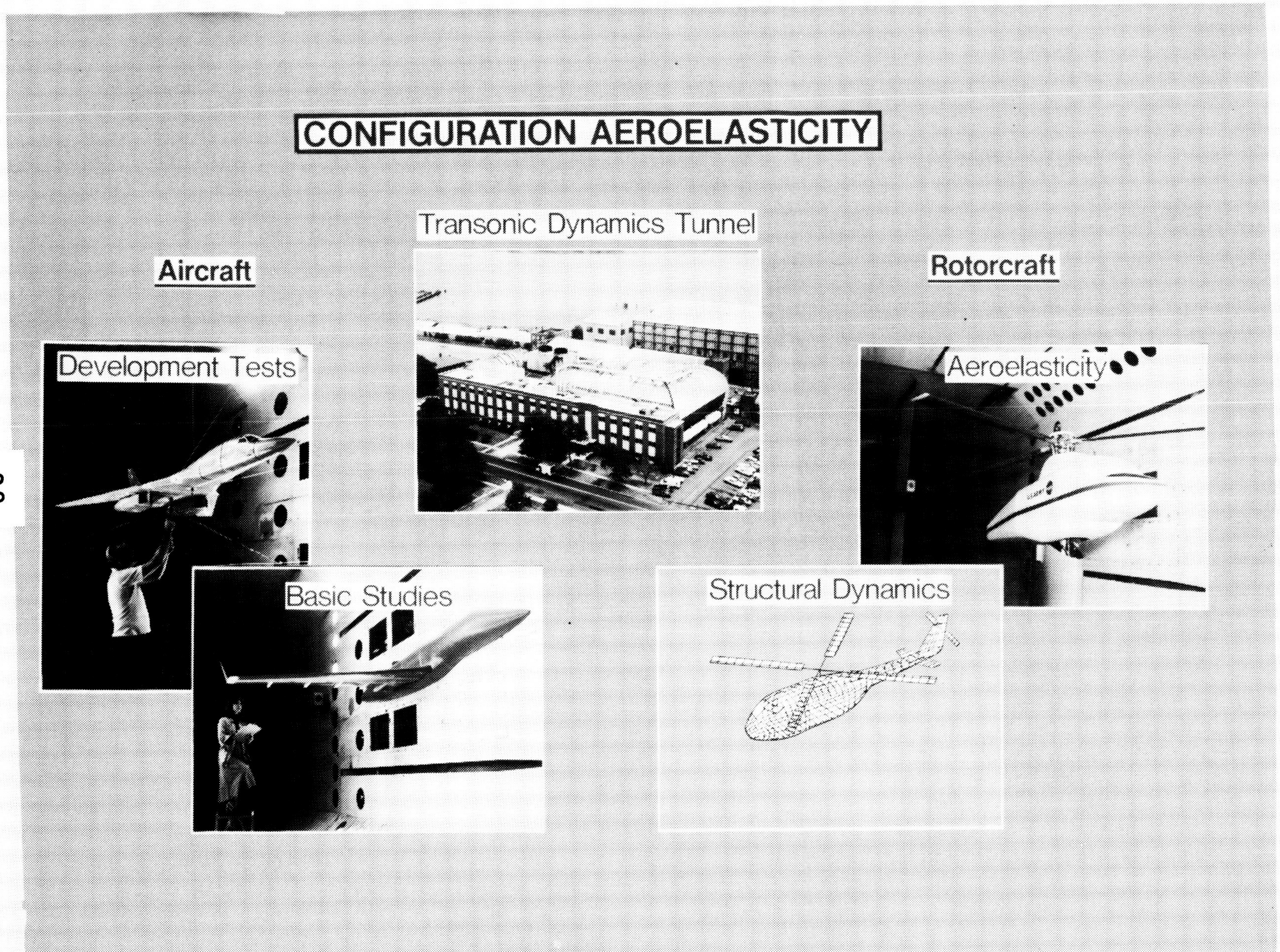


Figure 5.

CONFIGURATION AEROELASTICITY
FIVE YEAR PLAN

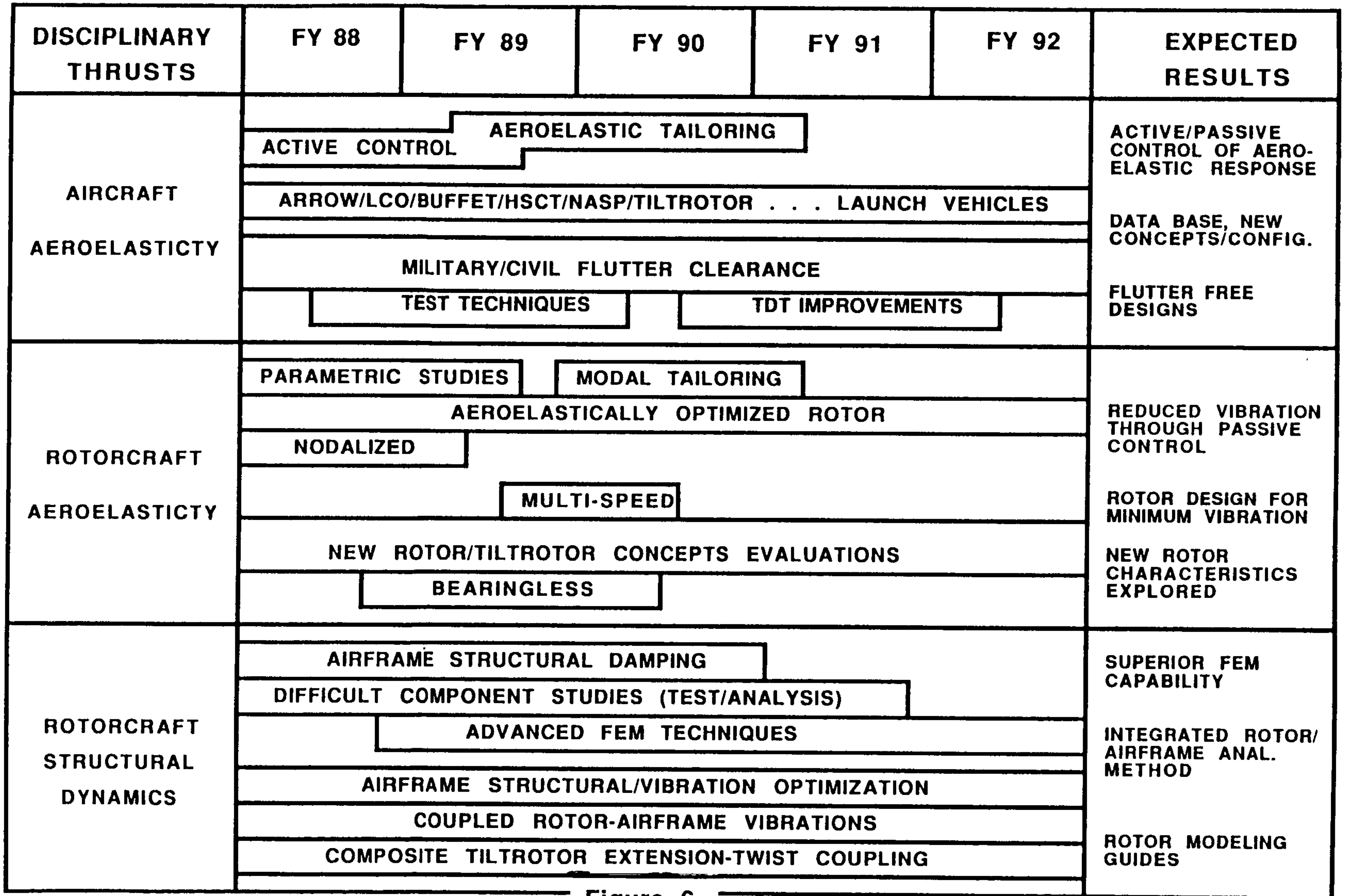


Figure 6.

EFFECTS OF SPAN REDUCTION ON FLUTTER OF ARROW WING SST CONFIGURATIONS DETERMINED IN TDT

Donald F. Keller and Ellen C. Parker (PRC)
Configuration Aeroelasticity Branch

RTOP 505-63-21

<u>**Research Objective**</u>: As a result of renewed interest in Supersonic Transports (SST's), a study was conducted in the Transonic Dynamics Tunnel to determine the flutter characteristics of a generic arrow-wing-configured SST. An arrow-wing planform was chosen because of documented flutter deficiencies, especially in the transonic region, of previous strength-designed arrow-wing SST's. The objective of the current test was to investigate the effects of span reduction on the flutter of various arrow-wing configurations.

<u>**Approach**</u>: Flutter models of a generic arrow wing were designed and fabricated based on a Supersonic Cruise Research (SCR) transport configuration. A flutter test on one of these models was conducted in the TDT over a Mach number range from 0.6 to 1.2. A model planform and a photograph of an arrow-wing model mounted in the TDT are shown in figure 7(b). The semi-span cantilevered wing consisted of a flat aluminum plate with cutouts to represent the stiffness and mass distribution of a typical rib-and-spar construction. Balsa wood was bonded to the wing plate and contoured to form a 6% bi-convex airfoil. Reductions in span of 4%, 10%, 20%, and 30% were accomplished by removing portions of the tip parallel to the root chord. The 30% reduction in span is represented by the shaded region on the model planform figure. A wing fin was mounted at 70% of wing span, vertically or canted outward at 45°, and two flow-through aluminum nacelles were used to represent wing mounted engines.

<u>**Accomplishment Description**</u>: The flutter boundaries presented in the figure are plotted as flutter dynamic pressure versus Mach number. The lower left plot illustrates the effects of span reduction on the arrow-wing configurations with and without engine nacelles. In the transonic region, the 30% reduction in span raised the flutter dynamic pressure approximately 35% for the configuration without nacelles and 130% for the configuration with nacelles. The data in the lower right shows the effects of reducing the span on the nacelle-configuration with and without wing-fins. The addition of a wing-fin at 70% of span, vertical or canted outward at 45 degrees, had no appreciable effect on flutter for the full-span configuration. However, the addition of a vertical wing-fin to the 30% span-reduced configuration lowered the flutter dynamic pressure 10% in the transonic region while the 45 degrees canted wing-fin lowered the entire boundary approximately 30%. Other flutter results (not presented) show that each reduction in span raised flutter dynamic pressure while the addition of a wing fin had little effect on any other configuration. Also, ground vibration tests were conducted on all configurations to acquire mode shapes and frequencies.

<u>**Significance**</u>: The test results indicated that reducing the span by removing portions of the tip raises the flutter boundaries of arrow-wing configurations with and without engine nacelles, especially in the transonic region. This experimental data also showed that nacelle configurations were more sensitive to the reductions in span than configurations without nacelles. Also, the addition of a wing fin significantly lowered the flutter boundary for the configuration with a 30% reduction in span. Finally, the experimental results from this study have increased the flutter data base for generic arrow-wing designs and can be used in the evaluation of analytical methods for predicting arrow-wing flutter phenomenon.

<u>**Future Plans**</u>: Various unsteady aerodynamic codes along with the NASTRAN finite element program are currently being used in an analytical study to compare predicted flutter trends to those obtained during this test, with these results will be published in a NASA report.

Figure 7 (a).

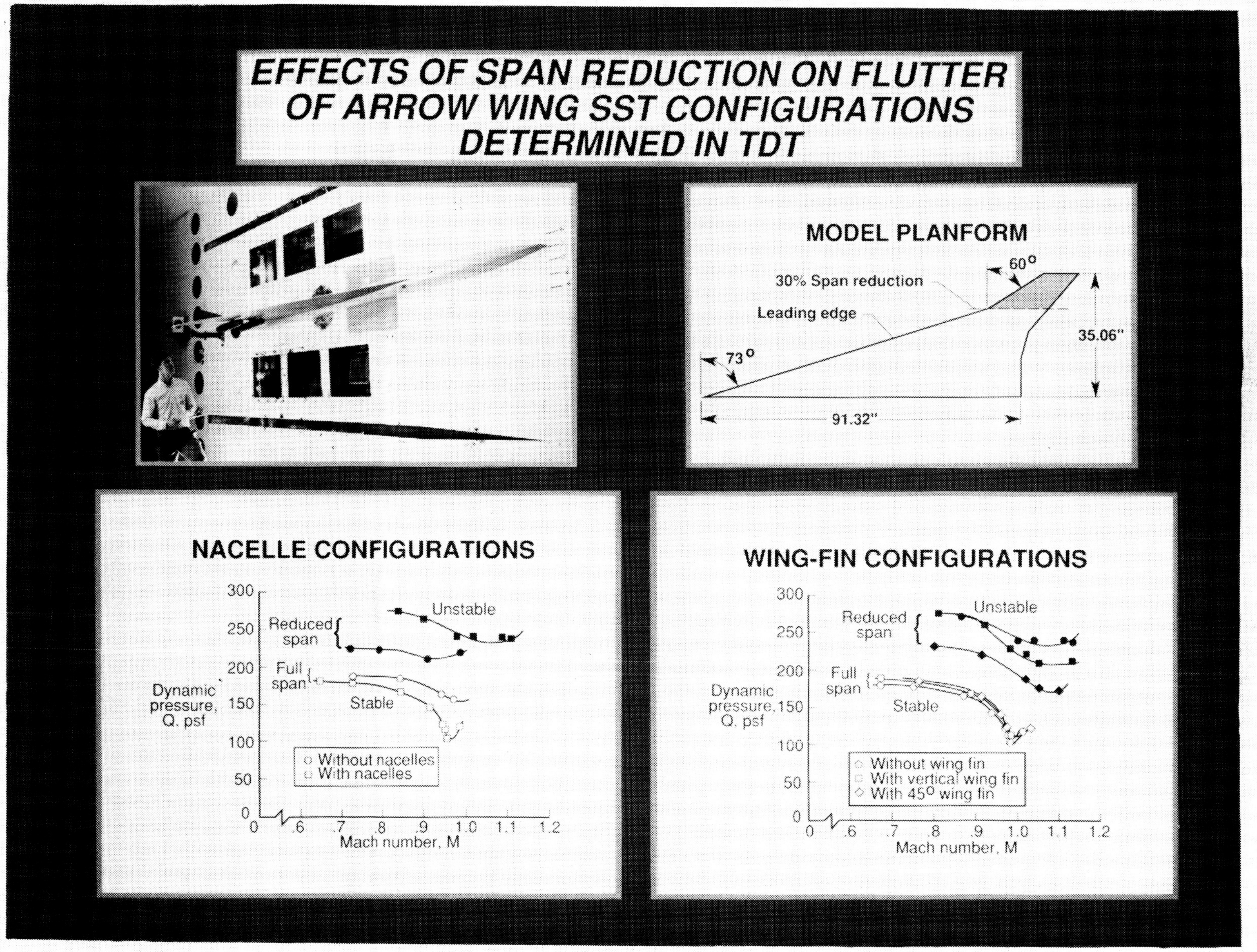

EFFECTS OF SPAN REDUCTION ON FLUTTER
OF ARROW WING SST CONFIGURATIONS
DETERMINED IN TDT
MODEL PLANFORM
60°
30% Span reduction
Leading edge
73°
35.06"
91.32"
NACELLE CONFIGURATIONS
Dynamic pressure, Q, psf
300
250
200
150
100
50
0
Reduced span
Full span
Unstable
Stable
Without nacelles
With nacelles
0
.6
.7
.8
.9
1.0
1.1
1.2
Mach number, M
WING-FIN CONFIGURATIONS
Dynamic pressure, Q, psf
300
250
200
150
100
50
0
Reduced span
Full span
Unstable
Stable
Without wing fin
With vertical wing fin
With 45° wing fin
0
.6
.7
.8
.9
1.0
1.1
1.2
Mach number, M

Figure 7 (b).

FLUTTER CHARACTERISTICS OF SUPERSONIC CRUISE CONFIGURATIONS DETERMINED IN TDT

Michael H. Durham, Stanley R. Cole, Frank W. Cazier, Jr.,
Donald F. Keller, Ellen C. Parker (PRC), and W. Keats Wilkie (Army)
Configuration Aeroelasticity Branch

RTOP 505-63-21

Research Objective - With a renewed interest in supersonic cruise transports (SST's) and documented flutter deficiencies in previous strength-designed SST's, a NASA LaRC study was conducted to determine the flutter characteristics of a generic arrow-wing-configured supersonic transport. The objective of this study was to build a flutter data base to better understand the arrow wing flutter mechanisms and to evaluate current analytical prediction methods.

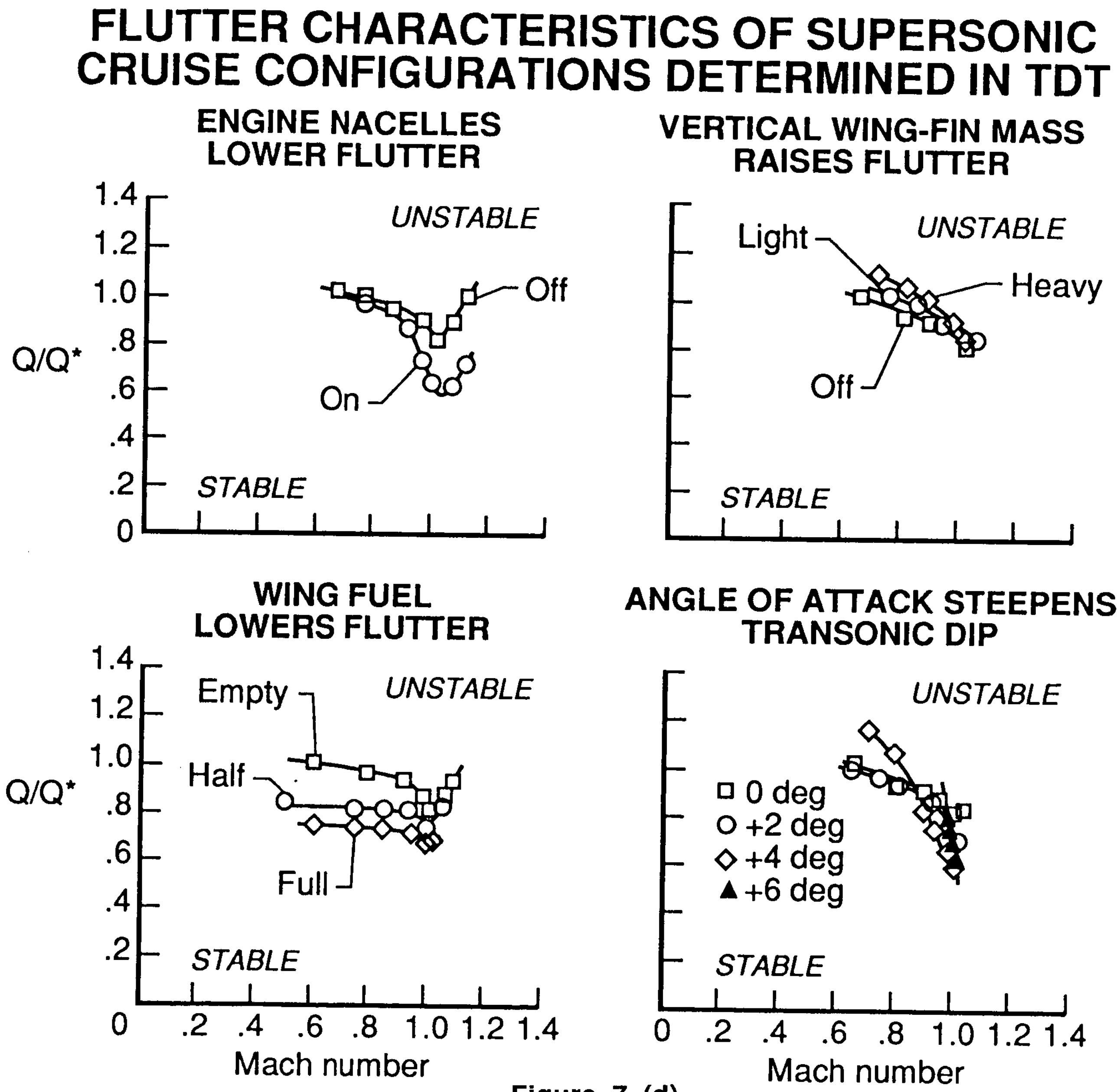
Approach - Flutter models of a generic arrow wing were designed and built based on a Supersonic Cruise Research (SCR) transport configuration. Flutter tests were conducted in the Transonic Dynamics Tunnel (TDT) over a Mach number range from .5 to 1.2 . An arrow-wing flutter model is presented in fig. 7(b). The semi-span cantilevered wing consisted of a flat aluminum plate with cutouts to represent the stiffness and mass distributions of a typical spar and rib construction. Balsa wood was bonded to the wing plate and contoured to form a 6% bi-convex airfoil. A vertical wing-fin was mounted at 70% wing span, and flow-through nacelles were used to represent wing-mounted engines. Wing fuel loadings were modeled with representative weights inserted in the area of typical wing fuel bays.

Accomplishment Description - Flutter boundaries presented in figure 7(d) are plotted as normalized flutter dynamic pressure (Q/Q*) vs. Mach number, where Q* is the flutter dynamic pressure at Mach=0.7 for each nominal case. The upper-left plot presents the effect of engine nacelles. In the transonic region, the addition of engine nacelles lowered the flutter boundary by 25%. The lower-left plot shows a lower flutter dynamic pressure for increased fuel loadings throughout the Mach range tested. In the subsonic region, the full fuel loading decreased flutter dynamic pressures by as much as 25%. In the upper-right plot the vertical wing-fin mass had a slightly stabilizing affect (5-10%) at subsonic speeds, with the transonic bucket remaining unchanged. In the lower-right plot, increased angle-of-attack lowered flutter in the transonic region. However, below a Mach of .9 and with angles-of-attack greater than +2 degrees, flutter speeds were dramatically increased. This combined effect resulted in a much steeper transonic dip. Other results (not plotted), show the aerodynamics of an added vertical wing-fin had negligible effects on the wing flutter. Also, variations in the tip mass & stiffness distributions indicated a more aft spar location was slightly de-stabilizing.

Significance - The results from this study provide a large flutter data base for a generic arrow-wing-design supersonic cruise transport. With this experimental data, analytical capabilities can be evaluated to determine their ability to predict SST flutter phenomena. Test results indicated both fuel loading and engine nacelles adversely affect flutter of an arrow-wing configuration. Therefore, aeroelastic studies should be conducted early in any future SST design.

Future Plans - An analytical study is currently underway to evaluate predicted flutter trends for these arrow-wing models using various unsteady aerodynamic computer codes. Tunnel tests of the arrow-wing flutter models are continuing with parameter investigations including wing-tip span reduction and wing-fin vertical angle. Arrow-wing experimental and analytical results will be published as NASA reports.

Figure 7 (c).



FLUTTER CHARACTERISTICS OF SUPERSONIC CRUISE CONFIGURATIONS DETERMINED IN TDT
ENGINE NACELLES LOWER FLUTTER
UNSTABLE
STABLE
Off
On
Q/Q*
1.4
1.2
1.0
.8
.6
.4
.2
0
VERTICAL WING-FIN MASS RAISES FLUTTER
Light
Heavy
Off
WING FUEL LOWERS FLUTTER
Empty
Half
Full
ANGLE OF ATTACK STEEPENS TRANSONIC DIP
0 deg
+2 deg
+4 deg
+6 deg
0
.2
.4
.6
.8
1.0
1.2
1.4
Mach number

Figure 7 (d).

EFFECTS ON F-16 FLUTTER OF NEW COMPOSITE LEADING EDGE FLAPS AND NEW AIR DEFENSE PYLONS STUDIED IN TDT

Moses G. Farmer and José A. Rivera, Jr.
Configuration Aeroelasticity Branch

RTOP 505-63-21

Research Objective - The General Dynamics F-16 is one of the Air Force's most versatile fighters and has been in the inventory since the mid 1970's. The objective of these tests was to investigate the flutter criticality of two modifications being planned for the F-16: a new composite leading edge flap and a new air defense pylon. The new leading-edge flaps will contain advanced antennas and avionics and will be heavier than the flaps they replace. This modification is planned for future F-16's. Existing National Guard F-16's will be modified with the new air defense pylon for carrying AIM-7 and AIM-120 missiles under the wing.

Approach - A test was conducted in the NASA Transonic Dynamics Tunnel (TDT) using a 1/4 scale aeroelastic model of the F-16 airplane. The wing span of the model is about 8 feet. Figure 8(b) shows the cable-mounted model with the new leading-edge flaps and the air defense pylon with an AIM-7 missile. Tests were conducted with the new leading -edge flaps and the current leading-edge flaps and their respective actuators for comparison. Systematic variations in external stores mounted to the air defense pylon were also performed during the test.

Accomplishment Description - The flutter characteristics of 38 configurations were determined during the test. The experimental results obtained showed that the flutter characteristics of the wing with the new composite leading edge flap were not significantly different than the flutter characteristics of the wing with the current leading edge flap. Test results for configurations with the new air defense pylon with stores showed that the airplane flight envelope will be restricted by flutter. This result is consistent with that obtained by analysis.

Significance - General Dynamics will use these wind tunnel test results together with analytical results to establish flight operation procedures and restrictions for the aircraft.

Future Plans - No additional wind tunnel tests are planned, however flight tests of selected critical store configurations will be performed before these modifications enter service.

Figure 8 (a).

NASA
L-88-6264

EFFECTS ON F-16 FLUTTER OF NEW COMPOSITE LEADING EDGE FLAPS AND NEW AIR DEFENSE PYLONS STUDIED IN TDT

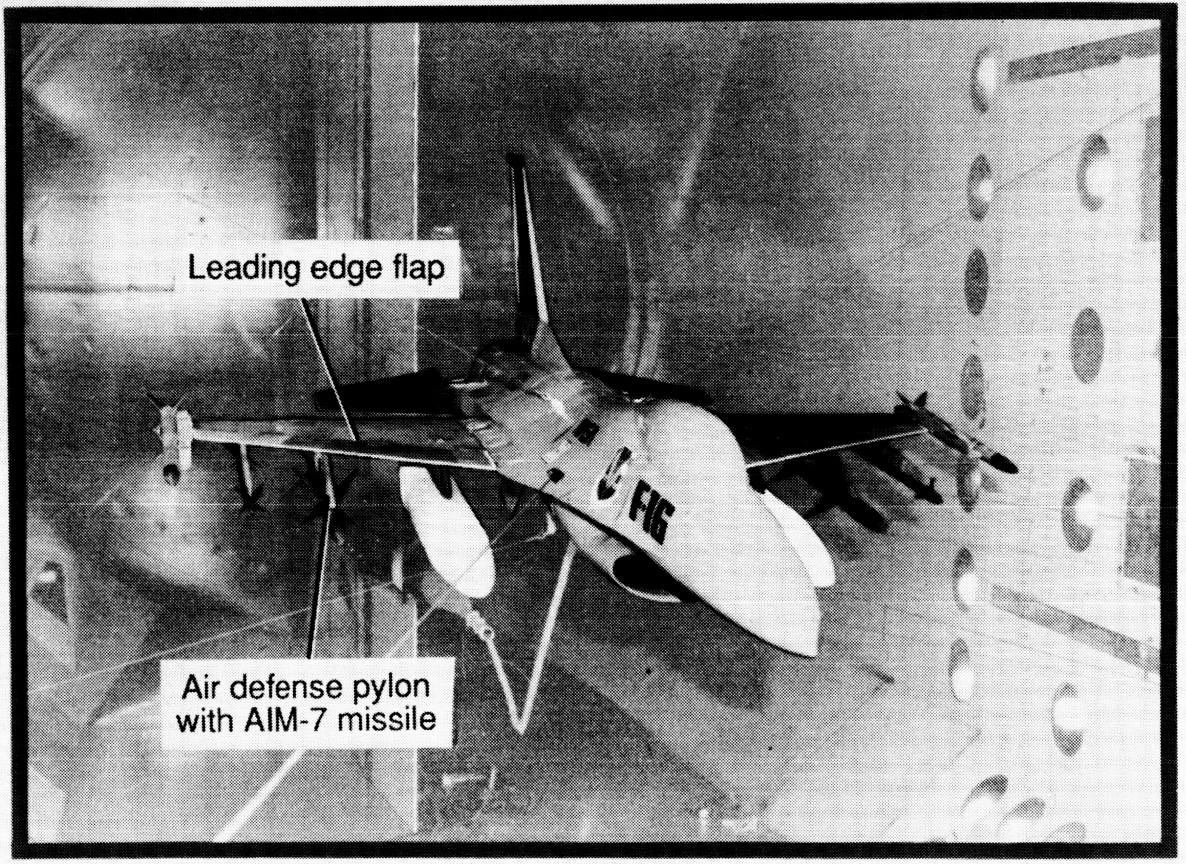


Figure 8 (b).

FLUTTER CHARACTERISTICS OF WING TIP GEOMETRY STUDIED IN THE VIGYAN WIND TUNNEL

Maynard C. Sandford and Ellen C. Parker (PRC)
Configuration Aeroelasticity Branch

RTOP 505-63-21

Research Objective - It is known that the flutter speed of swept-wings can be increased by reducing the wing span. Another approach to increasing the wing flutter speed might be to reduce the wing elastic axis. Therefore, the objective of this study was to determine if shortening the elastic axis of a wing is more beneficial from a flutter standpoint than shortening the span of a wing.

Approach - A 60-degree leading-edge sweep wing configuration was chosen for this study. The model physical properties necessary for flutter testing in the Vigyan tunnel were developed using a finite-element program known as EAL (Engineering Analysis Language) and a linear flutter analysis program known as FAST (Flutter Analysis System). The physical characteristics of each wing were measured and flutter tests conducted in the Vigyan Wind Tunnel. The Vigyan tunnel is an atmospheric, continuous running, open test-section facility. It has a 3x4-foot test section and can obtain dynamic pressures up to about 40 pounds per square foot.

Accomplishment Description - Two wing configurations with a 60-degree leading-edge sweep were studied. Each wing was shortened by cutting the tip off in increments of 10-percent and tested for flutter. A photograph of the 100% Span model mounted in the Vigyan facility is shown in figure 9(b). The results are presented in terms of dynamics pressure as a function of the wing length. The first wing designated as the 100% span model was tested, and the results compared very well with analysis as shown in the top right of the figure. A second wing designated as the 100% elastic axis model was tested and the results also compared very well with analysis as shown in the bottom right of the figure. Overall, the results were similar for both wing models showing about 100-percent increase in flutter dynamic pressure for a 30-percent decrease in either the wing span or wing elastic axis.

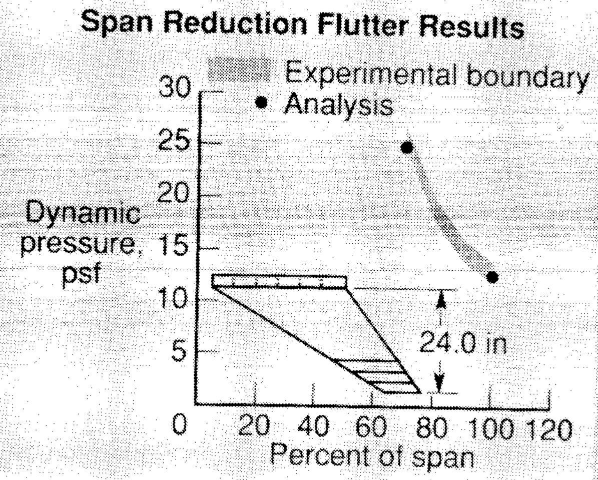
Significance - The test results showed that shortening the elastic axis of a swept wing increases the flutter dynamic pressure in a similar manner to that of shortening the span of a swept wing.

Future Plans - The Vigyan Wind Tunnel was beneficial to these flutter studies and will be considered in the future for similar studies of simple wing models.

Figure 9 (a).

FLUTTER CHARACTERISTICS OF WING TIP GEOMETRY STUDIED IN THE VIGYAN WIND TUNNEL

- Two models studied
- Experimental Results obtained
- Analytical results correlated



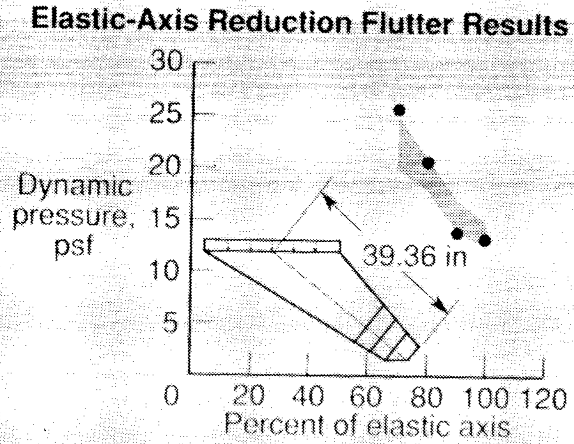


Figure 9 (b).

FLUTTER CHARACTERISTICS AND BOUNDARIES DEFINED FOR HIGHLY SWEPT DELTA WINGS

David L. Soistmann (PRC) and Charles V. Spain (PRC)
Configuration Aeroelasticity Branch

RTOP 506-80-31

Research Objective - Highly swept delta wings are considered likely candidates to be used on hypersonic vehicles. Though these vehicles will be designed to traverse the entire Mach number range, the transonic region is a critical region for flutter instabilities. Therefore, it was the purpose of this test to determine both the flutter characteristics and the flutter boundaries for highly swept delta wings in the transonic range.

Approach - The model used for this test was an aluminum flat plate covered with balsa wood on the upper and lower surfaces to form a 3% circular arc airfoil. The model was fully cantilevered along the root chord and was mounted on a splitter plate outside the tunnel boundary layer. Two configurations, each with a sweep angle of 72 degrees, were tested. The first was a full delta wing planform and the second was a clipped tip delta wing planform. These configurations were tested in the Transonic Dynamics Tunnel (TDT). Prior to the experiment, analytical predictions were made with a finite element structural model and unsteady lifting surface theory aerodynamics.

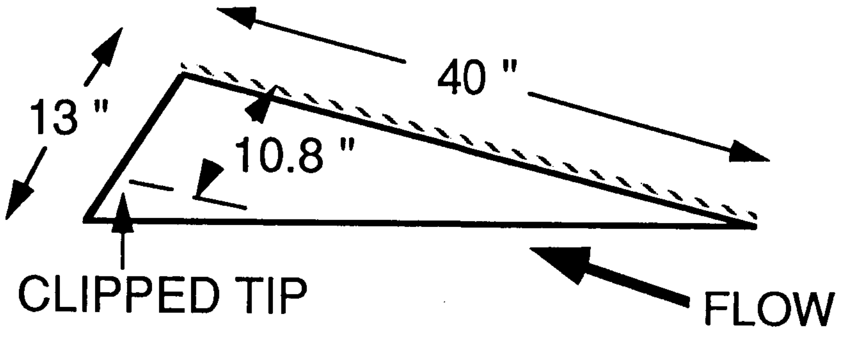
Accomplishment Description - Flutter dynamic pressure results through the transonic range are shown in figure 10(b) for experiment versus analysis for both the delta and the clipped delta configurations. In general, the analysis is conservative in the subsonic range for both the delta and clipped delta configurations. The data for the delta planform in the supersonic range showed the analysis to be slightly unconservative. Explosive flutter was not encountered for either of the configurations under any of the test conditions; instead, a limit cycle oscillation type flutter characterized each of the flutter points obtained.

Significance - This test defined flutter boundaries in the transonic range for 72 degree sweep delta and 72 degree sweep clipped delta wing planforms. It also characterized the type of flutter encountered for highly swept delta wings as being a limit cycle oscillation.

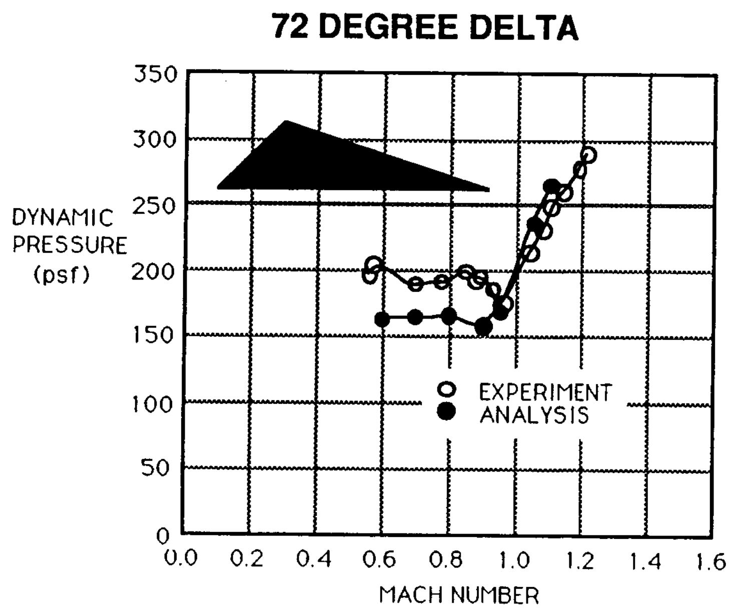
Future Plans - Models are currently under construction for testing of all-movable (free in pitch and plunge) delta wings. Models are also under construction to investigate delta wing control surface buzz. Beyond these tests, plans include investigating full-span wing-fuselage flutter models with delta wings.

Figure 10 (a).

FLUTTER CHARACTERISTICS AND BOUNDARIES DEFINED FOR HIGHLY SWEPT DELTA WINGS



Cantilevered aluminum plate with
3 % circular arc balsa airfoil
Sweep angle = 72 degrees



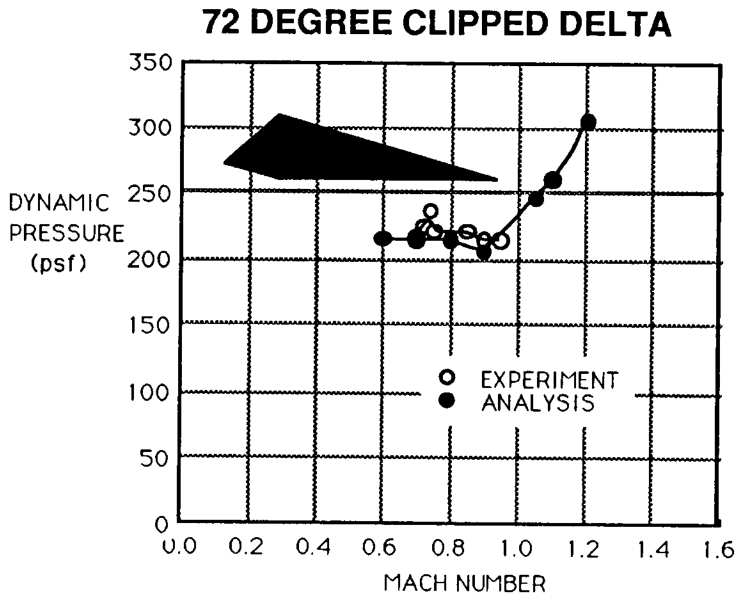


Figure 10 (b).

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TEMPERATURE EFFECTS INTEGRATED INTO FINITE ELEMENT MODELS

Charles V. Spain (PRC)
Configuration Aeroelasticity Branch
RTOP 506-80-31

Research Objective - Hypersonic flight vehicles must endure extreme temperatures and large temperature gradients which cause stuctural deformations, changes in material properties, and internal stresses. This can result in dramatic changes to the vehicle stiffness. The purpose of this investigation is to incorporate these thermal effects into analytical methods used to predict aeroelastic phenomena such as flutter.

Approach - Assuming that structural temperature distributions and temperature dependent material properties can be determined, a procedure was developed to assign material properties to each element in a finite element model (FEM) according to the local temperature. If a model is composed of all isotropic materials, the analyst can then apply standard methods of accounting for the thermally induced internal stresses in the system stiffness matrix. For models composed of anisotropic materials (composites), special routines were developed to assign laminae stacking sequences and calculate effective thermal expansion coefficients for each element.

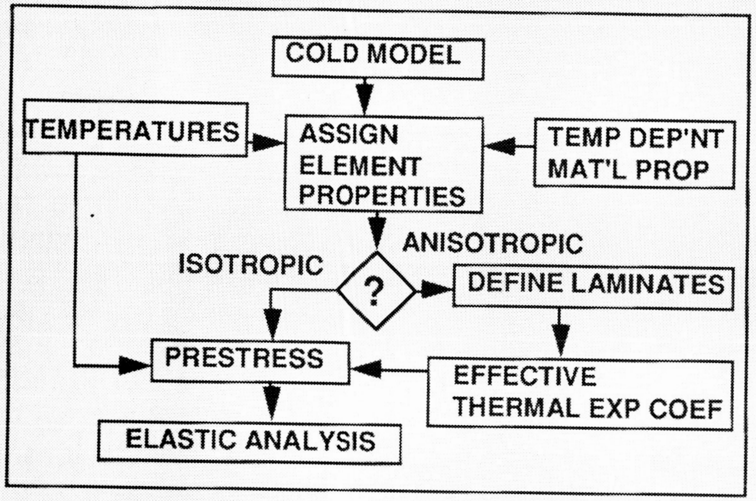
Accomplishment Description - The routines described above have been developed and implemented in Engineering Analysis Language (EAL) as shown in figure 11(b). The procedure was used to assess flutter of an aluminum wing in a hot environment as reported in TND-460. The upper right of the figure is a photo of the test arrangement. The lower left of the figure shows a good comparison of the change in torsional stiffness at the flutter point estimated in the reference, and that predicted by the present analysis. The analysis shows that the dominant factor in reducing the torsional stiffness is the internal stress state and not the change in material properties. Finally, the lower right figure shows good agreement of the analytical flutter dynamic pressure with the experiment.

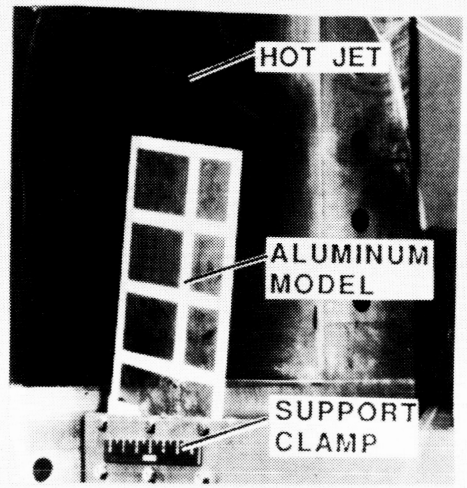
Significance - The method described here is a convenient means of accounting for the effects of a known temperature distribution when performing aeroelastic analyses. It is applicable to structures composed of isotropic or anisotropic materials. The automatic features allow the analyst to perform trend and sensitivity analyses of conceptual configurations which could include numerous possible temperature states.

Future Plans - This method will be used in the aerothermoelastic analysis of NASP conceptual configurations to asses flutter trends and the incorporation of active controls. Convenient methods of predicting aerodynamic heating rates and temperature distribution within a hypersonic vehicle are being studied.

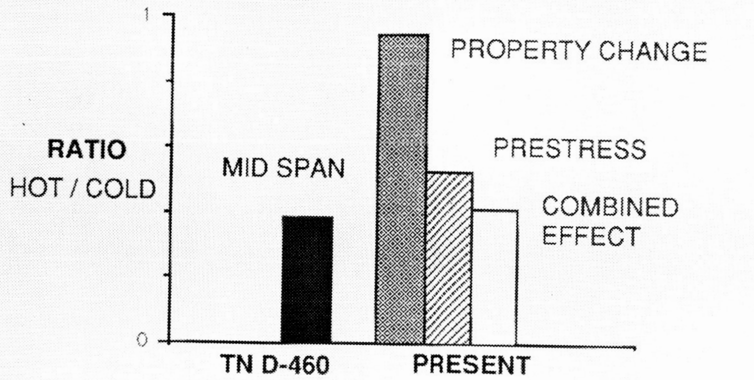
Figure 11 (a).

TEMPERATURE EFFECTS INTEGRATED INTO FINITE ELEMENT MODELS





TORSIONAL STIFFNESS REDUCTION



FLUTTER OF HOT WING

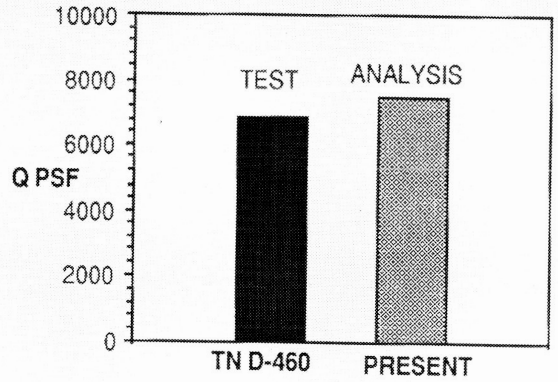


Figure 11 (b).

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TRANSONIC FLUTTER CHARACTERISTICS OF ADVANCED COMPOSITE A-6 REPLACEMENT WING DETERMINED IN TDT

Stanley R. Cole and José A. Rivera, Jr.
Configuration Aeroelasticity Branch

RTOP 505-63-21

Research Objective -Because the A-6 Intruder airplane is an important part of the Navy's air arm and will be used for many years to come, a new composite structure wing to improve fatigue characteristics and to remove flutter placards for certain external store configurations is being developed. This new design included modifications based on results of an earlier flutter test in the Langley TDT. In order to verify that this modified design has acceptable flutter speeds, an experimental study was undertaken to determine its transonic flutter characteristics.

Approach - A 1/4 size dynamically scaled aeroelastic model was tested in the TDT to determine the flutter characteristics of the wing with and without external stores. The store configurations to be studied were selected from those to be used most commonly and to have the most impact on flutter. Fuel tanks internal to the wing structure were also available and could be remotely filled with water to determine internal fuel effects on flutter. A photograph of the semi-span model mounted in the wind tunnel is shown in figure 12(b).

Accomplishment Description - 37 configurations were tested at transonic Mach numbers with twelve showing satisfactory flutter characteristics as tested. The other 25 exhibited flutter within the scaled airplane operating envelope. Based on flutter analyses adjusted by the wind-tunnel test results (not shown), there is substantial indication that the full scale aircraft will be flutter free primarily due to the differences between the support system of the semi-span wind tunnel model and the free-free condition of the flight vehicle. Flutter results obtained for a single 400-gallon fuel tank attached to the outboard pylon position and for a 400-gallon tank attached to both pylons are shown in the figure. The dual 400-gallon fuel tank configuration has a lower flutter boundary than the single tank configuration. Also shown is the effect of internal wing fuel. The empty wing fuel case was found to be more critical than the full wing fuel case.

Significance - The large amount of flutter data obtained has contributed to the available information on the effect of store configuration variables on flutter. Application of results from the wind-tunnel test lead to flutter analyses which indicate the airplane will be flutter free throughout the flight envelope for a variety of wing store configurations.

Future Plans - Wind-tunnel flutter testing is complete.

Figure 12 (a).

TRANSONIC FLUTTER CHARACTERISTICS OF ADVANCED COMPOSITE REPLACEMENT WING FOR A-6 DETERMINED IN TDT

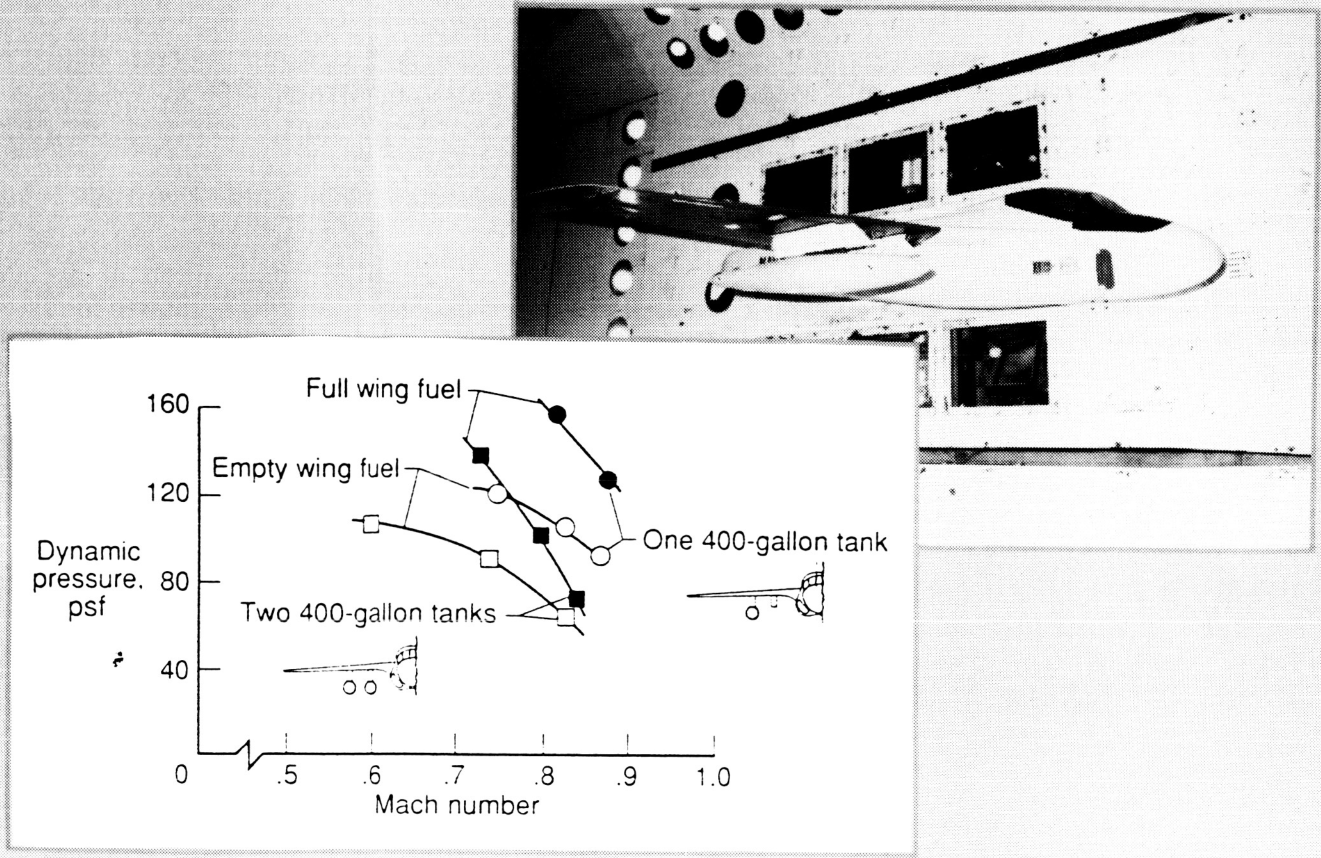


Figure 12 (b).

MILSTAR RADOME TESTED FOR PANEL FLUTTER IN TRANSONIC DYNAMICS TUNNEL

Donald F. Keller and Moses G. Farmer
Configuration Aeroelasticity Branch

RTOP 505-63-21

Research Objective - The Air Force is currently developing a new radome for the MILSTAR program. The radome will contain satellite communication hardware and will be flown on five Air Force aircraft: namely, the C-18, EC-135, RC-135, E-3, and E-4. It will be located on top of the aircraft as shown in the figure. The radome is made of a Kevlar shell 216 inches long with a thickness varying from 0.145 to 0.290 inches. During initial flight tests conducted in Spring 1988, low damping occurred in several of the radome's structural modes. Also, analyses predicted that panel flutter might occur when the local flow on the radome was supersonic. Both of these resulted in the halting of the flight test program. The objective of the wind-tunnel test in the Transonic Dynamics Tunnel (TDT) was to clear the radome of panel flutter throughout the tunnel operating envelope and to acquire both dynamic response and aerodynamic data for fatigue analysis and correlation with analytical results.

Approach - A test was conducted in the TDT of a full scale radome. The radome was mounted on a turntable to allow testing at various angles of attack which would simulate flight sideslip angles. It was instrumented to monitor the response for panel flutter and to measure accelerations, strains, and steady and unsteady pressures. A photograph of it mounted in the TDT is shown in figure 13(b). The radome was tested at various tunnel conditions and sideslip angles in both air and Freon.

Accomplishment Description - The radome was tested at transonic Mach numbers in Freon up to a dynamic pressure of 500 psf for sideslip angles of 0, 3, and 6 degrees and 350 psf for 9 degrees. No panel flutter occurred for any of the configurations. Static aerodynamic data was obtained in the transonic region in air up to a dynamic pressure of 150 psf and in Freon up to 250 psf. This data correlated reasonably well with analytical results.

Significance - Because no panel flutter occurred during the tunnel test, the Air Force has resumed flight testing of the new MILSTAR radome. The Air Force will use the wind tunnel test results along with both the analytical results and flight test data to establish further flight test procedures and possible restrictions on radome-equipped aircraft.

Future Plans - No additional wind tunnel tests are planned.

Figure 13 (a).

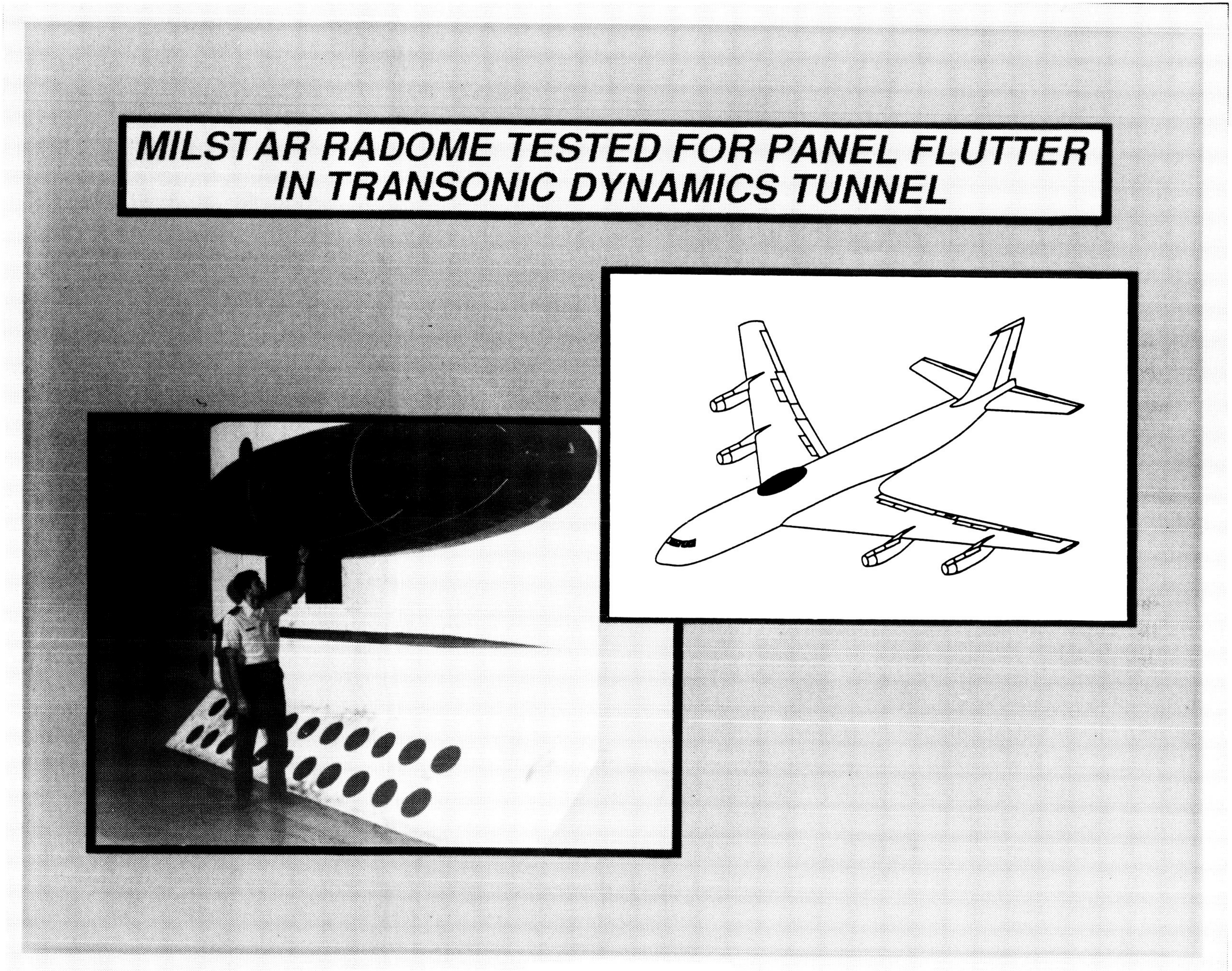

MILSTAR RADOME TESTED FOR PANEL FLUTTER
IN TRANSONIC DYNAMICS TUNNEL

Figure 13 (b).

UNUSUAL TORSION INSTABILITY EXCITED BY SPOILER SURFACES

Stanley R. Cole
Configuration Aeroelasticity Branch

RTOP 505-63-21

Research Objective - Many test techniques, facility capabilities, and model modifications have been developed to make aeroelastic testing safer. One such concept is the use of deployable spoiler surfaces on lifting surfaces to disrupt the flow field around a model and thereby delay the onset of flutter. The current wind-tunnel test was conducted to determine the effects of various spoiler shapes and sizes on the flutter of a low-aspect ratio, rectangular wing.

Approach - A half-span, wall-mounted flutter model was modified so that various spoiler surfaces could be mounted to the top and bottom of the wing for testing in the Langley Transonic Dynamics Tunnel (TDT). The wind-tunnel model had a 20 inch chord, a 1.5 panel aspect-ratio and a NACA 64A010 airfoil shape. The spoiler surfaces were mounted at the location indicated in figure 14(b). Four spoiler height and three width variations were tested.

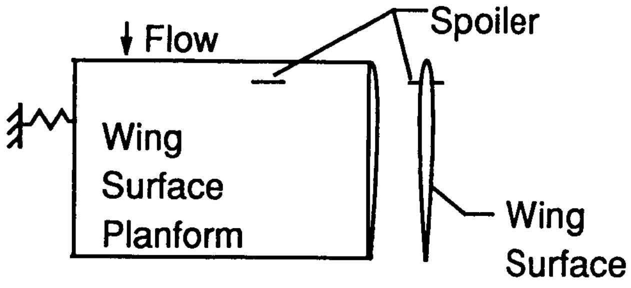
Accomplishment Description - Three spoiler width variations, (5, 10, and 15 percent of wing-panel span) were tested with a height of 0.5". Four height variations (0.25", 0.5", 0.75", and 1.0") were tested for the 10 percent width case. For both of these cases, the subsonic flutter dynamic pressures were slightly increased as the spoiler surface area was increased. In the transonic Mach number range, similar increases in the flutter boundaries were experienced for some spoilers with relatively small surface areas. For larger spoiler surface areas, instead of a slight increase in the flutter condition, a torsion instability was encountered which was extremely Mach number dependent. This torsion instability occurred within a relatively narrow Mach number band and could be found at dynamic pressures as small as 35 percent of the corresponding clean wing flutter condition. The boundary of this torsion instability tended to merge with the conventional flutter boundary (at slightly higher dynamic pressures) for subsonic Mach numbers.

Significance - This test has demonstrated that spoiler surfaces perpendicular to the flow may have little effect on the classical flutter of a wing. The test has further shown that such surfaces can induce a torsion instability that could possibly lead to the destruction of a model if deployed in an attempt to suppress flutter.

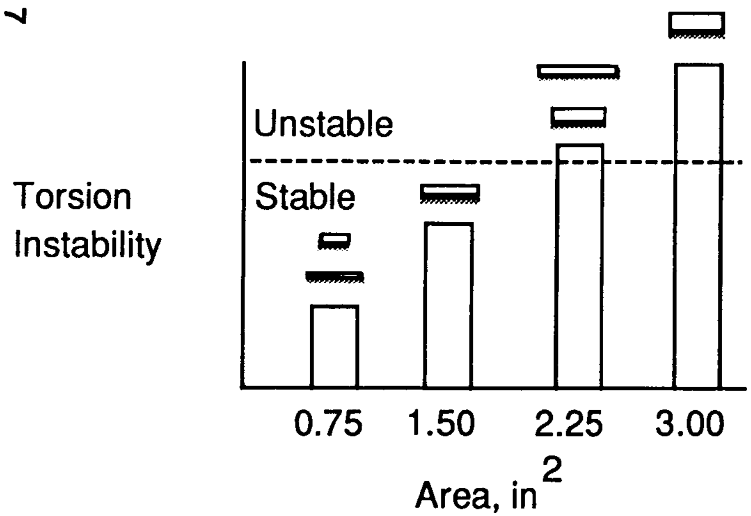
Future Plans - The results of this test will be documented in a formal NASA report.

Figure 14 (a).

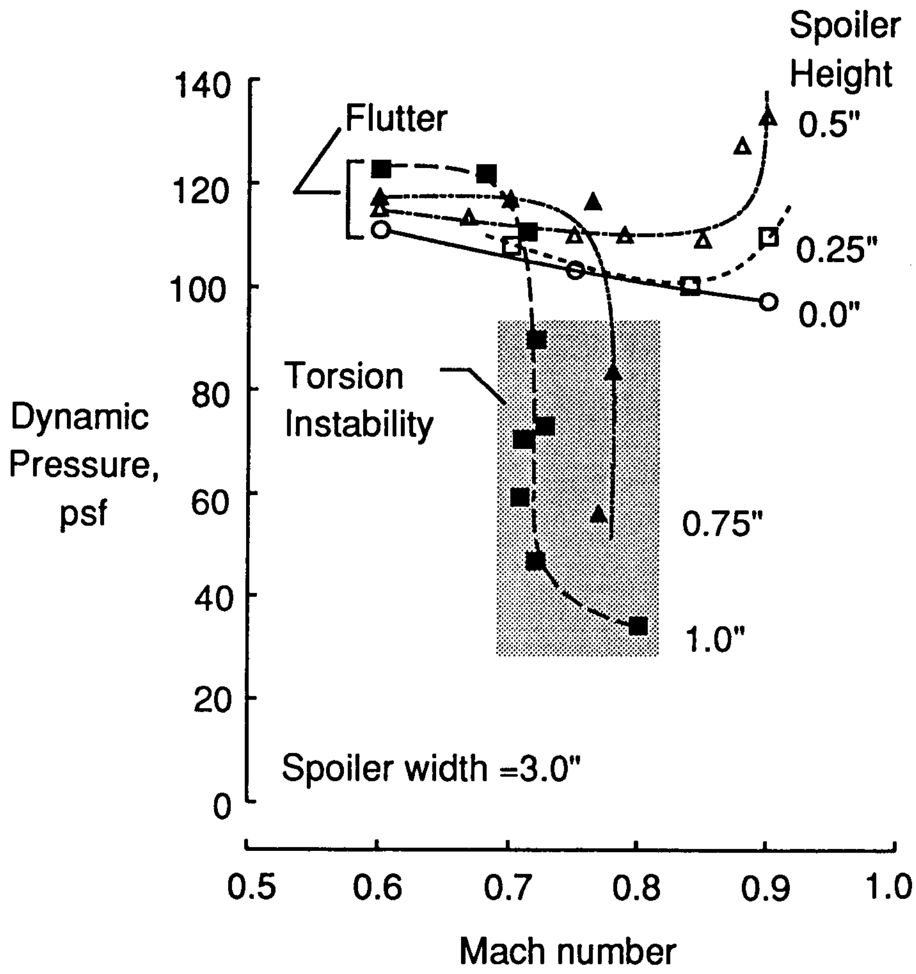
UNUSUAL TORSION INSTABILITY EXCITED BY SPOILER SURFACES



Planform and end view of spoilers mounted on wing



Effect of spoiler area on torsion instability



Spoiler height effect on flutter and torsion instability

Figure 14 (b).

ATLAS-CENTAUR LARGE PAYLOAD FAIRING MODEL INDICATES FLIGHT VEHICLE WILL BE FREE OF AEROELASTIC PROBLEMS

Stanley R. Cole and Mcses G. Farmer
Configuration Aeroelasticity Branch

RTOP 505-63-21

Research Objective - Ever increasing demands on launch vehicle payloads has resulted in an effort to design a larger payload capability for the Atlas-Centaur launch vehicle. In anticipation of possible buffet load limitations or aeroelastic instabilities associated with a large payload fairing, an experimental study was conducted to examine the effect of various fairing length-to-diameter (L/D) ratios.

Approach - A 1/10-size dynamically scaled aeroelastic model was tested in the Langley Transonic Dynamics Tunnel (TDT) to provide a data base of aeroelastic response for several fairing L/D ratios. The model consisted of a rigid shell simulating the forward portion of the Atlas-Centaur vehicle with internal structure to simulate the flexibility of the actual vehicle. Since the majority of the aerodynamic loading occurs near the mode of a typical rocket vehicle, this arrangement was considered to be a sufficient modeling of the flight vehicle. The mass distribution, structural stiffness, and pivot point of the rigid aerodynamic shell could be adjusted to simulate either of the first two body bending modes. This design assumes that the mode shapes for the flight vehicle are linear from the nose to the first node point.

Accomplishment Description - The Atlas-Centaur model was tested with L/D ratios of 0.3, 0.6, 0.8, 1.0, and 1.2 over a Mach number range of 0 to 1.2 and at several dynamic pressure levels (fig. 15(b)). These L/D ratios were tested on both first and second bending mode configurations. The model was found to be free of aeroelastic instabilities. Additionally, the model buffet response was at levels acceptable for the full-scale flight vehicle. The buffet response was remarkably similar for most of the fairing L/D ratios, however, the second mode L/D=0.3 configuration response was substantially greater than for all other second mode configurations.

Significance - The wind-tunnel test was demonstrated that the proposed flight vehicle fairing L/D ratio of 1.0 will be free of detrimental aeroelastic buffet response and instabilities. The parametric study of fairing configurations has resulted in a data base of response information on fairing L/D ratios.

Future Plans - Further data reduction will be accomplished and an evaluation of the information will be made as to its suitability for publishing as a NASA paper.

Figure 15 (a).

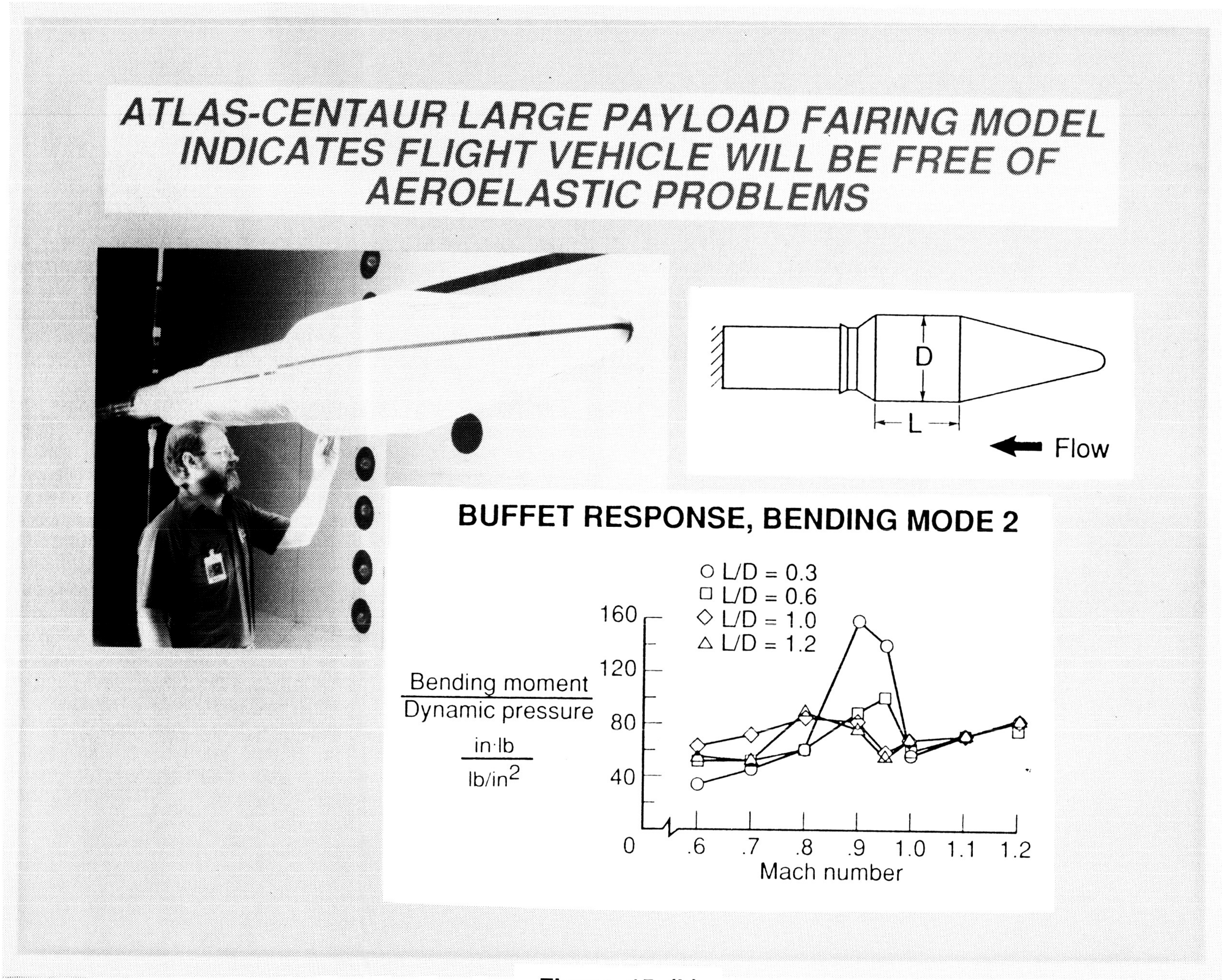

ATLAS-CENTAUR LARGE PAYLOAD FAIRING MODEL
INDICATES FLIGHT VEHICLE WILL BE FREE OF
AEROELASTIC PROBLEMS
D
L
Flow
BUFFET RESPONSE, BENDING MODE 2
○ L/D = 0.3
□ L/D = 0.6
◇ L/D = 1.0
△ L/D = 1.2
Bending moment / Dynamic pressure
in·lb / lb/in²
160
120
80
40
0
.6
.7
.8
.9
1.0
1.1
1.2
Mach number

Figure 15 (b).

SUPERCRITICAL WING TESTED FOR FLUTTER ON PAPA IN THE TDT

Moses G. Farmer and José A. Rivera, Jr.
Configuration Aeroelasticity Branch

RTOP 505-63-21

Research Objective - The objective is to obtain transonic flutter data for a wing with a cambered supercritical airfoil that can be used to validate new analytical methods, such as CAP-TSD, that are being developed to accurately predict transonic flutter.

Approach - A photograph of the wing mounted on the PAPA (Pitch And Plunge Apparatus) is shown in figure 16(b). A wind tunnel test has been conducted in the Langley Transonic Dynamics Tunnel (TDT) to study the flutter characteristics of the wing. The wing has a rectangular planform with a rounded tip. The chord is 16 inches and the semispan is 32 inches. The airfoil has a maximum thickness to chord ratio of 12 percent with a pronounced cusp on the lower surface near the trailing edge that is typical of cambered supercritical airfoils. The PAPA is a flexible mount system with only two degrees of freedom (pitch and plunge) on which rigid wings can be tested for flutter. The structural properties of the wing - PAPA assembly are very well defined. This means that, when making comparisons between experimental and analytical results, differences can be attributed to inaccuracies in the analytical modeling of the unsteady aerodynamic flow over the wing.

Accomplishment Description - The figure shows some of the experimental flutter data that has been obtained in the TDT. Eleven flutter points define the measured flutter boundary as a function of Mach number and dynamic pressure. The figure also shows a calculated flutter boundary that was obtained using a state-of-the-art kernel function method for analytically modeling the unsteady flow over the wing. The analysis does not account for the effects of flow viscosity, airfoil thickness, or airfoil shape. The calculated boundary is an essentially straight line that slopes upward with Mach number. Between 0.4 and 0.6 Mach numbers, where the flow is subsonic, there is a good agreement between the measured and calculated boundaries. Above 0.6 Mach number the measured boundary shows a transonic dip and then rises sharply above 0.8 Mach number. At 0.8 Mach number the measured flutter boundary is about 10 percent lower than the calculated boundary.

Significance - The transonic dip, followed by a sharp rise, shown by the measured flutter boundary is typical of rectangular wings. Based on previous experience, the transonic dip for this wing with a supercritical airfoil is probably greater than it would be for a similar wing with a conventional airfoil.

Future Plans - New analytical methods that are being developed will be used to make improved calculations of the flutter boundary. The extent to which these calculations agree with the measured boundary will indicate the accuracy of these new analytical methods.

Figure 16 (a).

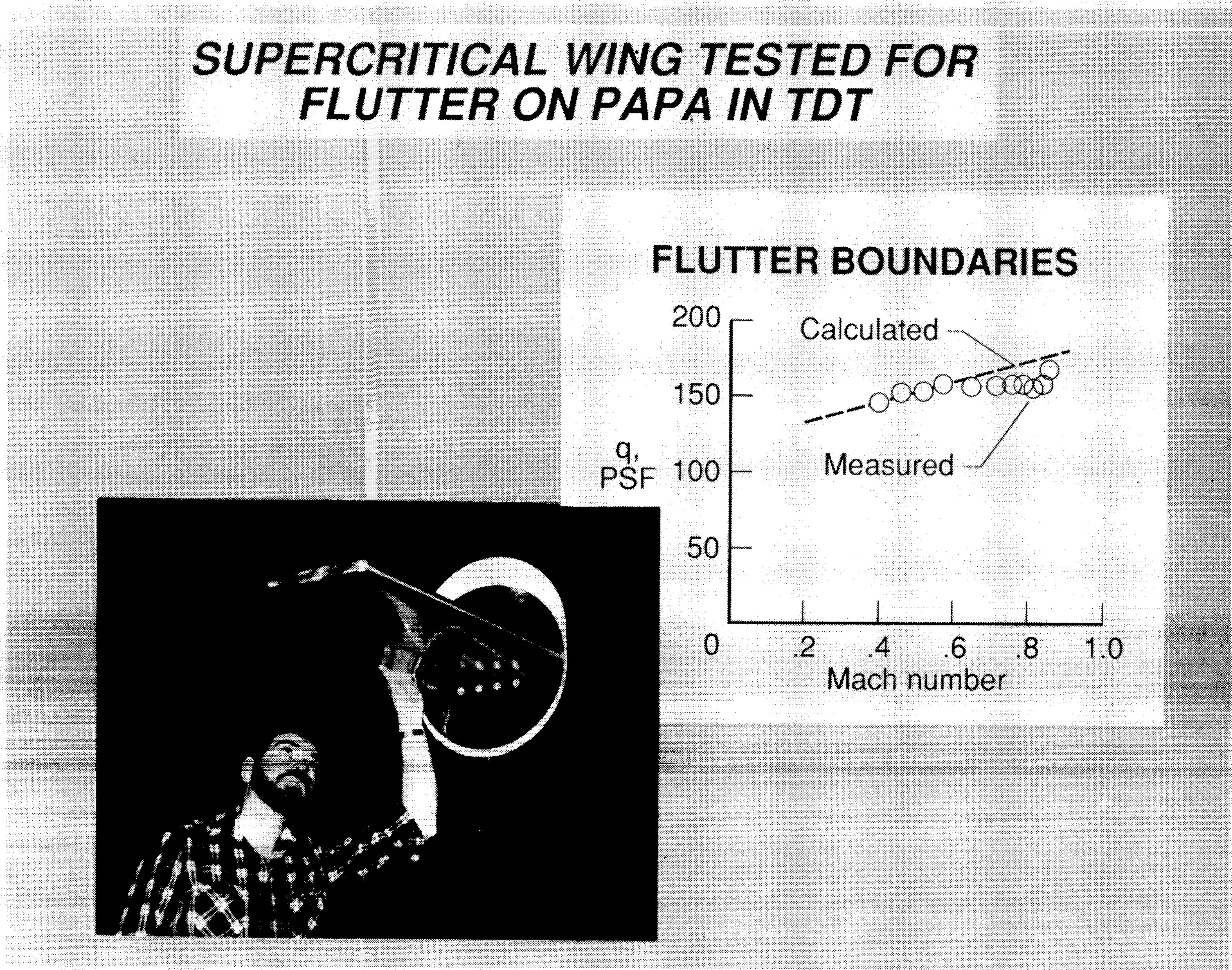

SUPERCRITICAL WING TESTED FOR
FLUTTER ON PAPA IN TDT
FLUTTER BOUNDARIES
q, PSF
200
150
100
50
0
.2
.4
.6
.8
1.0
Mach number
Calculated
Measured

Figure 16 (b).

LASER LIGHT SHEET FLOW VISUALIZATION SYSTEM DEVELOPED FOR TDT

José A. Rivera, Jr. and Bryan E. Dansberry
Configuration Aeroelasticity Branch

RTOP 505-63-21

Research Objective - The purpose of this effort is to develop and test a laser-based flow visualization system for use in the Transonic Dynamics Tunnel (TDT) to assist in visualizing complex steady and unsteady aerodynamic flows.

Approach - The system is composed of a high powered argon laser, an acousto-optic modulator (AOM), a computer controlled three-axis-rotation optical system, and a propylene glycol vapor generator. These components were installed within the TDT plenum as shown in the figure 17(b). In operation, the laser beam passes through the AOM and is reflected downwards into the optical system by means of a fixed mirror. The optical system transforms the laser beam into a sheet of light and controls its orientation (parallel or perpendicular to the flow) and location (upstream or downstream, and laterally) within the test section. The AOM provides for either a continuous or a strobed laser light sheet in order to "freeze" unsteady flow. The light sheet is used to illuminate particles injected into the flow upstream of the model by the propylene glycol vapor generator. Color video, 35mm, 70mm, and high-speed motion picture cameras were used to record results.

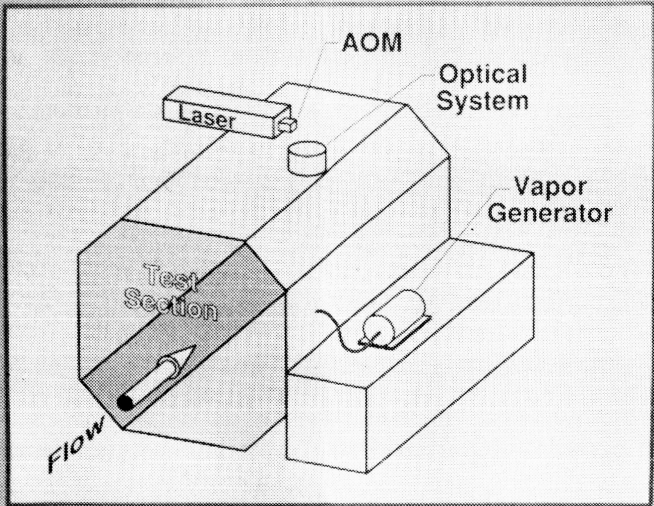
Accomplishment Description - The system was successfully demonstrated in air and Freon on a static and oscillating model during tests in the TDT. The maximum tunnel conditions were Mach-0.92, and dynamic pressure (q)=50 psf. The model maximum steady state angle of attack (α) was 25°. The model maximum oscillation frequency was 6.7 Hz, α=15° $\pm$9.5°. Excellent color video and still camera pictures were obtained showing the leading edge vortex on the model. The attached figure shows a sketch of the model with a leading edge vortex shown to assist in interpreting the multiple exposure 70mm photograph of the model. The model α is 12°.

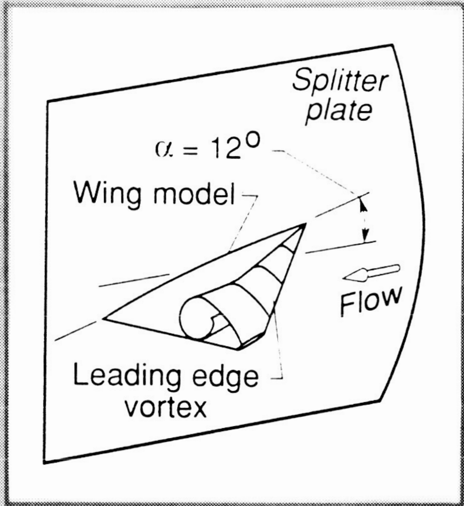
Significance - A laser light sheet flow visualization system for TDT has been developed and successfully tested. This system will allow the study and understanding of complex aerodynamic flows around advanced aerospace configurations.

Future Plans - The test identified a potential improvement of the optical system that may increase the area of the laser light sheet being generated. In addition, a need for further research to obtain quality results from the high-speed motion picture camera was identified. These two areas will be pursued.

Figure 17 (a).

LASER LIGHT SHEET FLOW VISUALIZATION SYSTEM DEVELOPED FOR TDT





SYSTEM COMPONENTS

- Laser
- Acousto-optic modulator (AOM)
- Optical system
- Vapor generator

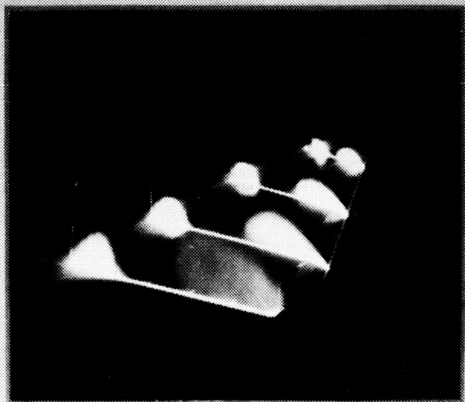


Figure 17 (b).

PAPA - A NEW WIND-TUNNEL MOUNT SYSTEM FOR FLUTTER RESEARCH

Moses G. Farmer and José A. Rivera
Configuration Aeroelasticity Branch

RTOP 505-63-21

Research Objective - The objective is to develop a wind-tunnel model support mechanism that provides a means for using structurally simple but aerodynamically representatives flutter models. The use of such a mechanism with its simple and easily mathematically modeled structure would be an invaluable tool in determining experimental data for correlation with analysis because disagreement between theory and experiment can be attributed directly to aerodynamic differences. To this end an aeroelastic model mounting system called the pitch and plunge apparatus (PAPA) has been developed for use in the Langley Transonic Dynamics Tunnel (TDT).

Approach - A photograph of the wing mounted on the PAPA and a schematic drawing of the PAPA apparatus are shown in figure 18(b). Most of the apparatus is behind a large splitter plate and is shielded from the wind by a fairing. The fairing is not shown. The moving plate is attached to an existing turntable in the tunnel wall by a system of flexible rods. The rods all have fixed-end conditions at both ends. The arrangement and stiffness of the rod system is such that only pitch and plunge motions are possible. The simulation of pitching and plunging motions is fundamental to simulating flutter mechanisms. The mass and inertia of the moving plate can be adjusted to change the natural frequencies of the system. The small end plate at the wing root, which is shown only in the photograph, is the only part of the apparatus that is exposed to the wind. The wing, end plate, and moving plate form a rigid structure. A small gap prevents the end plate from rubbing against the splitter plate.

Accomplishment Description - Two characteristics of the PAPA are particularly noteworthy. First, the system has a soft spring rate but can carry large lift loads, thus providing an effective means for conducting flutter studies of lifting wings. Second, it has very small structural damping that does not change as loading conditions change, thus providing an effective means of studying flutter phenomena wherein loss of aerodynamic damping is the primary mechanism. The fact that the PAPA provides the desired dynamic characteristics has been verified in a series of tests in the TDT. Some illustrative results from a recent application are presented in the figure. The data were obtained using a wing with a rectangular planform and a 10% thick symmetric airfoil. The data plot shows the effects of angle of attack on flutter at 0.60 Mach number. At an angle of attack of zero degrees classical flutter occurred with the characteristic coupling of the pitch and plunge degrees of freedom. As the angle of attack is increased from zero degrees, the flutter dynamic pressure gradually decreases in a linear manner until about five degrees is reached where an abrupt decrease begins. This decrease is caused by the transition form classical flutter to stall flutter resulting from flow separation occurring on the wing.

Significance - The PAPA provides a means for testing structurally simple and relatively inexpensive flutter models to evaluate effects of aerodynamic changes on flutter. Results so obtained are invaluable to assessing the accuracy of new unsteady aerodynamic theories and aeroelastic analysis methods.

Future Plans - The PAPA system has been added to the inventory of research tools available for conducting aeroelastic tests in the TDT and will be used in a variety of parametric studies of flutter phenomena.

Figure 18 (a).

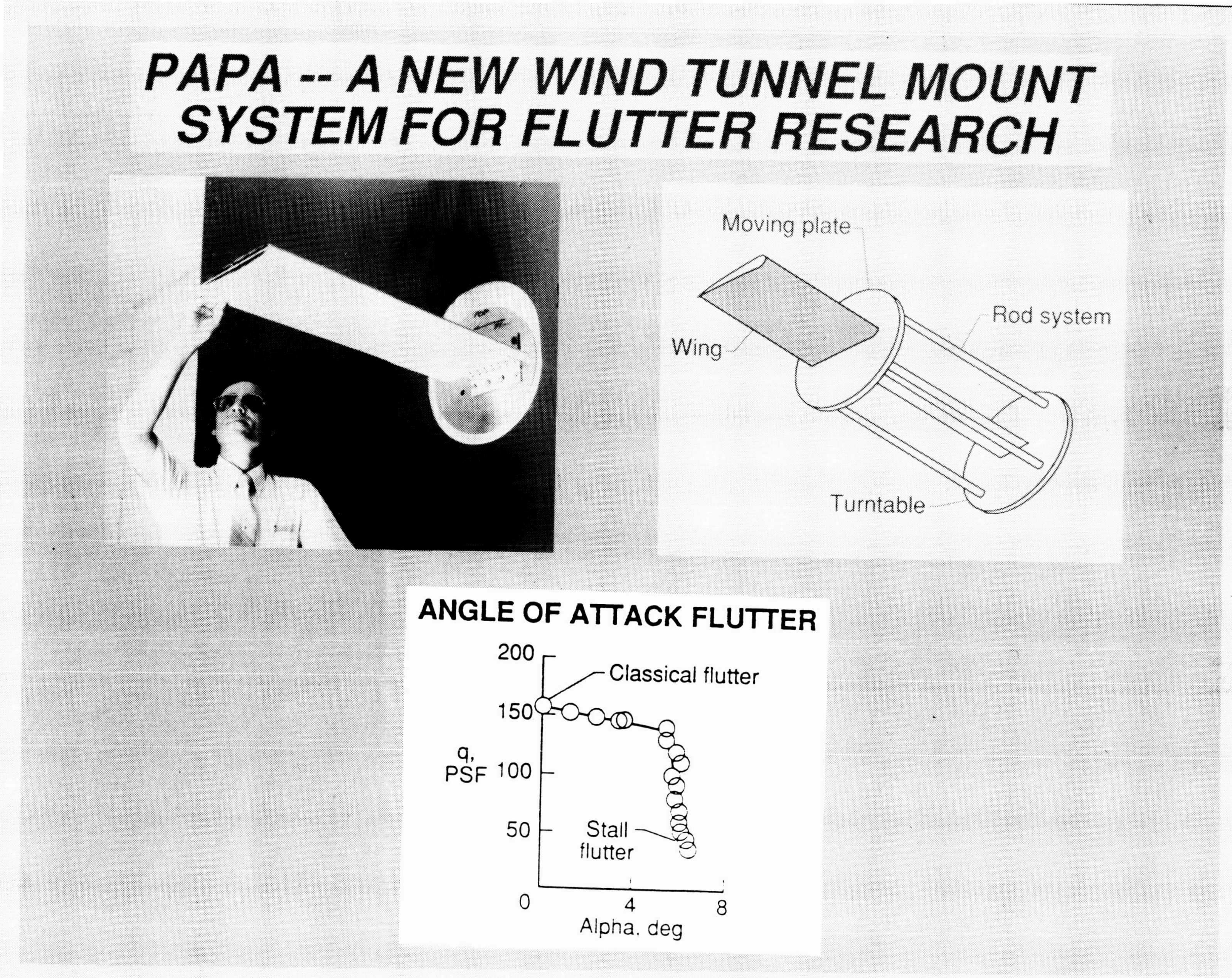

PAPA -- A NEW WIND TUNNEL MOUNT SYSTEM FOR FLUTTER RESEARCH
Moving plate
Rod system
Wing
Turntable
ANGLE OF ATTACK FLUTTER
200
150
100
50
0
q, PSF
Classical flutter
Stall flutter
4
8
Alpha, deg

Figure 18 (b).

PASSIVE BLADE TWIST CONTROL IMPROVES PERFORMANCE OF TILT-ROTOR VEHICLES

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RTOP 505-63-51

Research Objective - Tilt-rotor aircraft are designed to operate in both helicopter and airplane modes of flight. This operational flexibility results in several conflicting design requirements. One such design requirement, which has significant effects on aerodynamic performance, is the twist of the rotor blade. Typically, the twist is not optimum for either flight mode. Performance could be improved if it were possible to vary the blade twist between the airplane and helicopter modes. Tilt-rotor aircraft typically vary rotor speed by 20 percent between flight modes which induces a change in the centrifugal force. As illustrated in figure 19(b), centrifugal force could be used to change the blade twist of an extension-twist-coupled composite blade. Thus, the objective is to design an extension-twist-coupled composite blade to have optimum twist distributions in both airplane and helicopter flight modes.

Approach - First, the aerodynamic performance of the XV-15 with its existing metal rotor system was determined for hover in the helicopter mode and forward flight in the airplane mode. Second, a study was performed to determine the optimum linear twist and associated performance improvement for each flight mode. Lastly, several extension-twist-coupled blade designs were developed using realistic design parameters and applicable structural requirements. The blades were optimized using nonlinear programming techniques for maximum twist change between the flight modes so as to allow the greatest flexibility in the aerodynamic twist design. The performance associated with each of the resulting twist designs obtained for the extension-twist-coupled blades was then compared to the performance associated with the existing metal blade.

Accomplishment Description - Results of the linear twist study are shown in the lower left of the figure to have the optimum linear blade twist at -20 degrees in hover and at -42 degrees in forward flight. The associated aerodynamic performance improvements to the conventional twist-design were about six percent in both flight modes. Three extension-twist-coupled blades were designed subject to structural requirements in an attempt to obtain the performance improvements shown possible by the twist variation study. The results illustrated in the lower right of the figure show that the best of the three designs improved the performance in forward flight by 6.5 percent and simultaneously improved performance in hover by 4.8 percent.

Significance - This work shows that a change in blade twist between flight modes can substantially improve tilt-rotor performance and that the necessary twist changes can be obtained within realistic design requirements and material design strengths using extension-twist-coupled blade designs.

Future Plans - As discussed above, the current effort divided the design process into three steps. A follow-on effort is planned which will optimize the performance directly using one procedure that ties all the design steps together.

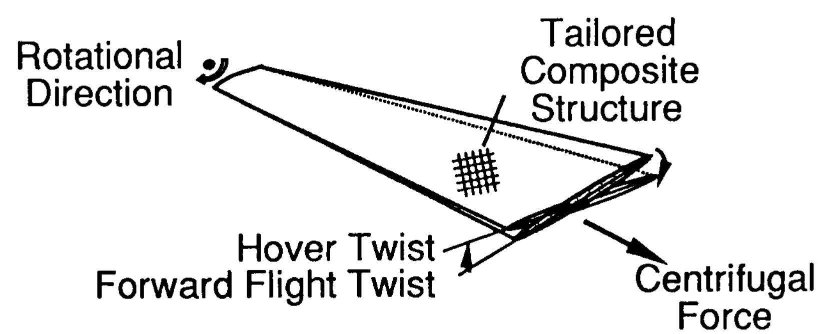
Figure 19 (a).

PASSIVE BLADE TWIST CONTROL IMPROVES PERFORMANCE OF TILT-ROTOR VEHICLES

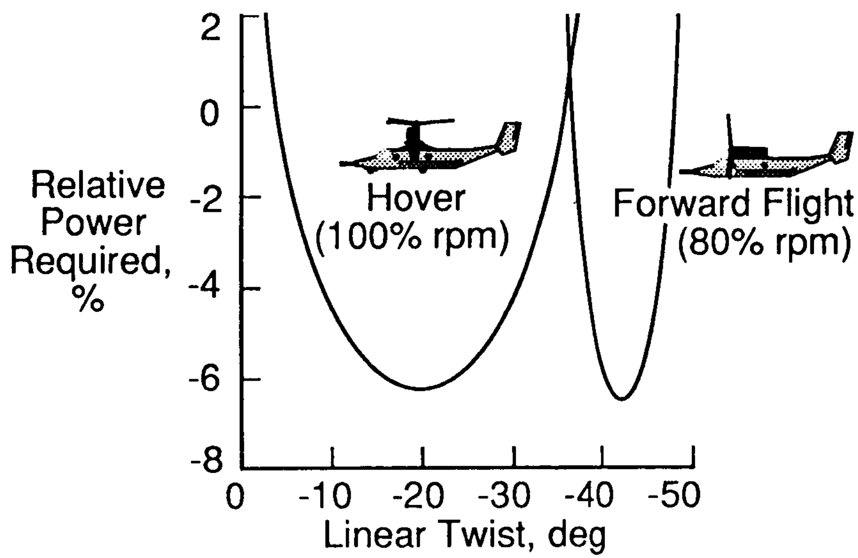
Multiple Design Requirements

- Conventional tilt rotor blade geometry
- Blade strength
- Twist deformation between flight modes

Design Concept



Linear Twist Performance Relative to XV-15 Twist Performance



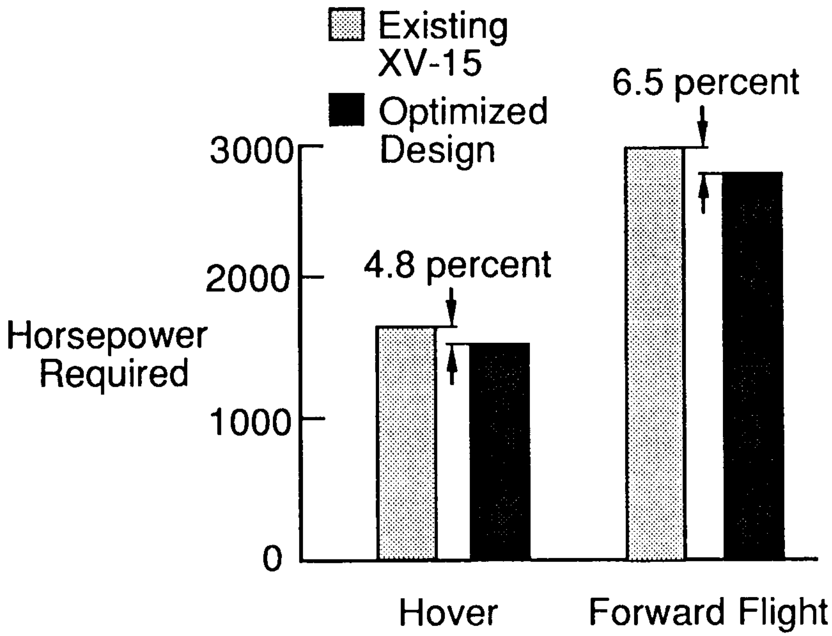


Figure 19 (b).

TDT OSCILLATING FLOW FIELD MEASURED FOR HELICOPTER ROTOR GUST STUDIES

Paul H. Mirick (Army), M-Nabil Hamouda (PRC), and William T. Yeager, Jr. (Army)
Configuration Aeroelasticity Branch

RTOP 505-63-51

Research Objective - The gust response of helicopter rotors has been the subject of several analytical studies; however, there have been few experimental efforts to investigate the gust response of helicopter rotors. A unique feature of the Transonic Dynamics Tunnel (TDT) is the oscillating vanes located upstream of the test section which can be used to generate a sinusoidal gust field. An objective of a recent test was to measure gust velocities which would be experienced by a model rotor tested on the Aeroelastic Rotor Experimental System (ARES) model in the TDT. The secondary objective of this test was to check out the operation of a new oscillating vane controller.

Approach - Measurements of the flow field generated by the oscillating vanes were made using flow direction vanes. Flow measurements were taken at 15 locations in the tunnel test section. These locations were selected to measure the flow in the region swept by a rotor on the ARES model. During the test, the oscillating gust vanes were operated at a nominal amplitude of ± 3 degrees at frequencies from 1 to 18 Hz. Preliminary results showed minimal flow field change when the vanes were oscillated above 10 Hz; therefore, data were only acquired at frequencies from 1 to 10 Hz. This test was conducted in a Freon atmosphere at a density of .006 slugs/ft^3 for a tunnel dynamic pressure from 10 to 50 psf. These conditions are typical of an ARES test.

Accomplishment Description - Sample results from this investigation are shown in figure 20(b). The data obtained showed a variation in gust amplitude in the spanwise direction. The gust amplitude decreased with oscillating vane frequency and with tunnel station. These results agreed with previous TDT oscillating vane calibration efforts.

Significance - This test verified the proper operations of the new gust vane controller. The test results document the gust field that would be experienced by a rotor on the ARES model and will be used for future tests which examine the gust response of helicopter rotors.

Future Plans - Tests will be conducted with the ARES model installed and the gust vanes operating to determine helicopter rotor gust responses.

Figure 20 (a).

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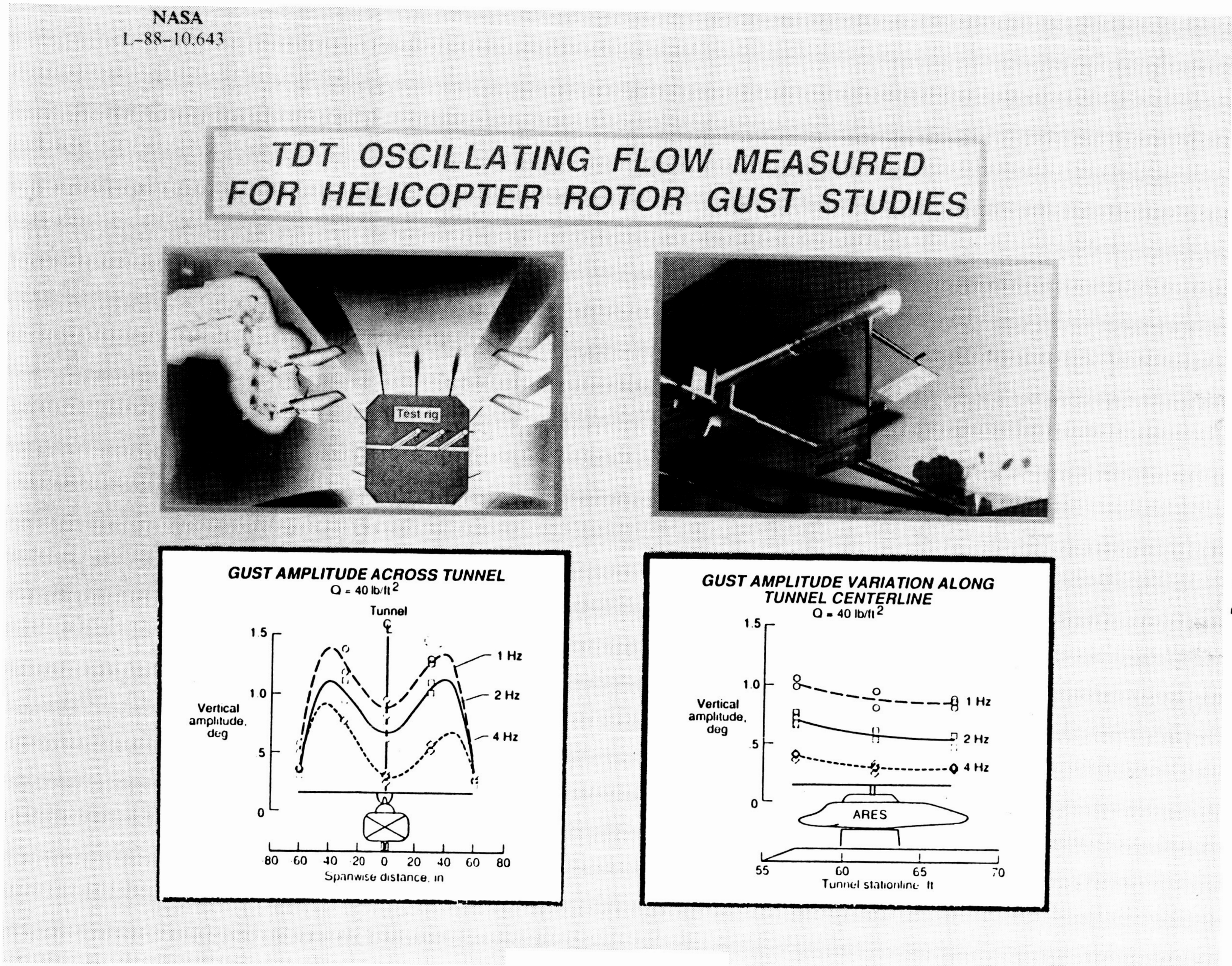

Figure 20 (b).

MOTORIZED PITCH LINK DEVELOPED FOR IMPROVED BLADE TRACKING IN TDT

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Configuration Aeroelasticity Branch

RTOP 505-63-51

Research Objective - In the testing of helicopter rotor models it is desirable for the rotor blades to be in track (that is the blades of the rotor should follow each other). If the blades are out of track, undesired loads and vibrations may be produced. The method used for tracking model rotors on the Aeroelastic Rotor Experimental System (ARES) is to visually observe the tip path of the blades with the assistance of a strobe light timed to fire four times per rotor revolution. This allows an observer to determine the tip location of each blade relative to a reference blade. Based on the observation, adjustments are made to the pitch link lengths. This affects the angle of attack of the blade and thus changes the tip path of the blade. Each rotor tracking adjustment requires about 15 minutes to complete. However, because ARES is tested in the TDT which uses a Freon gas test medium, additional time (about two hours) is required to remove and then replace Freon in the test section (considered a tunnel entry). The objective therefore is to develop a motorized pitch link which allows remote adjustment of the blade tracking and, thereby, eliminating the tunnel entries.

Approach - Pitch links have been designed and fabricated whose lengths can be remotely adjusted (Fig. 2(b)). These pitch links use a small DC electric motor to drive a jack screw. This jack screw allows the pitch link length to be changed up to 0.5 inch. This amount of adjustment is more than adequate to track the rotor.

Accomplishment Description - The motorized pitch links were successfully demonstrated and used on the ARES model during a recent tunnel test conducted in the TDT. As a result of using the links during this test, some minor changes were made to the pitch link design to improve their operation.

Significance - The TDT tunnel test was done to investigate the effect of rotor track on rotor loads and performance. The use of the motorized pitch links allowed the rotor to be more precisely tracked than could have been possible by the old tracking method and saved a number of tunnel entries to get the initial rotor track. Tracking adjustments to obtain data for five rotor out-of-track configurations were also made by using the motorized pitch links. During the two-week test at least six tunnel entries were eliminated saving a total of about 16 hours of Freon processing time and energy.

Future Plans - The new parts for the updated pitch-link design are being fabricated. Pitch links will be used on all future ARES tests which require precise tracking.

Figure 21 (a).

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L-88-11,735

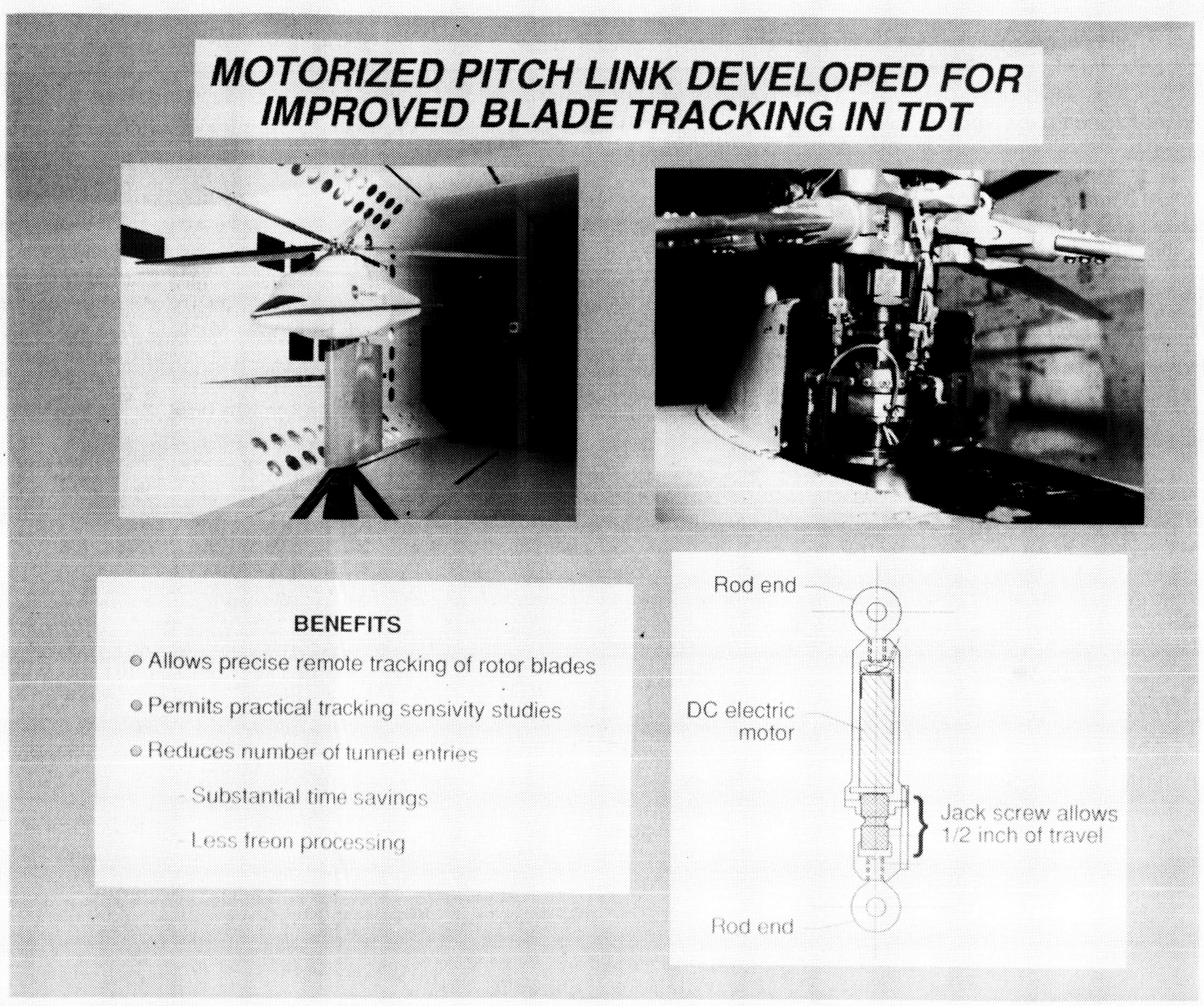


Figure 21 (b).

FORWARD FLIGHT ROTOR TRACKING CHARACTERISTICS INVESTIGATED IN TDT

W. Keats Wilkie (Army)
Configuration Aeroelasticity Branch

RTOP 505-63-51

Research Objective - Once-per-revolution vibrations due to out-of-track rotor blades have long been problems on helicopters. In an ongoing effort to identify the most significant causes of these blade out-of-track problems, and to aid in ultimately developing the analytical means for predicting track behavior, a series of forward flight helicopter model tests was recently conducted in the Langley Transonic Dynamics Tunnel (TDT). The principal objectives of these tests were (a) to acquire baseline forward flight rotor flapping data to compare with computer analysis, and (b) to examine the forward flight track sensitivity of several configurations to known adjustments in blade root pitch.

Approach - Three rotor inertial configurations were tested in the TDT using the NASA-Army Aeroelastic Rotor Experimental System (ARES). Data taken during these tests was compared to analysis performed with the Comprehensive Analytical Model of Rotorcraft Aerodynamics and Dynamics (CAMRAD) computer code. Model hardware consisted of two sets of four untapered, untwisted, NACA 0012 airfoil rotor blades. An articulated rotor hub was used throughout the test. Blade flapping inertias were varied by altering the distributions of insertable tungsten and aluminum weights inside each blade. For each configuration collective pitch sweeps were conducted in hover and at advance ratios of 0.10, 0.20, and 0.30. All collective sweeps were performed with the control system swashplate leveled, i.e., with no application of cyclic pitch trim and with no shaft angle. Blade flapping motion was recorded with potentiometers mounted at the flapping hinges for two of the four blades. Rotor performance coefficients were measured with a six component strain gage balance. Mean and first harmonic (1P) flapping angles were plotted against the rotor thrust coefficient (C_T) in hover and at each advance ratio for all configurations. Hover and forward flight blade track sensitivity to incremental root pitch adjustments was also examined.

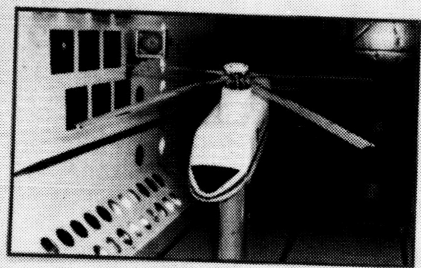
Accomplishment Description - Examples of some of the forward flight rotor track data obtained during this test are shown in figure 22(b). The illustration on the lower left is a plot of 1P flapping magnitude measured for one configuration versus C_T at a forward flight advance ratio of 0.20. CAMRAD analysis for this condition is shown also. For this particular condition and configuration CAMRAD seems to provide a relatively good estimate of the observed 1P flapping. CAMRAD estimates of mean flapping (not shown here) in hover and forward flight are significantly higher than the experimental data. Trends, however, seem to be adequately represented. In the example on the lower right a single rotor blade has been driven out-of-track by mechanically offsetting the root pitch of the blade by two known increments. The resulting differences in 1P flapping magnitude between the offset blade and a reference blade at a given advance ratio and C_T were recorded. The 1P flapping differences shown here are for an advance ratio case of 0.20. This type of data is useful in estimating the relative track sensitivity of a particular configuration to pitch link adjustment. This particular plot also illustrates some of the nonlinear track effects that can be induced by a relatively small amount of blade root pitch in forward flight.

Significance - Data obtained during this test has been most important in evaluating CAMRAD's ability to predict and analyze rotor flapping behavior in hover and forward flight. CAMRAD's suitability for use in an analytical study of rotor tracking sensitivity is being assessed.

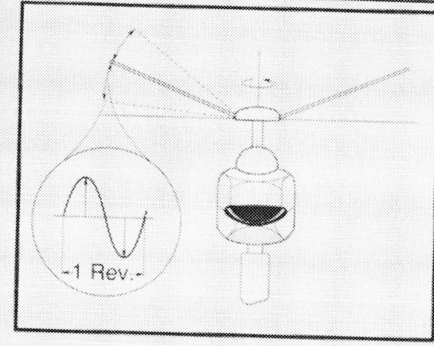
Future Plans - Additional forward flight testing in the TDT to examine some previously observed flapping characteristics in detail is planned for the near future. Results of these tests will be documented in a NASA formal publication.

Figure 22 (a).

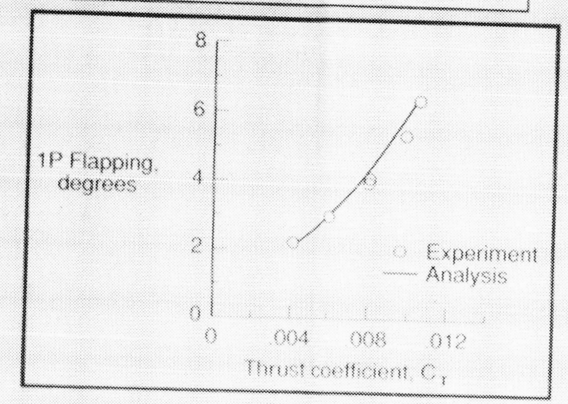
FORWARD FLIGHT ROTOR TRACKING CHARACTERISTICS INVESTIGATED IN TDT



FIRST HARMONIC (1P) FLAPPING ON ROTOR IN FORWARD FLIGHT



CAMRAD FLAP MOTION ANALYSIS VS. EXPERIMENT



ROTOR TRACK SENSITIVITY TO ROOT PITCH ADJUSTMENT MEASURED

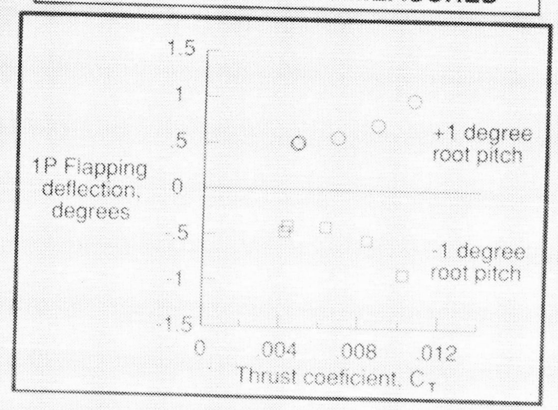


Figure 22 (b).

TDT TESTS EVALUATE PERFORMANCE CHARACTERISTICS OF ADVANCED DESIGN HELICOPTER ROTOR BLADES

Kevin W. Noonan (Army), William T. Yeager, Jr. (Army), Matthew L. Wilbur (Army), Paul H. Mirick (Army)
Configuration Aeroelasticity Branch

RTOP 505-63-51

Research Objective - A cooperative program with Bell Helicopter Textron to couple advanced aerodynamic blade shapes with a blade structure designed for minimum hub vibratory loads is in progress. One objective of this effort is the evaluation of the effects of rotor planform shape on aerodynamic performance.

Approach - Two sets of rotor blades were designed for specified aerodynamic performance requirements: 1) a rectangular planform blade designed by Bell, and 2) a tapered planform blade designed by Army Aerostructures Directorate personnel. Each blade set was also structurally designed to result in minimum hub shears and moments. The thrust-weighted solidity, twist distribution, and inboard airfoil section are the same for both blade sets. Tests were conducted in the Langley Transonic Dynamics Tunnel (TDT) because of its unique ability to use Freon-12 as a test medium. Both rotors were 1/5-size aeroelastically scaled models and were tested at conditions simulating various rotor tasks defined by aircraft gross weight and propulsive force requirements up to an advance ratio of 0.425. At each test condition, main-rotor torque measurements were made to evaluate performance between the 2 configurations.

Accomplishment Description - Illustrative results are shown in the figure 23(b). These results were obtained for a hover tip Mach number of 0.646, which represents sea level standard conditions. The hover data indicates that the tapered planform provides a performance improvement for thrust coefficients both above and below the design value of 0.0051. The forward flight data also indicates a performance advantage for the tapered blades at advance ratios above 0.15. The hover performance trend is consistent with pretest predictions while the forward flight trend is not entirely consistent with analytical results (not shown).

Significance - The use of a highly tapered rotor blade can result in hover improvements without degradation of forward flight performance relative to a rectangular blade. The inability of the design analysis to correctly predict forward flight performance trends may lead to modification or abandonment of this analysis in the future.

Future Plans - These test results will be documented in a formal NASA publication. A correlation of these measured performance trends with analytical results from an analysis not used in the design process will also be conducted.

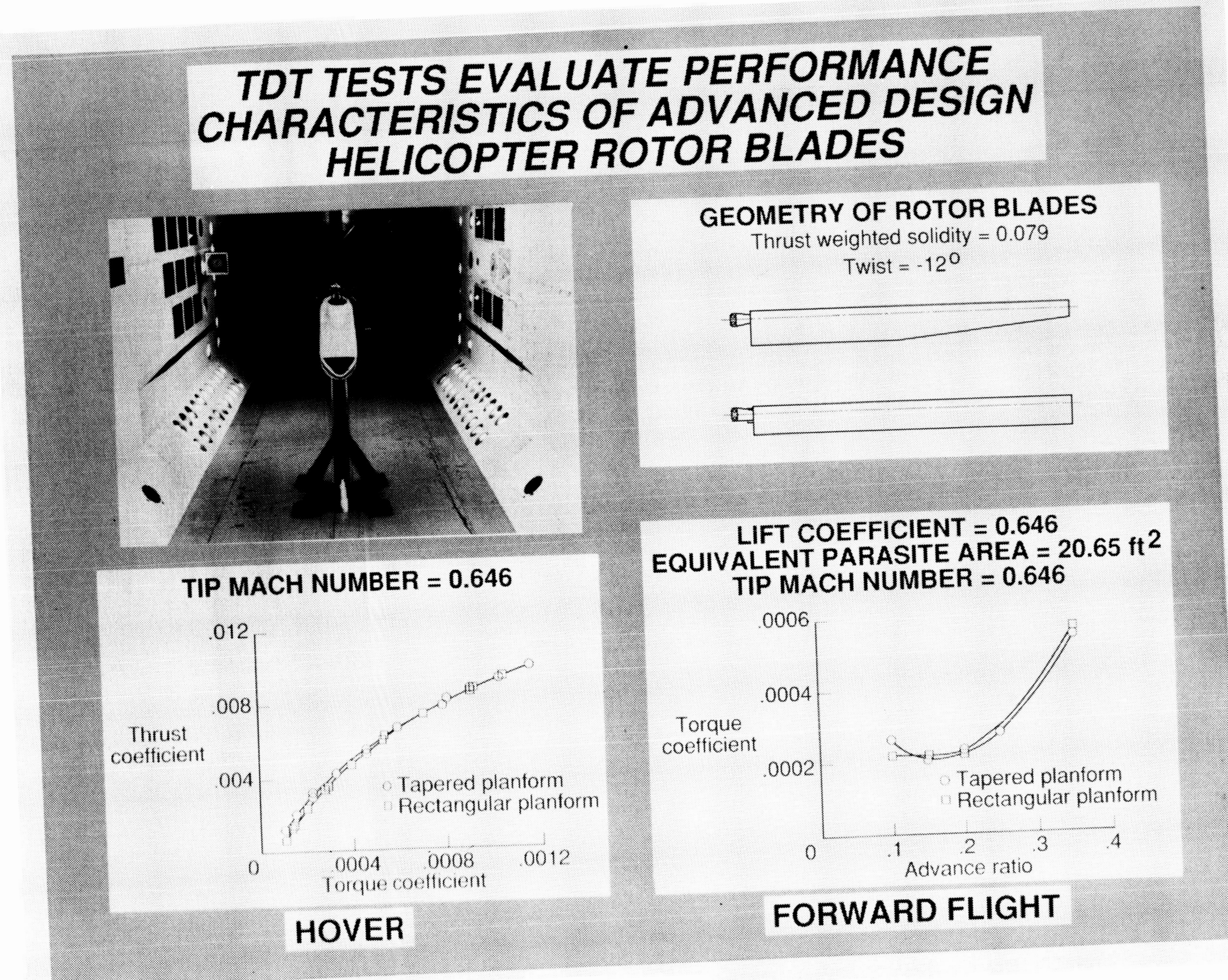

TDT TESTS EVALUATE PERFORMANCE
CHARACTERISTICS OF ADVANCED DESIGN
HELICOPTER ROTOR BLADES
GEOMETRY OF ROTOR BLADES
Thrust weighted solidity = 0.079
Twist = -12°
TIP MACH NUMBER = 0.646
Thrust coefficient
.012
.008
.004
0
.0004
.0008
.0012
Torque coefficient
○ Tapered planform
□ Rectangular planform
HOVER
LIFT COEFFICIENT = 0.646
EQUIVALENT PARASITE AREA = 20.65 ft2
TIP MACH NUMBER = 0.646
Torque coefficient
.0006
.0004
.0002
0
.1
.2
.3
.4
Advance ratio
○ Tapered planform
□ Rectangular planform
FORWARD FLIGHT

Figure 23 (b).

OPTIMIZATION APPROACH FOR HELICOPTER VIBRATION REDUCTION DEMONSTRATED

T. Sreekanta Murthy (PRC)
Configuration Aeroelasticity Branch

RTOP 505-63-51

Research Objective - Considerable emphasis to achieve significantly lower vibrations in the advanced helicopters under development has brought about an increased need to establish better analytical methods to be used during design for vibration reduction. A vibration reduction method which has gained attention recently is based on the idea of designing the helicopter airframe structure so as to minimize the vibration responses in the airframe under the action of rotor-induced loads. The design of an airframe structure for vibration reduction involves extensive study of the airframe dynamic characteristics to guide the modification of vibration related design parameters. The design study requires multidegree of freedom structural analysis, multidimensional search of design variables, and multidisciplinary considerations. In this regard, the use of extensively-developed structural optimization tools can be beneficial in such studies and hence needs to be explored. Therefore, the objective of the present study is to investigate and develop analytical and computational tools for optimization of helicopter structures for vibration reduction.

Approach - The nonlinear mathematical programming approach is being pursued for optimization of helicopter airframe structures. Successful application of this approach to real helicopter airframe structures requires a detailed investigation of several important areas. Some of the areas addressed in the study are (1) mathematical formulation of the airframe optimization problem, including establishment of a relevant set of design variables, constraints and objective function; (2) analytical methods of sensitivity analysis which are unique to helicopter airframe structures; (3) computer implementation of optimization procedures; and (4) demonstration of the approach to real helicopters.

Accomplishment Description - A computer program system called DYNOPT for dynamics optimization of airframe structures for vibration reduction has been recently developed. The DYNOPT code features a unique operational combination of the MSC/NASTRAN finite element structural analysis code, extended to include calculation of steady-state dynamic response sensitivities with the CONMIN optimizer. Initial application of the DYNOPT program to a Bell AH-1G helicopter airframe structure has been completed. Several airframe structural design models of varying complexity and differing in the number and types of design variables have been developed. Figure 24(b) schematically indicates one such design model (referred to as a preliminary design model) which considers the distribution of the depth of the primary structure as design variables. Typical numerical results obtained for the model using the program are also shown in the figure. The sensitivity analysis results indicate that a reduction in forced response displacement at the pilot seat can be brought about by reducing the depth of the tail structure and increasing the depth of the forward fuselage structure. The design iteration histories indicate a reduction in the vibration acceleration at the pilot and gunner locations and also a reduction in the structural weight as a result of the airframe optimization.

Significance - The demonstration of the airframe optimization approach for vibration reduction proves feasibility of the approach and provides a basis for the helicopter industry to pursue the design of low vibration airframes.

Future Plans - The overall research study will be extended to include structural damping, computational methods for large-scale structural optimization and multidisciplinary airframe design considerations.

Figure 24 (a).

OPTIMIZATION APPROACH FOR HELICOPTER VIBRATION REDUCTION DEMONSTRATED

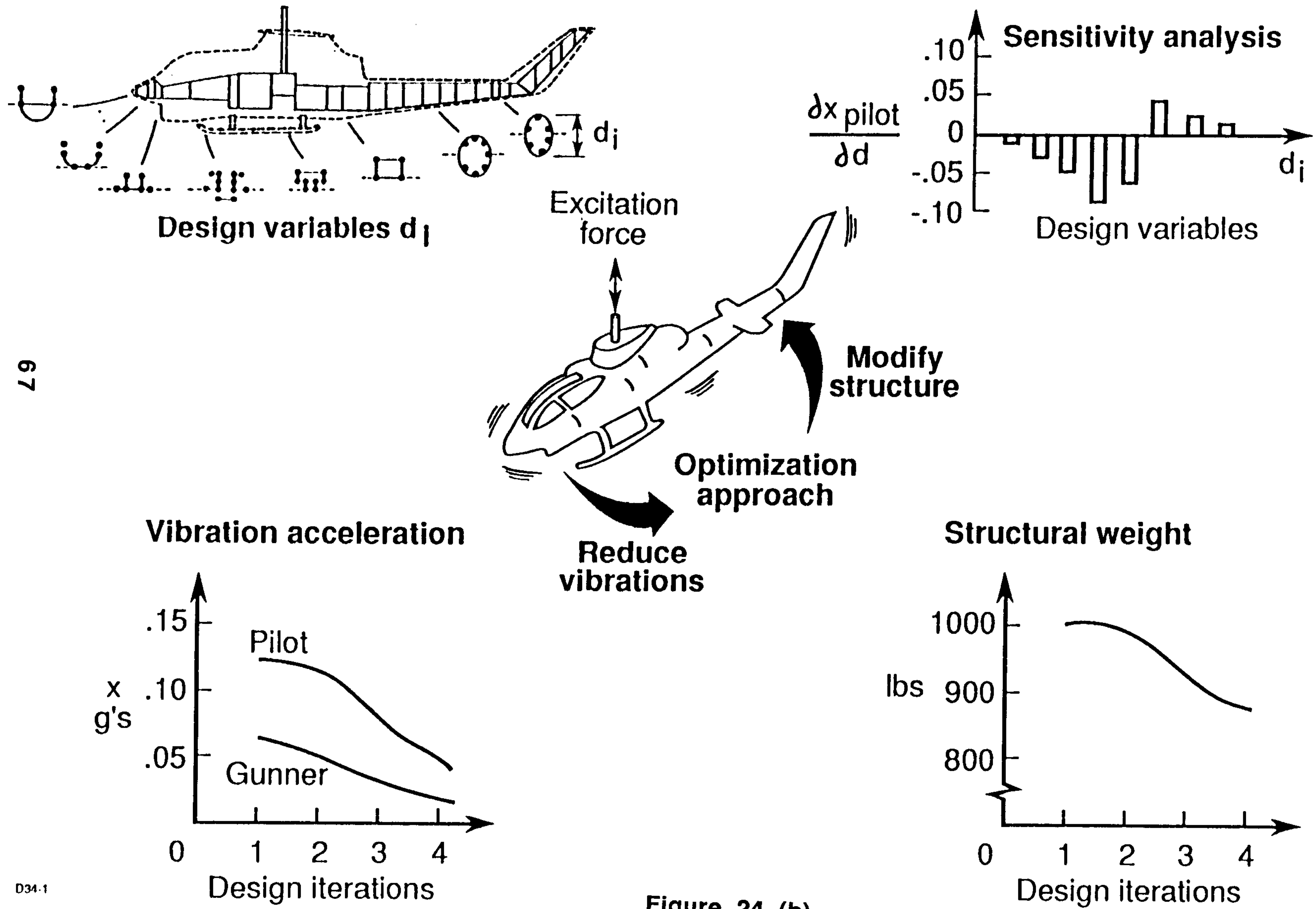


Figure 24 (b).

GROUND VIBRATION TEST OF HELICOPTER AIRFRAME IDENTIFIES IMPORTANT CONTRIBUTORS TO VIBRATORY RESPONSE

Raymond G. Kvaternik
Configuration Aeroelasticity Branch

RTOP 505-63-51

Research Objective - Excessive vibration is the most common technical problem to arise as a "show stopper" in the development of a new rotorcraft. With only a few exceptions, vibration problems have not been identified until flight test. Solutions at that stage of development are usually add-on fixes which adversely impact cost, schedule, and vehicle performance. Vibration predictions have not been relied on by the industry during design because of deficiencies in current vibration analysis methods. Because of increasing demands for further reductions in vibrations, it is recognized that vibration must be seriously addressed during design. With a view toward establishing a capability in the industry to fully utilize vibration analysis during design, the NASA Langley Research Center has underway a program, designated DAMVIBS (Design Analysis Methods for VIBrationS), with the overall objective of establishing the foundations for developing a superior design analysis capability for vibrations. Among the many activities being conducted under the DAMVIBS program is one which is aimed at identifying those "difficult components" which are the important contributors to airframe vibratory response and which require more detailed finite element representation.

Approach - Typically, only the primary (major load carrying) structure is represented fully (stiffness and mass) when forming the finite element model (FEM) of an airframe. Many components (e.g., transmission, engines, and stores) and secondary structure (e.g., fairings, doors, and access panels) are represented as lumped masses. This may be a major factor in the poor agreement which has been obtained between test and analysis at the higher frequencies of interest. To isolate the effects of each component on overall vibratory response, multiple ground vibration tests will be conducted with each test representing a progressive removal of the suspect component until only the primary airframe structure remains. At each stage, analyses will be performed using an existing FEM of the airframe modified as necessary to reflect the specific configuration tested. Both full-scale airframes and their components and small-scale generic models of both metal and composite construction would be studied.

Accomplishment Description - The initial effort in this area was conducted by Bell Helicopter Textron utilizing an AH-1G helicopter airframe. To isolate the effects of each component on vibratory response, ground vibration tests were conducted on 8 aircraft configurations. Figure 25(b) shows the aircraft in its complete and fully stripped down configurations. In addition to the aircraft tests, separate tests were conducted on several of the components. The results given in the figure show the importance of secondary panels and canopy glass on airframe response at the higher frequencies.

Significance - The difficult components studies on the AH-1G helicopter represent the first such studies conducted on an airframe structure. The results obtained have identified several components which are important contributors to airframe vibratory response at the higher frequencies of interest and which require improved representation in the FEM. This indicates that finite element models for vibrations work may need to be more detailed than models for static internal loads analysis. Because the effects are configuration dependent, the studies need to continue.

Future Plans - The difficult components studies are to continue through a combination of tests and analyses utilizing both full-scale composite airframes and their components as well as small-scale generic models of both metal and composite construction.

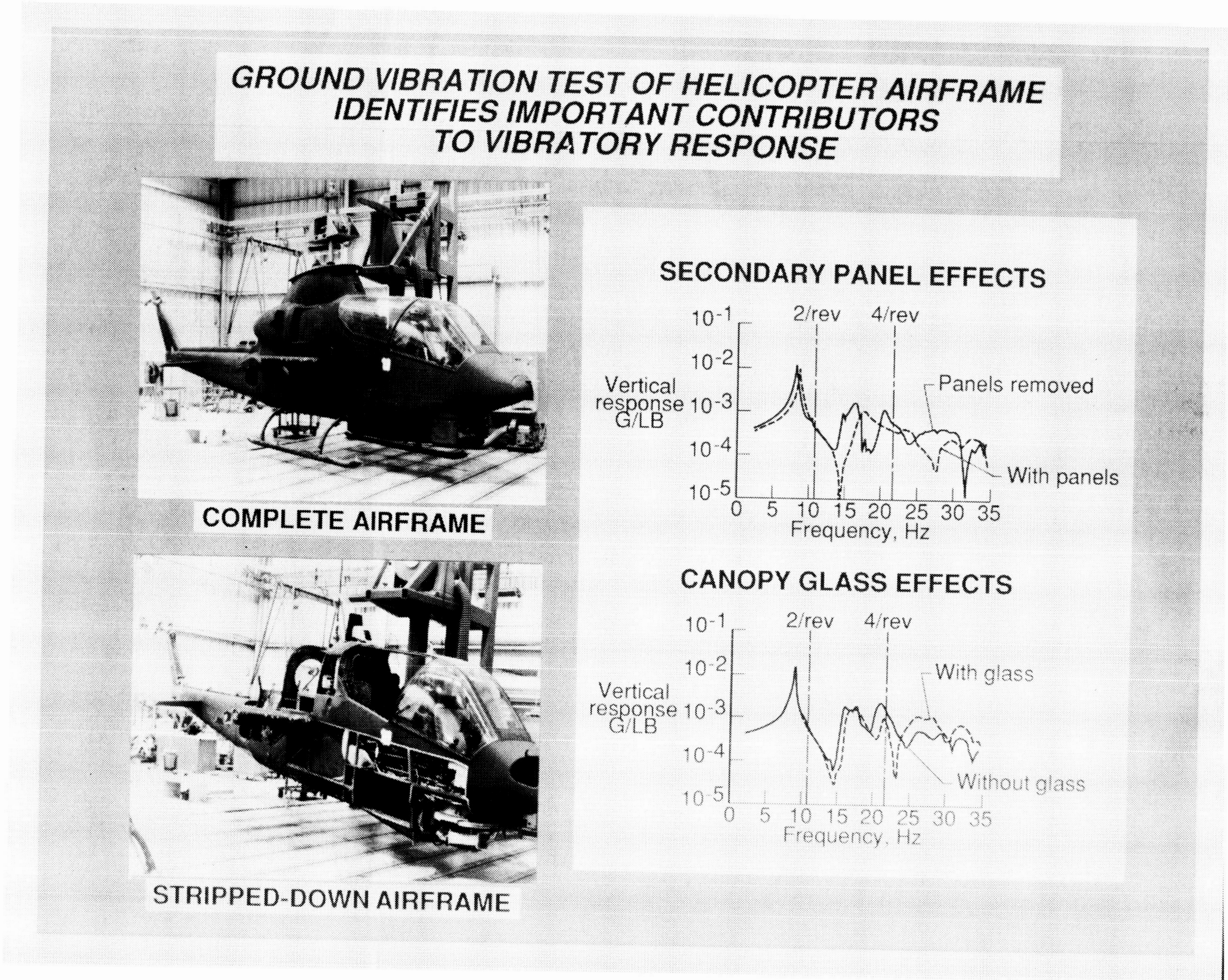

GROUND VIBRATION TEST OF HELICOPTER AIRFRAME
IDENTIFIES IMPORTANT CONTRIBUTORS
TO VIBRATORY RESPONSE
COMPLETE AIRFRAME
STRIPPED-DOWN AIRFRAME
SECONDARY PANEL EFFECTS
Vertical response G/LB
10-1
10-2
10-3
10-4
10-5
2/rev
4/rev
Panels removed
With panels
0 5 10 15 20 25 30 35
Frequency, Hz
CANOPY GLASS EFFECTS
Vertical response G/LB
10-1
10-2
10-3
10-4
10-5
2/rev
4/rev
With glass
Without glass
0 5 10 15 20 25 30 35
Frequency, Hz

Figure 25 (b).

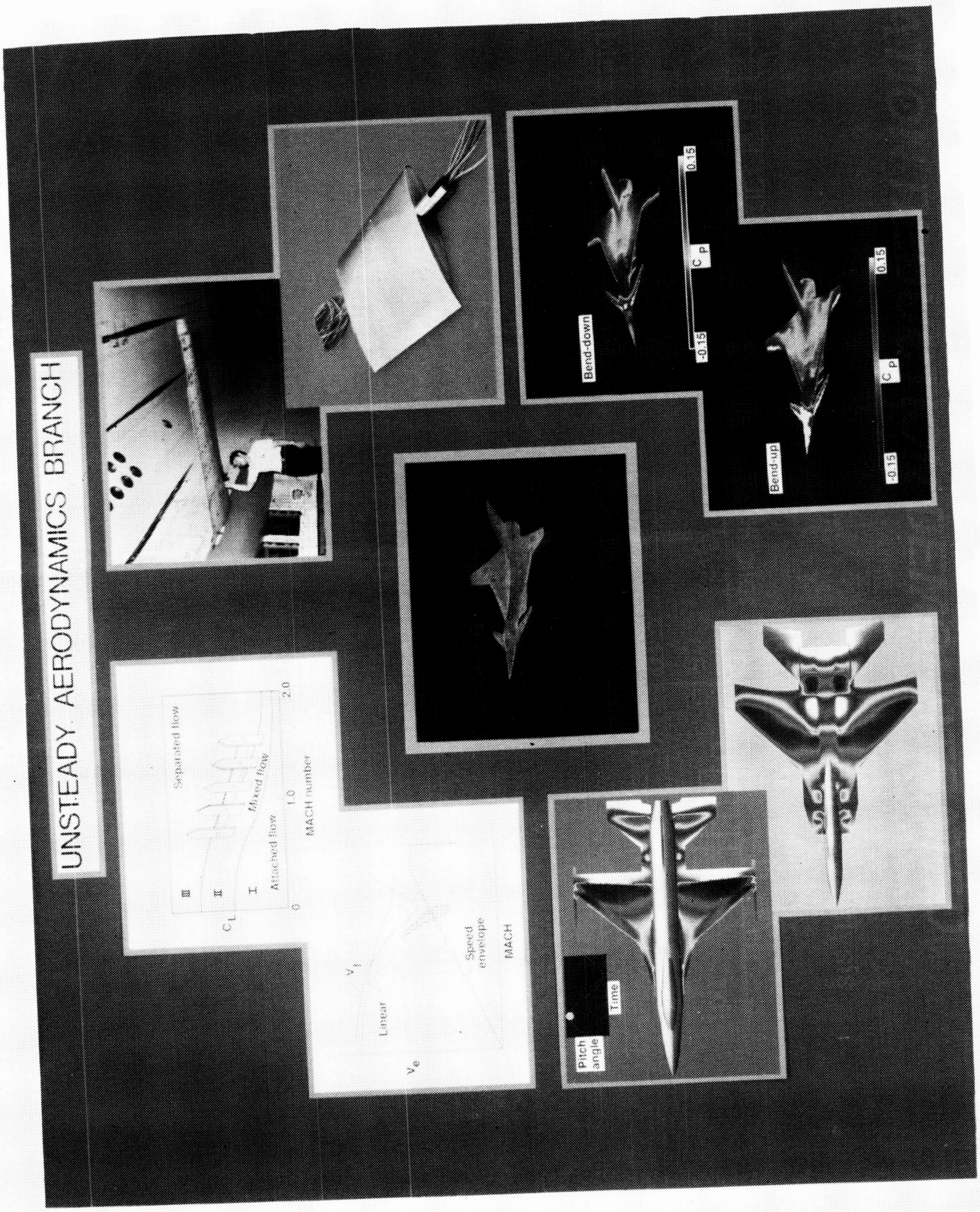

UNSTEADY AERODYNAMICS BRANCH
Separated flow
Mixed flow
Attached flow
MACH number
Speed envelope
MACH
Linear
Bend-down
Bend-up
Pitch angle
Time

Figure 26.

UNSTEADY AERODYNAMICS BRANCH

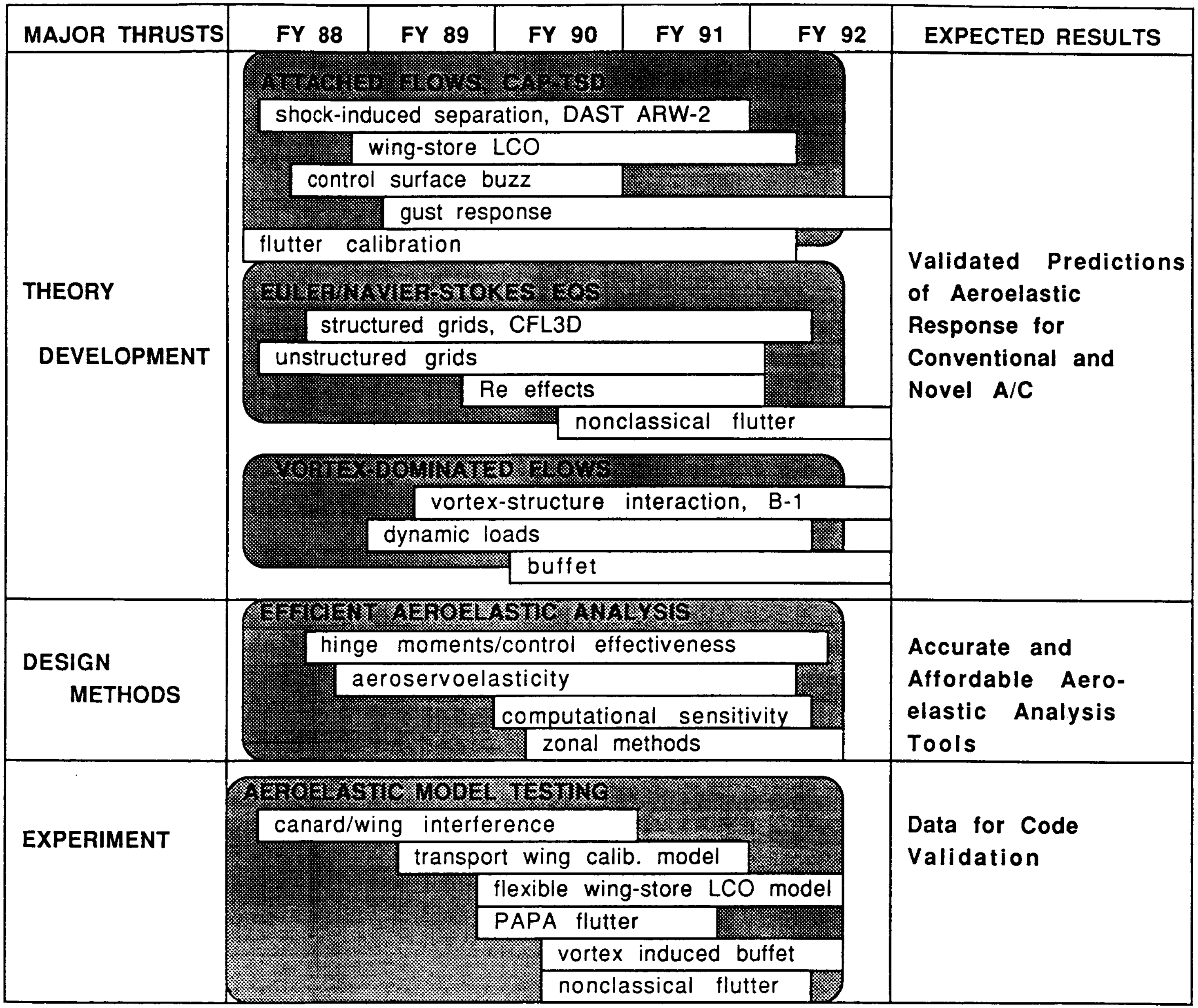


Figure 27.

SOLID STATE DISC VERSION OF CAP-TSD DEVELOPED

John T. Batina and Robert W. Neely (PRC)
Unsteady Aerodynamics Branch

RTOP 505-63-21

Research Objective - Applications of the CAP-TSD code to complete aircraft configurations require a large amount of computer memory. For example, an F-16C half-span aircraft model requires 14.7 million words of memory. Since most of the U.S. aerospace companies have four million word Cray X-MP computers, there is a need for a smaller-memory version of CAP-TSD. Therefore, this research objective was to modify CAP-TSD for use on X-MP computers with solid-state disc (SSD) for high speed external storage, to minimize the required in-core memory of the program.

Approach - The approximate factorization algorithm within the CAP-TSD code was modified to eliminate most of the 3-D arrays by restructuring the way in which the algorithm sweeps through the computational grid to solve the governing flow equations. In the original code, the algorithm requires three sweeps through the grid at each time step, one sweep in each coordinate direction. During each sweep, the solution is determined in the other two coordinate directions which requires that all of the flow field information be in core memory. The algorithm was thus modified so that in each sweep, the solution is determined in only one of the other two coordinate directions at a time, which reduces significantly the amount of information which must be in core memory. The remaining information is then alternately written and read from SSD as a part of the new solution procedure. The modifications made to the code, however, reduce the length of vectors which control the speed of solution and consequently CPU time is increased.

Accomplishment Description - For a coarse grid wing-alone test case containing 45,600 grid points, the in-core memory was reduced from 2.3 to 1.0 million words with an increase in CPU time of a factor of 1.5 on an X-MP computer with SSD, as indicated in figure 28(b). For larger problems such as the F-16C calculation with 324,000 grid points, the memory was reduced from 14.7 to 3.9 million words.

Significance - With the SSD version of CAP-TSD run on a Cray X-MP computer, decreases in the required memory of a factor of from two to four with a corresponding increase in CPU time of at most a factor of two were obtained. Therefore, the U.S. aerospace companies can now run CAP-TSD on their four million word X-MP computers for transonic aeroelastic analysis of complete aircraft configurations.

Future Plans - The SSD version of CAP-TSD is now available for release to industry upon request.

Figure 28 (a).

SOLID STATE DISC VERSION OF CAP-TSD DEVELOPED

- **STRONG INDUSTRY INTEREST IN 4 MW CRAY X-MP VERSION OF CODE**
  - McDonnell Aircraft Company
  - Boeing Commercial Airplane Company
  - General Dynamics
  - Near Inc.

- **OBJECTIVE: 14.7 MW F-16C HALF-SPAN AIRCRAFT MODEL ON 4 MW CRAY X-MP**

- **3 MONTH CONVERSION COMPLETED ON TIME**
  - CPU execution time increase less than factor of 2

- **SUPPORTS LARGER GRID CALCULATIONS ON LARGER MEMORY MACHINES**

MACHINE CAPACITIES

- CDC VPS-32	32 MW
- CRAY II (NAS)	256 MW
- CRAY X-MP	(2-8) MW
- CRAY Y-MP	32 MW

Configuration	Grid Size	Core Mem,MW Orig./Mod.
Wing alone, coarse grid	46K	2.3/1.0
F-16C half-span model	324K	14.7/3.9

Figure 28 (b).

STRIP BOUNDARY LAYER CAPABILITY IMPROVES ACCURACY OF CAP-TSD RESULTS

James T. Howlett
Unsteady Aerodynamics Branch

RTOP 505-63-21

Research Objective - The objective of this research is to develop accurate and economical techniques for predicting unsteady transonic airloads for fully 3-D configurations. Techniques are sought which are capable of calculating flows with significant viscous effects, including flow fields with a moderate amount of unsteady flow separation and reattachment.

Approach - An unsteady 3-D transonic small disturbance computer code called CAP-TSD has been developed recently for aerodynamic and aeroelastic applications. Transonic calculations with the CAP-TSD code for fully 3-D configurations have been quite accurate for flows with negligible viscous effects. As Mach numbers increase, shock waves increase in strength and viscous effects must be included for accurate predictions of aerodynamic loading. An interactive strip boundary layer method which includes an inverse calculation for moderately separated flows has been incorporated into the code. With the inclusion of this boundary layer technique, the CAP-TSD computer code may be applied to flows with significant viscous effects, including flows with moderate amounts of separation.

Accomplishment - The CAP-TSD computer code with viscous corrections has been applied to calculate steady pressure distributions on the F-5 wing at Mach number $M = 0.95$ and $\alpha = 0°$. The steady pressure distribution shown on the left part of figure 29(b) indicates a moderately strong shock wave on the aft part of the wing. The calculated boundary layer displacement thickness shows a sudden increase across the shock wave. The comparison with experiment shown on the right part of the figure for a station near mid span indicates a significant improvement in the calculated pressures due to the inclusion of viscous effects, particularly in the vicinity of the shock at $x/c \approx 0.8$.

Significance - The good agreement between the experimental results and the CAP-TSD viscous calculations indicates that the CAP-TSD computer code with viscous corrections can be used for accurate predictions of aerodynamic loads on 3-D configurations with significant viscous effects. As a result, the range of applicability of the CAP-TSD computer code is significantly extended.

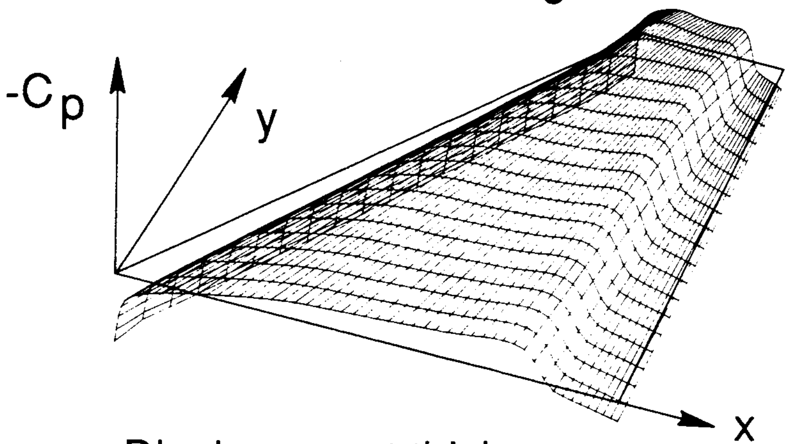
Future Plans - Applications of the CAP-TSD viscous computer code are underway to determine the limits of validity of the improved method. These applications will include 3-D configurations with moderate amounts of flow separation.

Figure 29 (a).

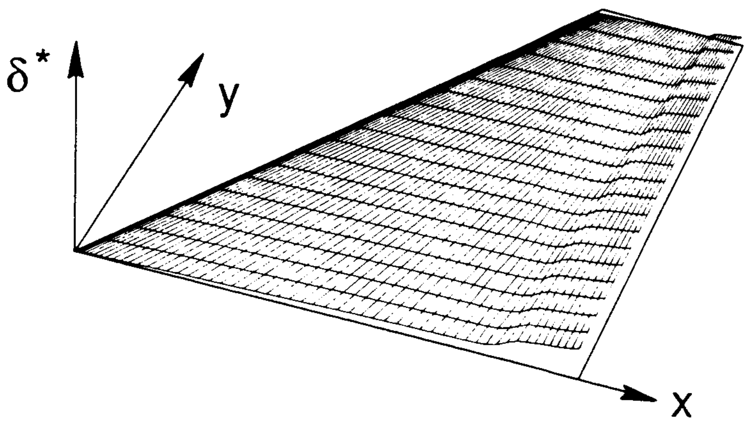
STRIP BOUNDARY LAYER CAPABILITY IMPROVES ACCURACY OF CAP-TSD RESULTS

- Viscosity modeled with Lag-entrainment boundary layer
- Improved agreement with measured pressures near shocks

Steady pressure on F-5 wing
$M = 0.95, \ \alpha = 0$ deg



Displacement thickness



Comparison with experiment
$\eta = 0.512$

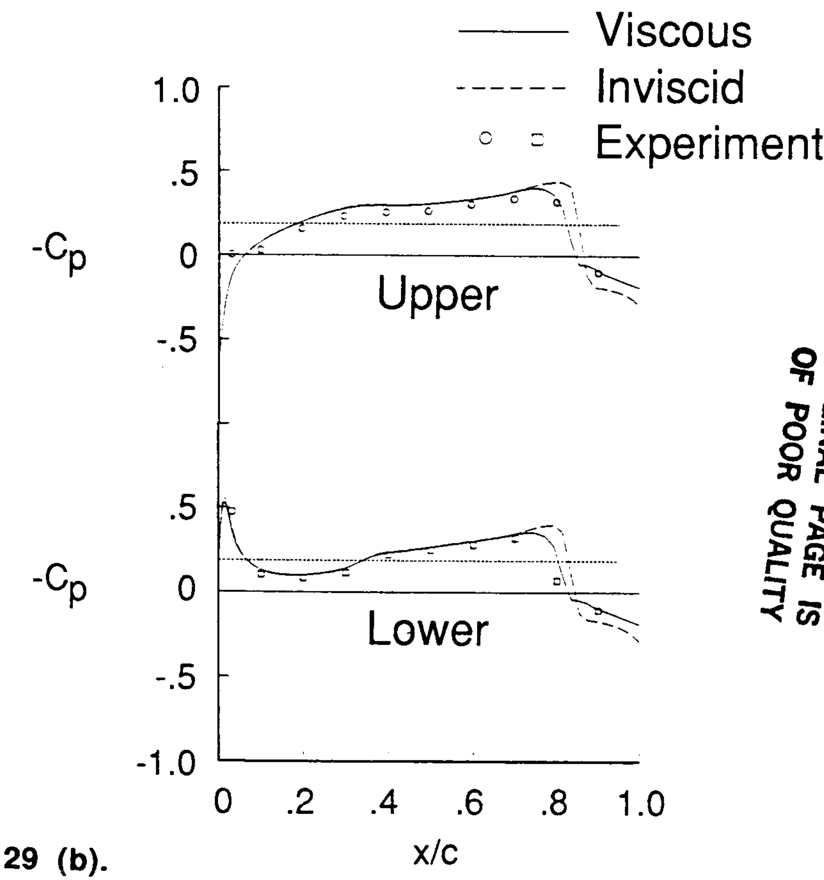


Figure 29 (b).

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SUPERSONIC FAR-FIELD BOUNDARY CONDITIONS IMPROVE ACCURACY AND EFFICIENCY FOR AEROELASTIC CALCULATIONS

Michael D. Gibbons (PRC) and John T. Batina
PRC, Inc. and Unsteady Aerodynamics Branch

RTOP 505-63-21

Research Objective - The objective of this research was to improve the accuracy of a transonic small-disturbance code (CAP-TSD) for supersonic freestreams.

Approach - The supersonic boundary conditions were determined using the method of characteristics. A simple boundary condition was desired to avoid excessive computational time in solving the new boundary condition. Thus the method of characteristics was applied to the 2D steady full potential equation to determine the appropriate Riemann invariants to use along the far-field boundaries. These Riemann invariants were then implemented in the Computational Aeroelasticity Program - Transonic Small Disturbance program (CAP-TSD). The CAP-TSD code was developed for aeroelastic analysis of complete aircraft configurations in the transonic range.

Accomplishment Description - To assess the new supersonic boundary conditions, calculations were performed on the F-5 fighter wing. The F-5 wing has a leading edge sweep of 31.9°, a taper ratio of 0.28 and is approximately 4 percent thick. Calculations were performed at $M_\infty = 1.05$ and $\alpha = 0°$. Figure 30(b) shows a comparison of pressure distributions calculated using the supersonic boundary condition and the original boundary condition. The pressure distributions were calculated using a very small sectional x-z grid which contained 109 x 62 points and was 5 chordlengths square. When the original boundary conditions are used the pressures are overpredicted. Using the supersonic boundary condition brings the calculated pressure distributions into very good agreement with experiment.

Significance - The supersonic boundary conditions allow the use of small computational grids for supersonic freestreams without losing accuracy in the solution. Using smaller grids also reduces the computational cost.

Future Plans - More calculations for steady as well as unsteady cases will be done to further assess the supersonic boundary conditions.

Figure 30 (a).

SUPERSONIC FAR-FIELD BOUNDARY CONDITIONS IMPROVE ACCURACY AND EFFICIENCY FOR AEROELASTIC CALCULATIONS

- Inhibits contamination of solutions by boundary reflections
  - Smaller grids
  - Reduced costs

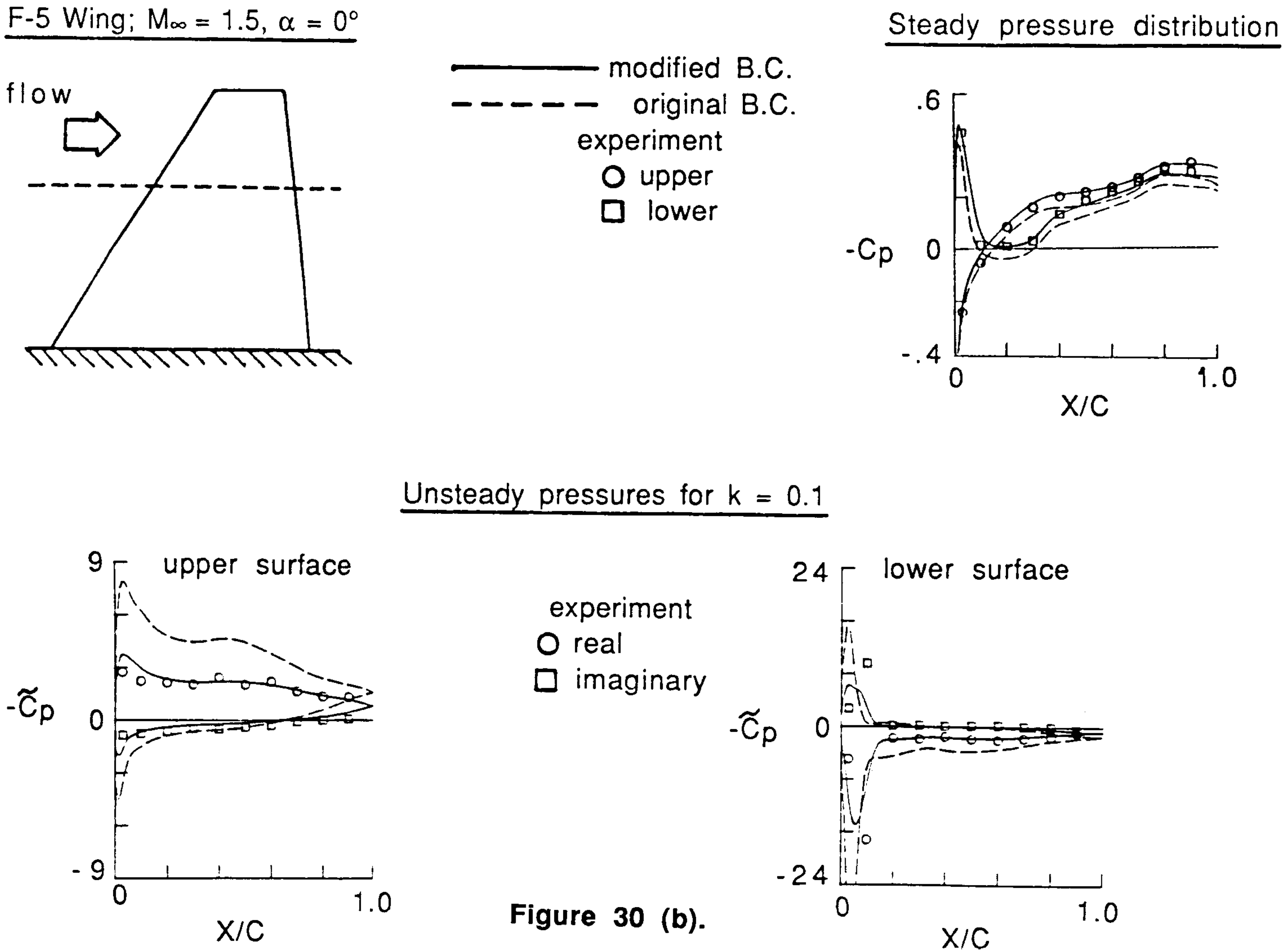


Figure 30 (b).

UNSTRUCTURED DYNAMIC GRID METHOD DEVELOPED FOR AEROELASTIC ANALYSIS WITH HIGH LEVEL CFD CODES

John T. Batina
Unsteady Aerodynamics Branch

RTOP 505-63-21

Research Objective - In computational fluid dynamics (CFD) codes which solve the higher governing fluid flow equations, such as the Euler or Navier-Stokes equations, grids are employed which conform to the surface of the geometry under consideration. For unsteady applications where the geometry moves or deforms, the grid must move to maintain alignment with the instantaneous position of the geometry. The objective of this research was to develop a dynamic grid method which allows the grid to move for complete aircraft configurations.

Approach - To allow the treatment of complete aircraft configurations, the method was developed for use with unstructured grids. Such grids involve a large number of tetrahedral cells which are assembled in an unstructured fashion to conform to the surface of the aircraft. The advantage of this approach lies in the ability to freely vary the cell size and cell density to treat regions of large flow gradients. For structured grids such "grid-refinement" leads to denser grids throughout the flow field whereas the unstructured approach allows the refinement to be done only locally with a significant saving in total number of grid cells. In the dynamic grid method, the outer boundaries of the original grid are held fixed and the instantaneous location of the inner boundary (aircraft) is prescribed. At each time step, the interior points of the grid are moved by modeling each edge of each tetrahedron by a spring with the spring stiffnesses inversely proportional to the length of the respective edge. The static equilibrium equations of the resulting spring network are then solved at each interior grid point for the displacements in the three coordinate directions. This procedure is repeated at each time-step such that the grid follows the dynamic motion of the aircraft.

Accomplishment Description - An example application of the dynamic grid method is shown in the lower left of figure 31(b). The calculation was performed for the Pathfinder I configuration undergoing a sinusoidal rigid pitching motion. The oscillation amplitude was 15° and the figure shows the grid at the maximum displacement angle. The grid moves smoothly even for such large displacements of the aircraft, and the method can also treat aeroelastic deformations such as the bending oscillations depicted in the lower right of the figure.

Significance - The unstructured dynamic grid method can treat arbitrary motions and aeroelastic deformations of complete aircraft geometries. It is more general and accurate than previous methods based on interpolation or grid shearing.

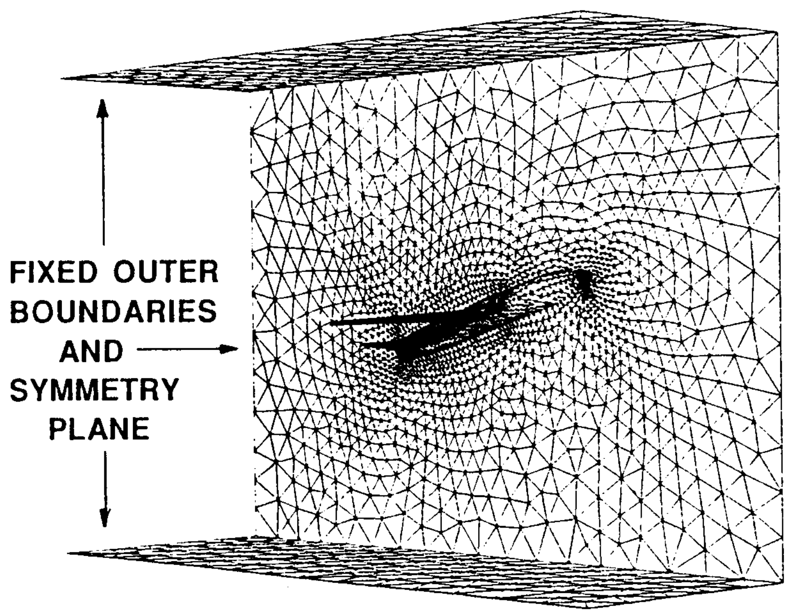
Future Plans - The dynamic grid method is being used within a new time-accurate Euler code based on unstructured grids for unsteady aerodynamic analysis of a supersonic fighter configuration.

Figure 31 (a).

UNSTRUCTURED DYNAMIC GRID METHOD DEVELOPED FOR AEROELASTIC ANALYSIS WITH HIGH LEVEL CFD CODES

- O UNSTRUCTURED GRIDS FOR TREATMENT OF COMPLEX GEOMETRIES
  - tetrahedron grid cells clustered in regions of high gradients
- O GRID MOVES TO MAINTAIN ALIGNMENT WITH FLEXING BODY
  - stretching of "steady" grid via spring analogy
- O MORE GENERAL AND ACCURATE THAN METHODS BASED ON INTERPOLATION OR SHEARING

TRANSPORT MODEL PITCHING NOSE-UP 15 DEG



BODY SURFACE GRIDS FOR BENDING OSCILLALTIONS

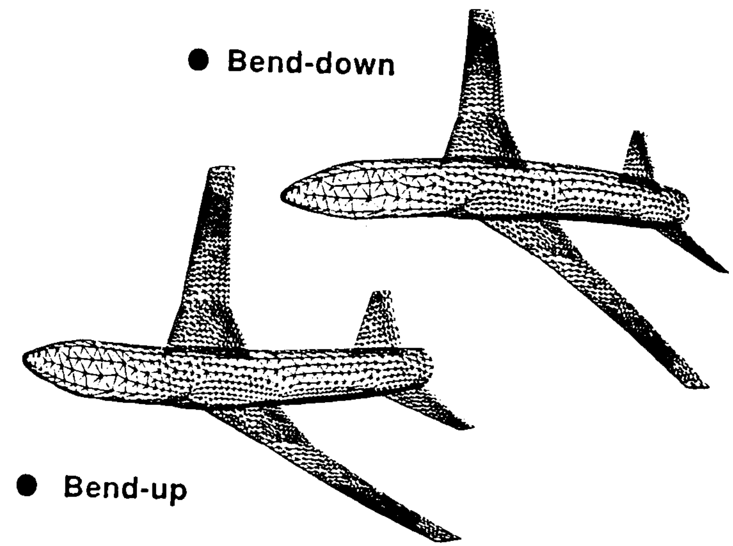


Figure 31 (b).

UNSTEADY EULER ALGORITHM DEVELOPED BASED UPON DYNAMIC UNSTRUCTURED GRID METHODOLOGY

John T. Batina
Unsteady Aerodynamics Branch

RTOP 505-63-21

Research Objective - The objective of the research was to develop a numerical algorithm for the solution of the Euler equations for unsteady aerodynamic analysis of complex aircraft configurations.

Approach - The Euler equations are solved using a finite-volume algorithm developed for analysis with unstructured grids made up of tetrahedra. The unstructured grid methodology was used to allow the treatment of complex aircraft geometries. The flow solver involves a multi-stage Runge-Kutta time-stepping scheme to march the Euler equations in time. By solving the Euler equations, strong shock and rotational flow effects are modeled correctly. A novel aspect of the capability is the dynamic mesh algorithm that is employed for problems involving static or dynamic deformation of the aircraft. The dynamic mesh algorithm is a general procedure that can move the original mesh to maintain alignment with the instantaneous surface of the aircraft and can treat arbitrary motions and deformations of complex aircraft configurations.

Accomplishment Description - Calculations were performed for the supersonic fighter configuration that is shown in the lower left part of figure 32(b). The grid that was used contained 13,832 nodes and 70,125 tetrahedra for the half-span airplane. Steady flow results were obtained for the fighter at a freestream Mach number of $M = 2$ and an angle of attack of $\alpha = 0°$. The resulting steady pressure coefficient (C_p) contours on the aircraft surface are shown in the lower right part of Fig. 32(b). The pressures indicate flow compression on the forward part of the aircraft nose, canopy, and canards, with expansion on the aft part of the canopy. The outboard region of the cranked wing and the vertical tail show similar flow features with compression along the leading edges and expansion along the trailing edges which is typical for supersonic flow. Unsteady flow results for the fighter oscillating harmonically in a complete vehicle bending mode, as shown in the left half of figure 32(c), were obtained at a reduced frequency based on wing tip semichord of $k = 0.1$. Calculated instantaneous pressure contours at the maximum (bend-up) and minimum (bend-down) amplitudes of oscillation are shown in the right half of figure 32(c). During the first half of the cycle when the aircraft bends up, there is a significant increase in the level of flow compression along the nose and canopy as well as in the outboard region of the wing. During the latter half of the cycle, the opposite situation occurs.

Significance - The results presented demonstrate that time-accurate inviscid flows can now be computed for complex aircraft configurations undergoing arbitrary motion or structural deformation.

Future Plans - The aeroelastic equations of motion will be implemented to allow flutter analysis, and alternative solution algorithms are being investigated to improve the efficiency of the present procedures.

Figure 32 (a).

UNSTEADY EULER ALGORITHM DEVELOPED BASED UPON DYNAMIC UNSTRUCTURED GRID METHODOLOGY

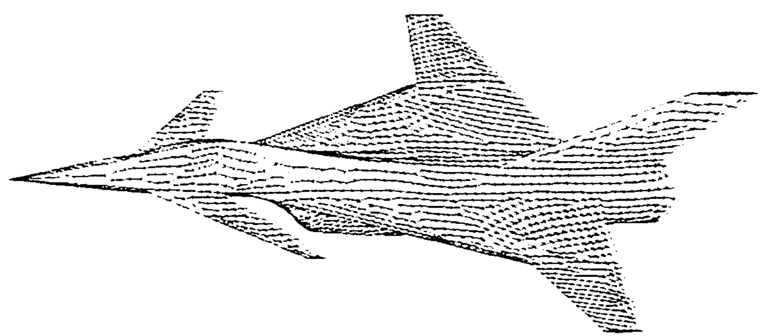
O UNSTRUCTURED GRIDS ALLOW TREATMENT OF GENERAL AIRCRAFT GEOMETRIES

- structured grids require many times more effort to develop

O STRONG SHOCK AND ROTATIONAL EFFECTS CORRECTLY MODELED

- dynamic grid method for correct boundary condition treatment of aeroelastic motions

SURFACE GRID ON SUPERSONIC FIGHTER



STEADY PRESSURES FOR M = 2, $\alpha = 0°$

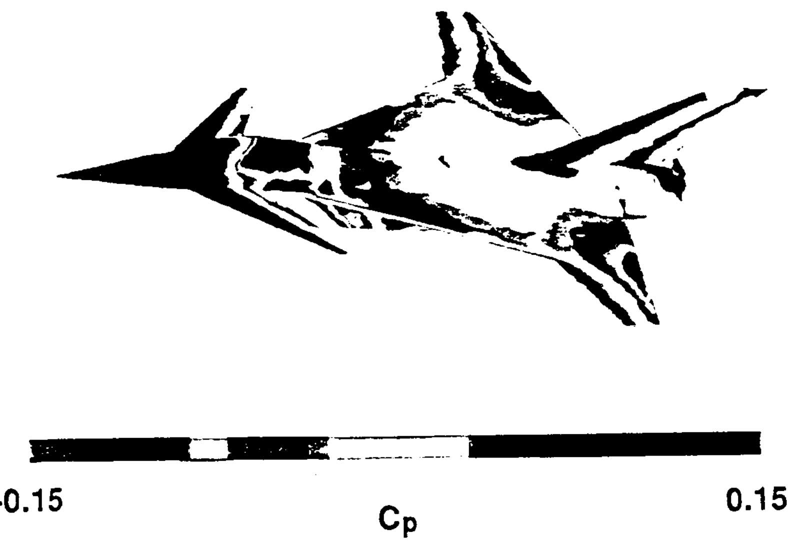


Figure 32 (b).

UNSTEADY EULER ALGORITHM DEVELOPED BASED UPON DYNAMIC UNSTRUCTURED GRID METHODOLOGY

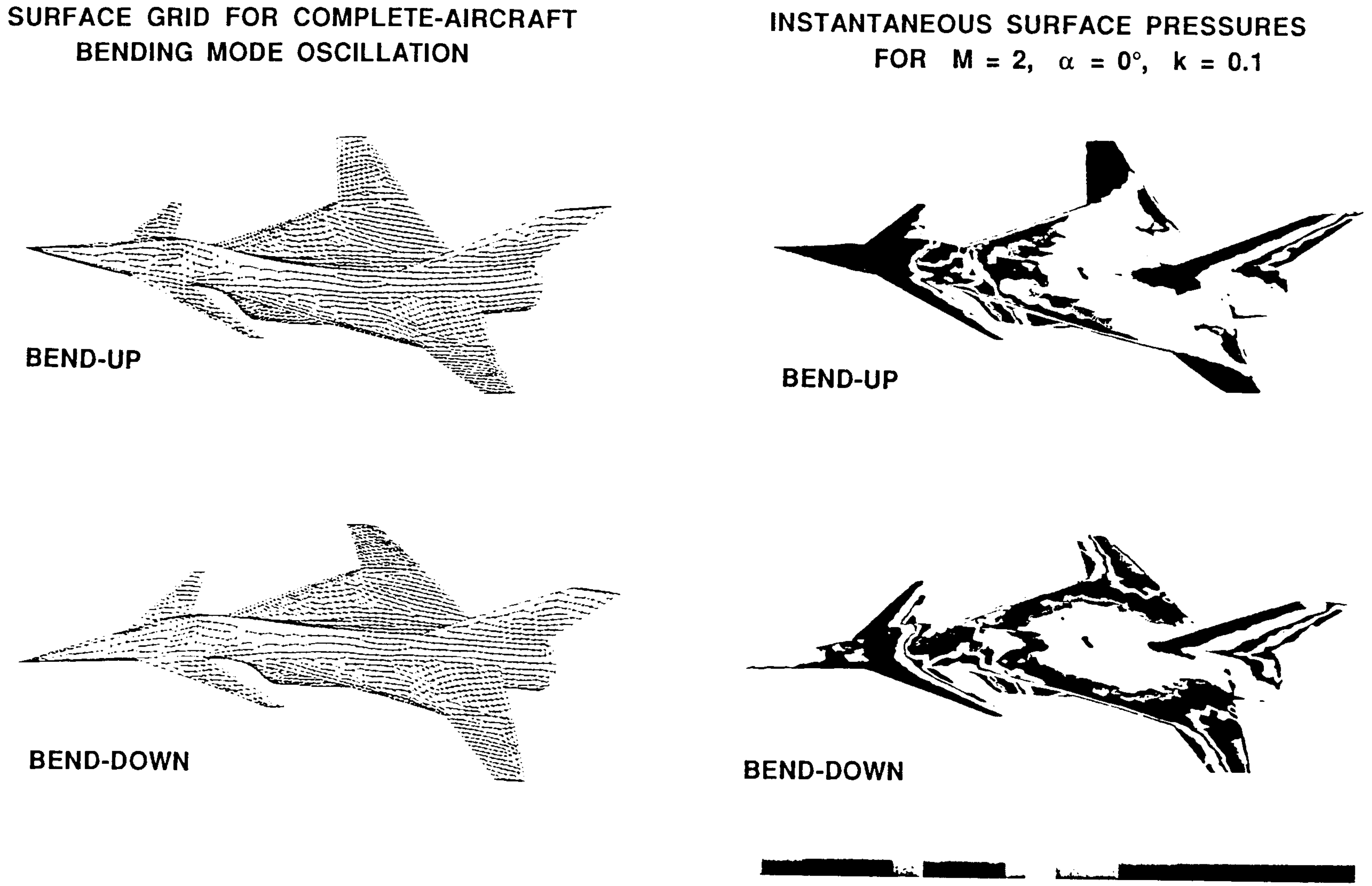


Figure 32 (c).

WING TORSIONAL FLUTTER FEATURES STUDIED WITH CAP-TSD

John T. Batina and Ross W. Mohr
Unsteady Aerodynamics Branch and Purdue University

RTOP 505-63-21

Research Objective - The objective of the research was to study the aeroelastic characteristics of a generic transport-type wing to gain insight into fundamental flow mechanisms responsible for flutter at transonic speeds.

Approach - The recently developed CAP-TSD (Computational Aeroelasticity Program - Transonic Small Disturbance) finite-difference code was used to calculate the steady and unsteady aerodynamics required by the aeroelastic analysis. The CAP-TSD code is an unsteady transonic small-disturbance program developed for transonic aeroelastic analysis of complete aircraft configurations. The wing selected for the present study was the Royal Aircraft Establishment (RAE) wing "A" which has a symmetric RAE 101 airfoil section with a maximum thickness-to-chord ratio of 9%. As shown in the upper left portion of figure 33(B), the wing has a midchord sweep angle of 30° and an aspect ratio of six. The wing structural dynamics were modeled using simple bending and torsion modes and aeroelastic behavior was studied for Mach number ranging from 0.8 to 0.98. The torsion-mode node-line was assumed to be at the local midchord of the wing.

Accomplishment Description - Representative steady pressure distributions for M = 0.8 and 0.94 are shown in the upper right portion of the figure. At M = 0.8 the flow is subcritical and at M = 0.94 there are shock waves on the upper and lower wing surfaces. Dynamic pressure root-loci for M = 0.8 and 0.94 are shown in the lower left portion of the figure. These stability plots show the migration of bending and torsion modes as dynamic pressure is increased. At M = 0.8 the curves indicate a classical bending- dominated flutter behavior with the onset of flutter at V_f = 0.75. Between M = 0.80 and 0.94 the character of the flutter mode changes from bending-dominated to torsion-dominated. At M = 0.94 the flutter crossing originates from the wind-off torsion mode rather than the bending mode and occurs at a higher frequency. These aeroelastic characteristics are further explained by considering the flutter boundary versus Mach number shown in the lower right portion of the figure. Two dips in flutter speed are shown, the second of which is deeper than the first. The first dip, with minimum V_f near M = 0.92, is the "usual" transonic dip involving a bending-dominated flutter mode. The second dip is characterized by a single degree-of-freedom torsion oscillation of the wing. This single degree-of-freedom flutter occurred with increasing Mach number due to the change of aerodynamic damping in torsion, from positive to negative, as the steady shock location migrates across the torsion-mode node line, in the outboard region of the wing.

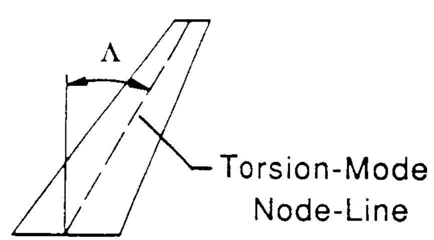
Significance - A novel flutter boundary with two transonic dips in flutter speed was computed and physical insight into the flow mechanisms which controlled the phenomenon was gained. Such double minimas with mode changes in flutter boundaries have been observed in transonic flutter model tests. Also, torsional "buzz" wing oscillations have been encountered experimentally. These calculations indicate the ability of the CAP-TSD code to treat such phenomena.

Future Plans - The work was conducted as part of a larger effort on aeroelastic analysis in the transonic speed range. Further calculations will study the dependence upon aeroelastic stability of structural dynamic and aerodynamic parameters such as torsion- mode node line location and lifting conditions.

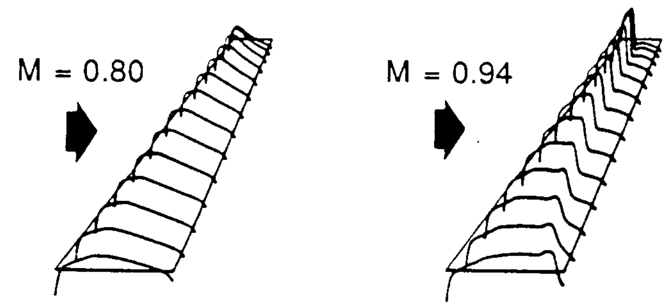
WING TORSIONAL FLUTTER FEATURES STUDIED WITH CAP-TSD

- Flutter mode change caused by aft movement of shock wave relative to torsion-mode node-line

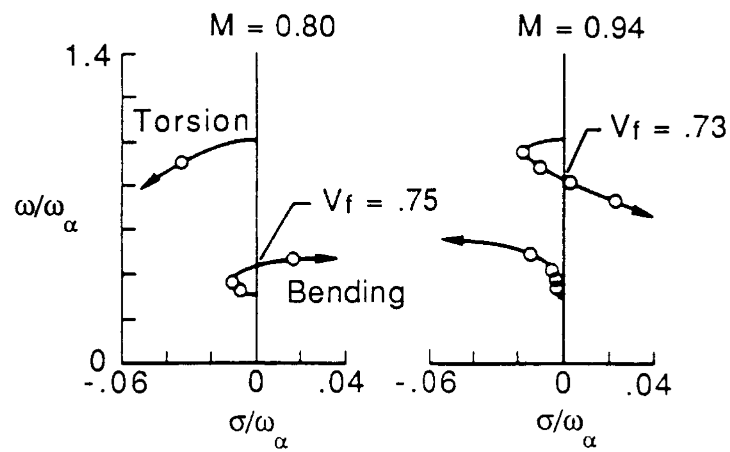
- RAE Wing A, Λ = 30°, AR = 6



- Steady pressure distributions



- Dynamic pressure root locus



- Flutter boundary

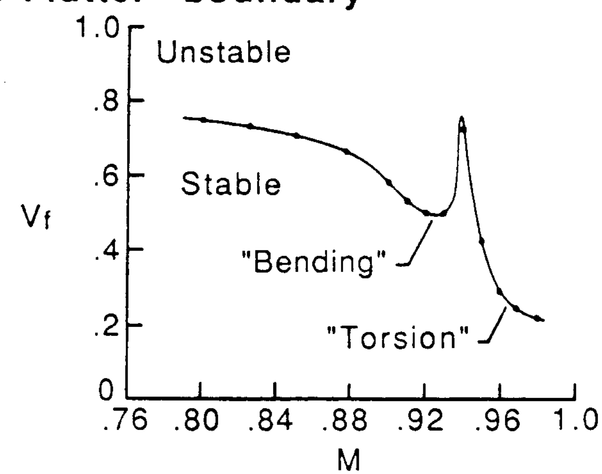


Figure 33 (b).

RIGOROUS CALIBRATION OF FLUTTER ANALYSIS METHODS FOR SLENDER DELTA WINGS

Michael D. Gibbons (PRC)
Unsteady Aerodynamics Branch

RTOP 505-63-21

Research Objective - The objective of this research is to determine the accuracy of predicting flutter for highly swept delta wings using linear aerodynamic theories, piston theory and a transonic small disturbance code.

Approach - Comparisons of flutter conditions are made between analytical results and experimental flutter values. The analytical methods used include FAST (based on a subsonic kernel function), ACUNN (based on a subsonic/supersonic kernel function), second order piston theory and CAP-TSD (based on transonic small disturbance theory).

Accomplishment Description - The flutter calculations were performed on three delta and three clipped delta wings with sweeps ranging from 54° to 80°. Flutter calculations were made over a range of Mach numbers M_∞ = 0.2 to 3.0. A finite element program was used to calculate the required mode shapes which were tuned to match the experimental frequencies. Comparisons of theory and experiment are shown in figure 34(b) for the 70° delta wing. Calculations using FAST show good agreement with experiment. The ACUNN results agree fairly well with experiment although a hump mode was present at M_∞ = 2.5 denoted by the flagged triangle. The piston theory results below M_∞ = 2.0 shows surprisingly good agreement even though it is not generally considered applicable in this region. The CAP-TSD results compare well with experiment in the subsonic-transonic region but become conservative in the supersonic region. The differences between CAP-TSD and experiment in the supersonic region may be due to the mounting system which was wedge shaped. The mounting system may have produced a bow shock thus altering the flow around the model.

Significance - The results show that fairly good agreement between calculated flutter boundaries and experiment can be achieved for highly swept delta wings in subsonic and supersonic flows.

Future Plans - Further calculations involving more complicated flows and geometries are planned using CAP-TSD and codes based on higher equation levels.

Figure 34 (a).

RIGOROUS CALIBRATION OF FLUTTER ANALYSIS METHODS FOR SLENDER DELTA WINGS

- Six delta/clipped delta wings; 54° - 80° sweep; 0.6 < M < 3.0

Classical theory comparison

Finite difference code comparison

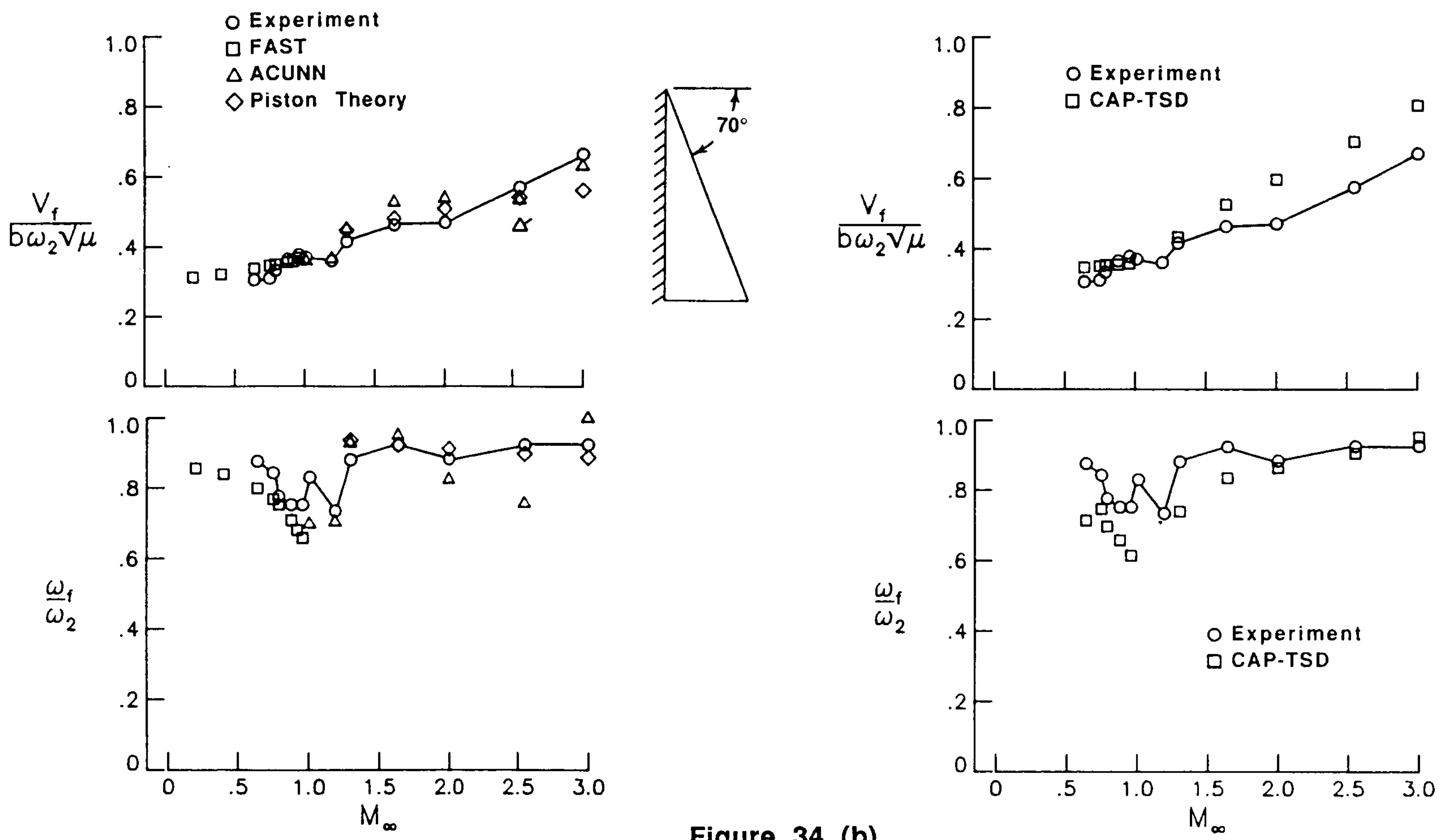


Figure 34 (b).

SUPERSONIC FLUTTER OF F-20 HORIZONTAL TAIL MODEL ACCURATELY PREDICTED

Woodrow Whitlow, Jr. and Wendy S. Pi
Unsteady Aerodynamics Branch and Northrop Aircraft Division

RTOP 505-63-21

Research Objective - The objective of this research is to assess the capability of the CAP-TSD finite difference code to predict flutter of wings at supersonic speeds.

Approach - Calculated and measured flutter conditions are compared for a 0.2 scale flutter model of the F-20 horizontal tail at Mach number 1.2. The model, whose planform is shown in the upper right of figure 35(b), has NACA 64A004 airfoil sections. The measured flutter conditions were obtained from tests in Canada's National Aeronautical Establishment's 5 ft. by 5 ft. blowdown wind tunnel. The calculated flutter conditions were obtained using the CAP-TSD code, which solves the unsteady transonic small disturbance potential equation to predict unsteady aerodynamic loads. In the CAP-TSD analysis, a cartesian grid with 90, 40, and 62 points in the chordwise, spanwise, and vertical directions was used to define the computational domain. The aeroelastic response was calculated using three measured natural vibration modes--a 92.45 Hz first bending mode, a 222.53 Hz first torsion mode, and a 329.6 Hz second bending mode. Initial disturbance velocities in one or more of the modes were used to excite the structure, and several cycles of the dominant flutter mode were calculated. Transients of the generalized coordinates then were analyzed to determine if the aeroelastic response was stable or unstable.

Accomplishment Description - Flutter conditions were measured experimentally and were calculated, with one percent modal damping, for the horizontal tail modeled to include thickness effects and for the tail modeled as a flat plate with zero mean angle of attack. Comparisons of the calculated and measured results are shown in the root locus plot in the lower right of the figure. CAP-TSD predicted the experimental flutter conditions within 1 percent in frequency and within 2 percent in speed. The flutter dynamic pressure predicted using using linear aerodynamic theory (flat plate analysis) is 20 percent lower than the measured value.

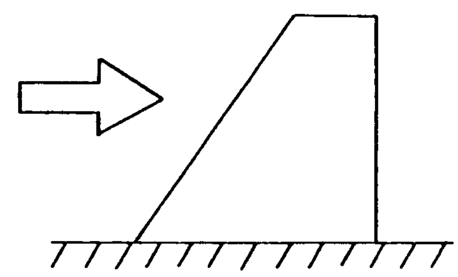
Significance - The results shown in the figure show the capability of the CAP-TSD code to predict accurately the flutter of wings at supersonic speeds. The comparisons with linear theory show the importance of including thickness effects in the supersonic flutter analysis.

Future Plans - Future efforts will include correlation of measured and calculated flutter conditions for a range of low supersonic Mach numbers.

Figure 35 (a).

SUPERSONIC FLUTTER OF F-20 HORIZONTAL TAIL MODEL ACCURATELY PREDICTED

- Model tested in Northrop blowdown tunnel
  - Experimental flutter dynamic pressure, $q_f = 24.5$ psi
- CAP-TSD flutter calculations predict $q_f = 23.5$ psi
  - 1% accuracy in flutter frequency
  - 2% accuracy in flutter speed
- Linear theory flutter result 20% low (conservative) in dynamic pressure



Root locus of flutter mode versus dynamic pressure; M = 1.2

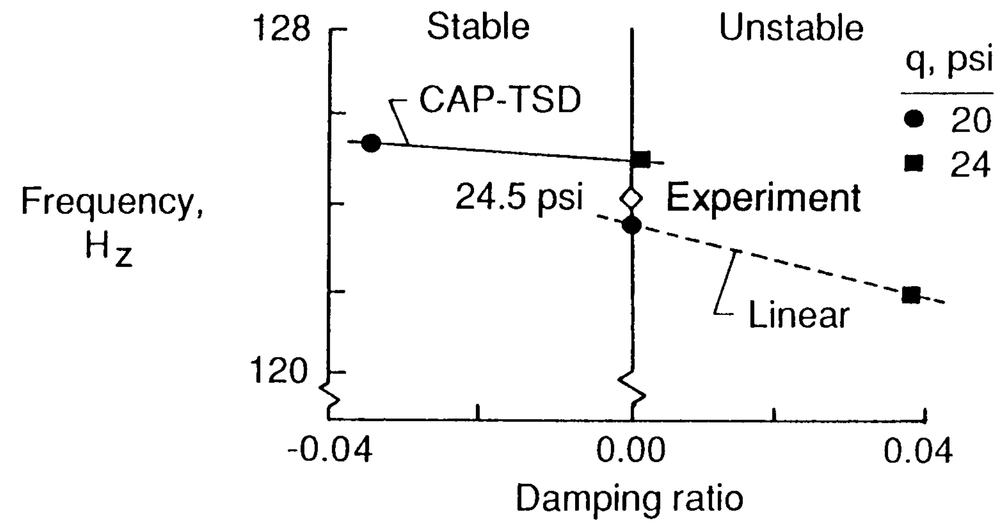


Figure 35 (b).

TSD POTENTIAL CODE PREDICTS P - 80 AILERON BUZZ

James T. Howlett
Unsteady Aerodynamics Branch

RTOP 505-63-21

Research Objective - The objective of this research is to determine if an unsteady transonic small disturbance computer code with an interactive boundary layer can be used to calculate the single degree-of-freedom flutter referred to as aileron buzz. An accurate and economical method is sought for predicting the unsteady transonic airloads associated with flow separation and reattachment frequently associated with the buzz phenomena.

Approach - Unsteady transonic flow fields with weak shock waves are routinely calculated with computer codes based on Transonic Small Disturbance (TSD) potential theory. As the shock waves increase in strength, viscous effects must be included for accurate predictions of aerodynamic loading. For attached flows, accurate predictions are obtained by interacting an integral boundary layer analysis with the inviscid solution. For separated or nearly separated flows, the interactive technique is based upon an inverse formulation of the boundary layer equations. This interactive technique has been implemented in the unsteady transonic small disturbance code XTRAN2L, enabling this code to be used for calculating separated flows.

Accomplishment Description - The XTRAN2L computer code with viscous corrections has been applied to the P-80 airfoil for calculating the self-induced limit cycle oscillations called aileron buzz. As figure 36(b) shows, for $M = 0.8$ and -1 degree angle of attack the steady pressures indicate moderately strong shock waves slightly upstream of the aileron hinge line with the upper surface shock downstream of the lower surface shock. When the aileron is released, a self-induced oscillation of approximately 6 degrees amplitude develops due to coupling between the aileron motion and the oscillating position of the shock waves. As shown by the buzz boundary in the lower right part of the figure, the XTRAN2L viscous calculations agree with the experimental wind tunnel results and the Navier-Stokes calculations.

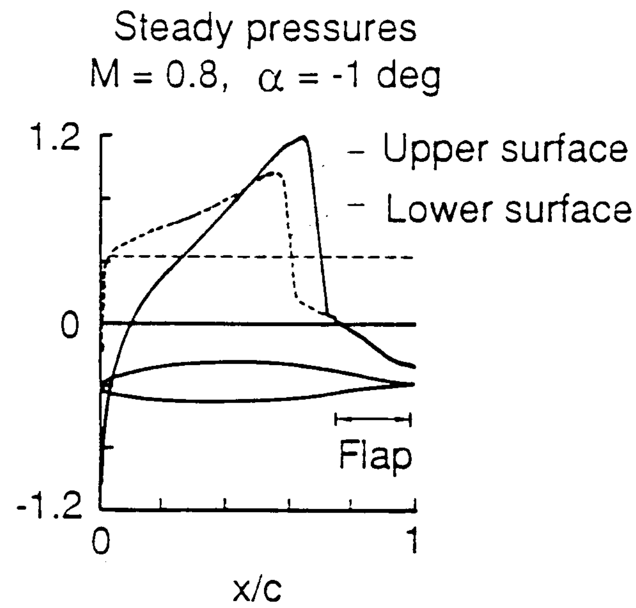
Significance - The close agreement between the present calculations, Navier-Stokes results and experimental data indicates that transonic small disturbance theory combined with the interactive boundary layer technique can be used to predict the onset of aileron buzz. The calculation of aileron buzz with the less expensive TSD methods offers a significant advantage for applications to aeroelastic analyses where many conditions must be computed.

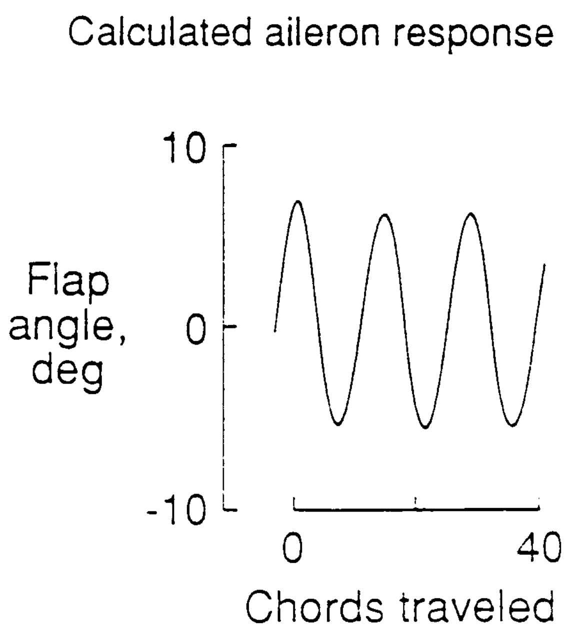
Future Plans - The computer code is being applied to various buzz conditions to develop improved understanding of the physical mechanisms involved in aileron buzz and to investigate quantitative methods for the prediction of buzz onset.

Figure 36 (a).

TSD POTENTIAL CODE PREDICTS P - 80 AILERON BUZZ

- Buzz limit cycle oscillations encountered at transonic speeds
- "Free" aileron response calculated with XTRAN2L-V
- XTRAN2L-V results match wind tunnel buzz and Navier-Stokes calculations





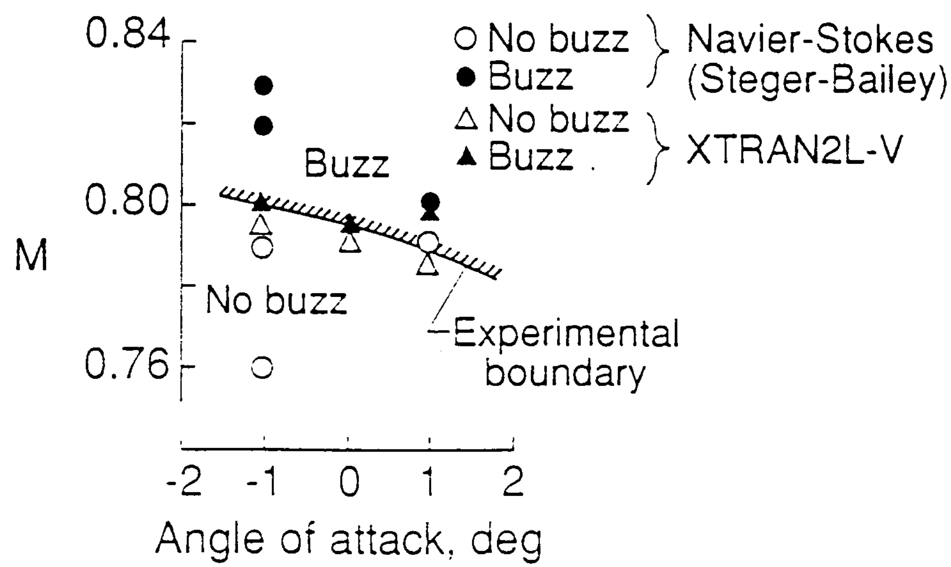


Figure 36 (b).

ANALYSIS OF NTF ARC-SECTOR/FIXED FAIRING INDICATES HUMP MODE FLUTTER

Woodrow Whitlow, Jr.
Unsteady Aerodynamics Branch

RTOP 505-63-21

Research Objective - The objectives of this research are to analyze vibrations of the NTF model support system and to determine methods for alleviating the vibrations.

Approach - The vibrations are analyzed with the CAP-TSD code that is used for aeroelastic analysis of complex configurations. The model support system, shown at the lower left of figure 37(b), consists of a movable arc sector and a fixed aerodynamic fairing. For this analysis, the arc sector/fixed fairing is represented as a flexible wing with six natural vibration modes that are obtained from a finite element analysis. Those modes do not include the effects of either the sting or models mounted on the sting. Initial disturbance velocities in one or more of the modes are used to excite the structure, and several cycles of the dominant flutter mode are calculated. Transients of the generalized coordinates are analyzed to determine if the aeroelastic response is stable or unstable. Aeroelastic responses are calculated using solid wind tunnel wall boundary conditions and using free-air (no walls) conditions.

Accomplishment Description - Aeroelastic responses of the arc sector/fixed fairing were calculated for a tunnel test section Mach number of 0.5. A "hump" mode instability of the third mode was calculated using wind tunnel wall and free-air boundary conditions. These results, at the lower right of the figure, show that removing the effects of the tunnel walls reduces the instability and increases the dynamic pressure at which it occurs. Contours of constant displacement for the third mode are shown at the upper right of the figure. The contours show that the unstable mode is one in which the fixed fairing behaves as an oscillating flap. It should be noted the the instability is of relatively small amplitude, and structural damping could stabilize the response.

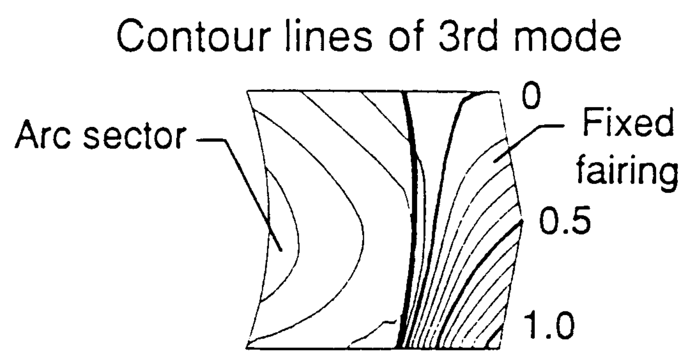
Significance - The results show that vibrations of the NTF model support system could be caused by an aeroelastic instability that is adversely affected by the close proximity of the tunnel side walls. The nature of the instability appears to be the fixed fairing oscillating as a trailing edge flap.

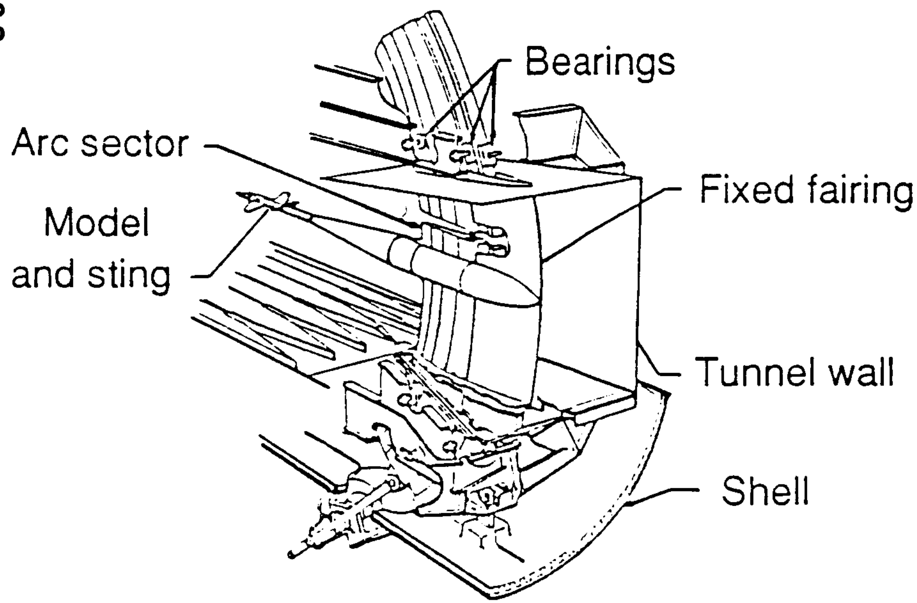
Future Plans - Future plans are to calculate the aeroelastic response at higher Mach numbers using the finite element vibration modes and measured modes that include sting and model effects. In addition, design changes that alleviate the vibrations will be proposed and analyzed.

Figure 37 (a).

ANALYSIS OF NTF ARC - SECTOR / FIXED FAIRING INDICATES HUMP MODE FLUTTER

- Models on sting mount experience vibrations
- Fixed fairing dynamics resemble aft flapping motion
- 3rd Mode stability adversely affected by tunnel walls
- Effects of sting and model masses under study





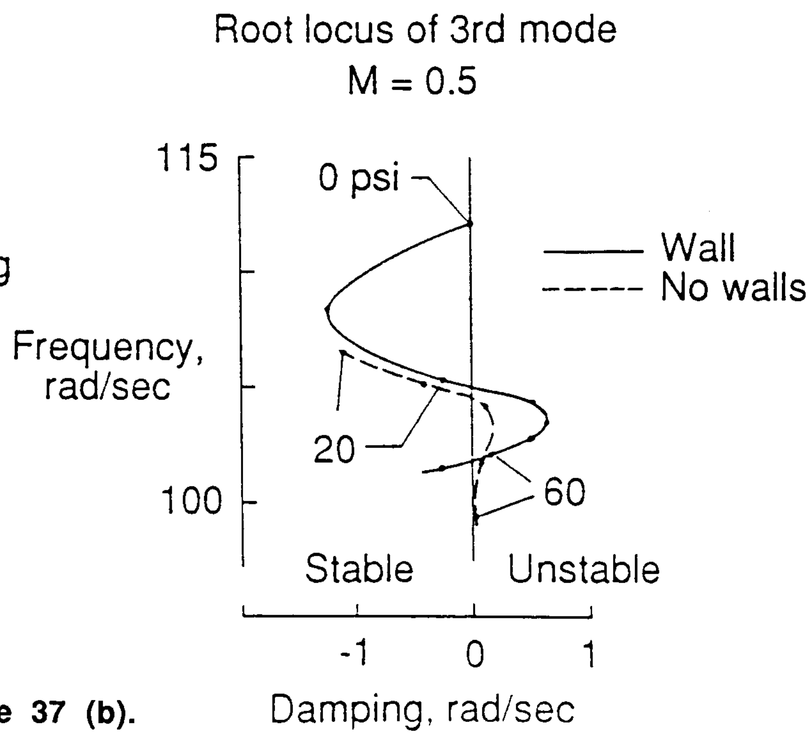


Figure 37 (b).

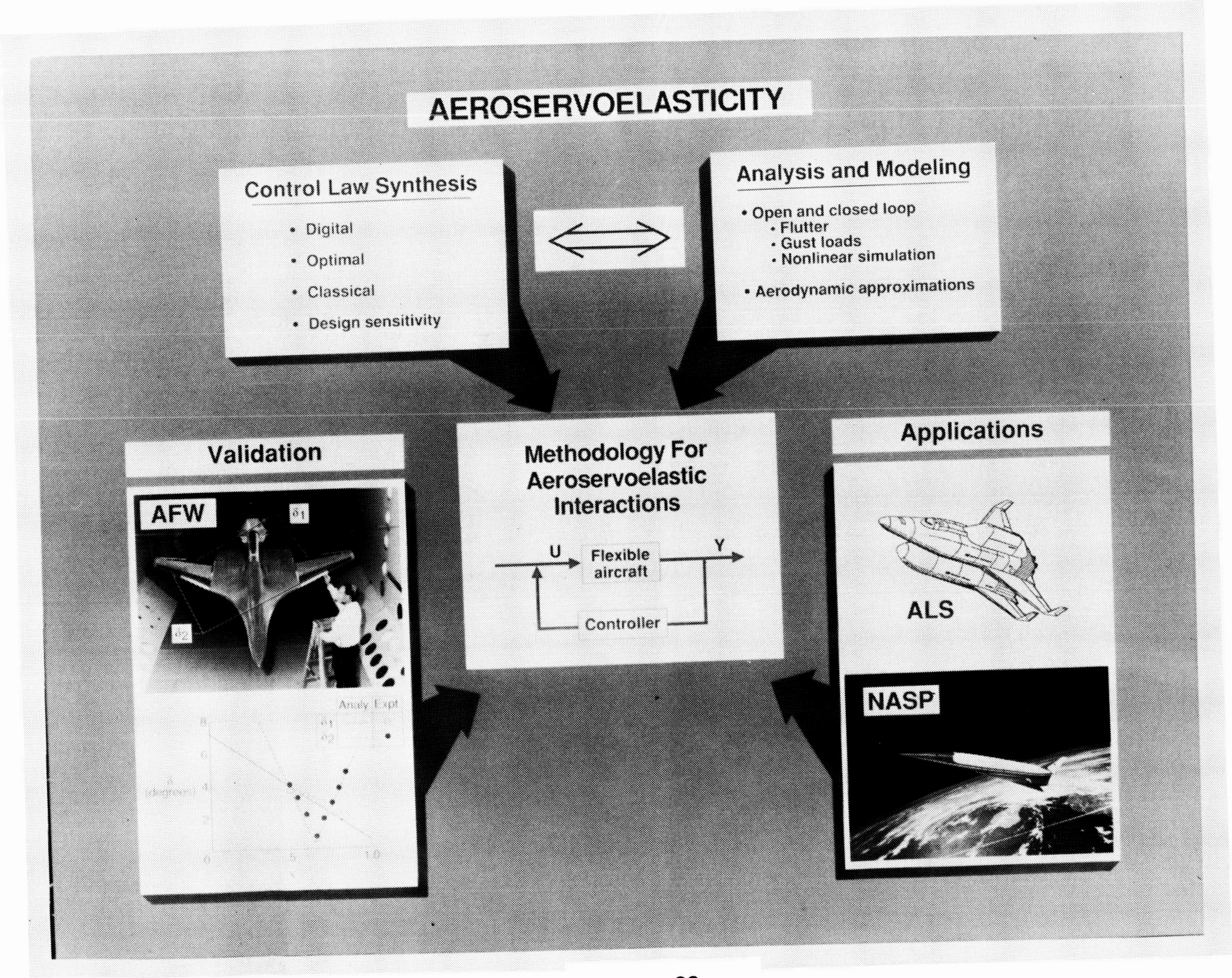

AEROSERVOELASTICITY
Control Law Synthesis
• Digital
• Optimal
• Classical
• Design sensitivity
Analysis and Modeling
• Open and closed loop
• Flutter
• Gust loads
• Nonlinear simulation
• Aerodynamic approximations
Validation
AFW
Methodology For Aeroservoelastic Interactions
U
Flexible aircraft
Y
Controller
Applications
ALS
NASP

Figure 38.

AEROSERVOELASTICITY
FIVE YEAR PLAN

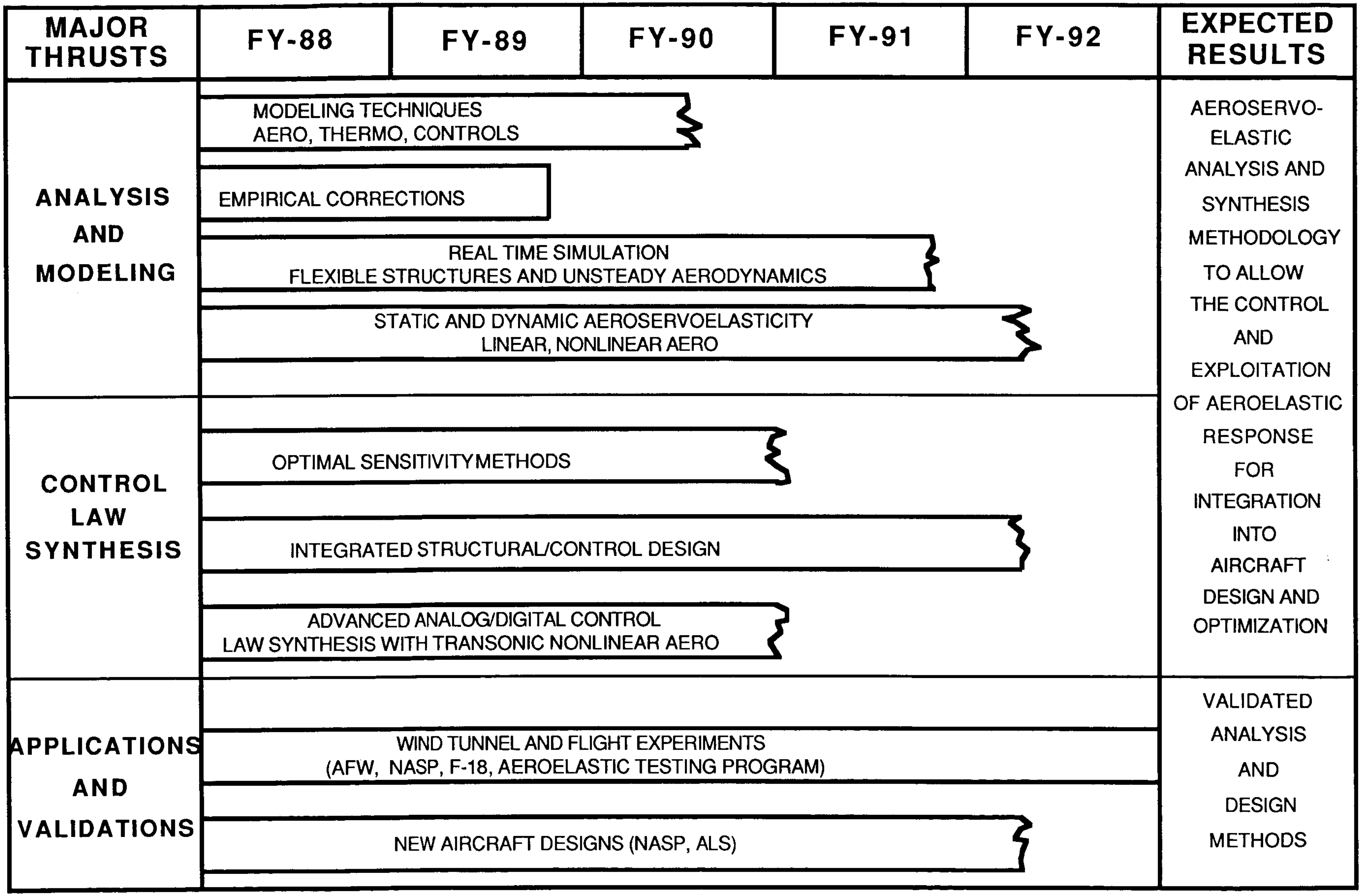


Figure 39.

MINIMUM-STATE APPROXIMATIONS OF UNSTEADY AERODYNAMICS PERMITS LARGE ORDER REDUCTION OF AEROSERVOELASTIC EQUATIONS

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Aeroservoelasticity Branch

RTOP 505-63-21

<u>Research Objective</u> - The objective of this research is to obtain accurate low-order aeroservoelastic state-space models by applying various constraining, weighting, and lag coefficient selection techniques to Minimum-State unsteady generalized aerodynamic force approximations.

<u>Approach</u> - Time-domain aeroservoelastic modeling requires rational approximations of the unsteady aerodynamic force coefficients in the Laplace domain. Various approximation procedures are available. Among these, the Minimum-State Method, which approxi-mates all the force coefficients simultaneously, yields the lowest sized model. Effective application of this method, however, requires careful study of the aerodynamic data for deleting modes and frequency ranges that do not affect the aeroservoelastic characteristics of interest. A systematic method for selecting denominator (lag) coefficients, choosing constraints, and weighting the various data terms according to their importance in subsequent analyses enables more effective, easier to apply, approximation procedures.

<u>Accomplishment Description</u> - High-accuracy aeroservoelastic models for subsequent flutter-suppression control law design and near real-time simulation have been generated with one-tenth the number of aerodynamic states as compared to more common methods (See figure 40(b)). A physical weighting algorithm, which weights the individual data terms according to the affect of their errors on the aeroservoelastic characteristics has been developed. These weights also indicate low-interest, frequency ranges. The Minimum-State approximation code has been modified to accept the weighted data and to allow more efficient application of some constraint options. These developments are implemented in the MIST (MInimum STate) computer program which interfaces with NASA's ISAC (Interaction of Structures, Aerodynamics, and Controls) system of programs. A simple method, based upon mode contribution to flutter, for selecting denominator coefficients which provide a good set of lag terms even without optimization has also been demonstrated.

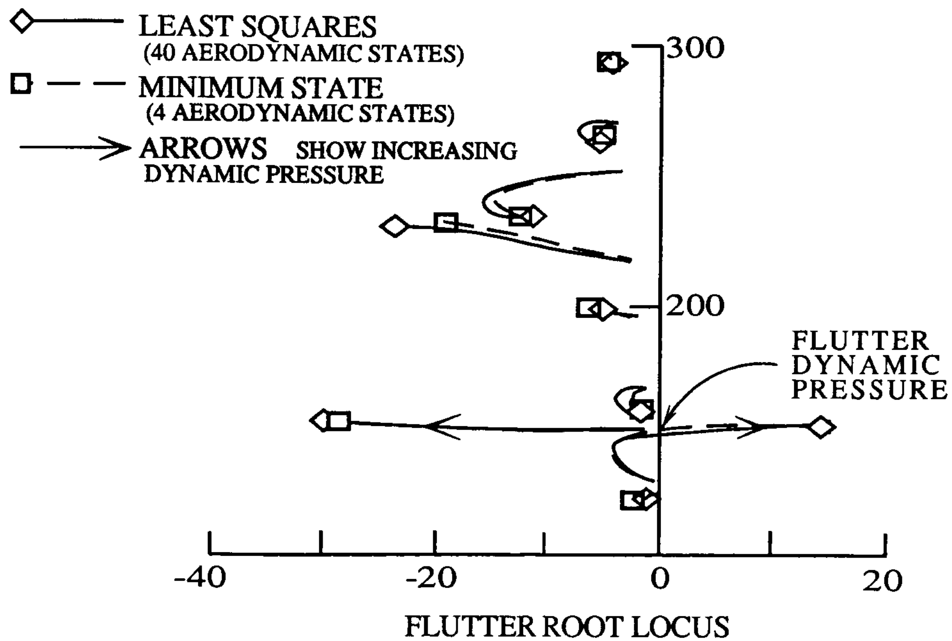
<u>Significance</u> - By reducing the number of aerodynamic states by an order-of-magnitude, these various constraining, weighting, and lag selection techniques provide an effective, systematic approach to generating aerodynamic approximations. Using these techniques, the total size of a typical time-domain aeroservoelastic model can be efficiently reduced by 50 percent. In addition to significant computer time savings for control design and analysis, lower-size models yield more realizable optimal control laws and facilitate near real-time simulations which include unsteady aerodynamic affects.

<u>Future Plans</u> - Results will be presented at the 1989 AIAA Structures, Structural Dynamics and Materials Conference to be held at Mobile, Alabama in April 1989. Analyses will be performed to provide low order models of the Active Flexible Wing wind-tunnel model for simulation and design. After the wind-tunnel tests, results using these models will be compared to the experimental data for validation.

Figure 40 (a).

MINIMUM-STATE APPROXIMATIONS OF UNSTEADY AERODYNAMICS PERMITS LARGE ORDER REDUCTION OF AEROSERVOELASTIC EQUATIONS

- Factor-of-ten reduction in equations describing unsteady aerodynamics
- Greater than factor-of-two reduction in total system of equations
- No significant change in aeroservoelastic characteristics
- Necessary step for near real-time simulation



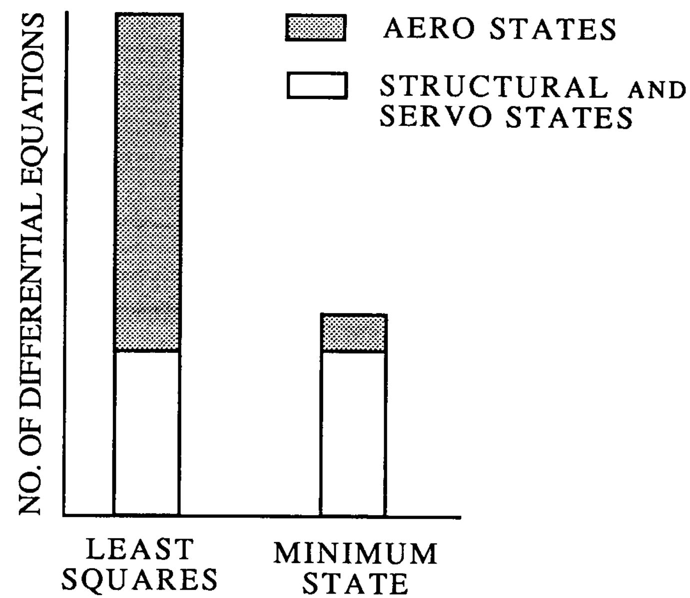


Figure 40 (b).

OVERLAP BETWEEN SDG AND PSD GUST ANALYSIS METHODS ESTABLISHED

Boyd Perry, III & Anthony S. Pototzky (PRC) and Jessica A. Woods (PRC)
Aeroservoelasticity Branch

RTOP 505-63-21

Research Objective - The objective of the present work is to investigate the so-called "overlap" between two gust response analysis methods: Statistical Discrete Gust (SDG) method and Power Spectral Density (PSD) method.

Approach - The developer of the SDG method, J. G. Jones of the Royal Aerospace Establishment, claims that, under certain conditions (indicated in the bottom left-hand corner of the figure), the SDG and PSD methods "overlap" and produce essentially the same numerical results, meaning that the responses from one method may be predicted if the responses from the other method are known. The approach taken is to apply the SDG and PSD methods to several aircraft configurations and compare the corresponding responses from each method to see if a "10.4 factor" is obtained.

Accomplishment Description - Figure 41(b) contains typical results from a fully-flexible analysis of an experimental drone aircraft at an analysis condition of 15,000 feet altitude and Mach number 0.7. Plotted in the lower right-hand corner are ratios of SDG-responses-to-PSD-responses for five typical airplane responses. All ratios (including many others not shown in the figure) fall within +6% and -25% of 10.4, indicating an "overlap" between the two methods.

Significance - Federal Aviation Regulations (specifically, FAR 25.305(d)) require that, unless a more rational method is used, an airplane manufacturer must use the PSD method to determine the dynamic response of its airplanes to atmospheric turbulence. In recent years, many foreign civil airworthiness authorities, many foreign transport manufacturers, and some U.S. transport manufacturers have looked to the Federal Aviation Administration (FAA) to encourage research into alternate means of compliance with FAR 25.305(d). The SDG method is a candidate alternate means of compliance and it has the advantage (over the PSD method) of yielding time-correlated gust loads. Before the FAA will approve a new method as a means of compliance it must satisfy itself that the new method is valid and that the claims made about the method are true. In an effort to gain this satisfaction the FAA requested NASA's assistance in investigating the claim made by J. G. Jones regarding the SDG and PSD methods. The present work has provided the FAA with the desired information.

Future Plans - Because the results of the NASA investigation indicate some deviation from the 10.4 factor, the FAA has requested further NASA investigation into a refinement of the SDG method which the developer claims will yield ratios closer to 10.4. NASA is performing this work now and a final report will be prepared and presented to the FAA in April, 1989 and will also be presented at the 1989 AIAA Structures, Structural Dynamics and Materials Conference.

Figure 41 (a).

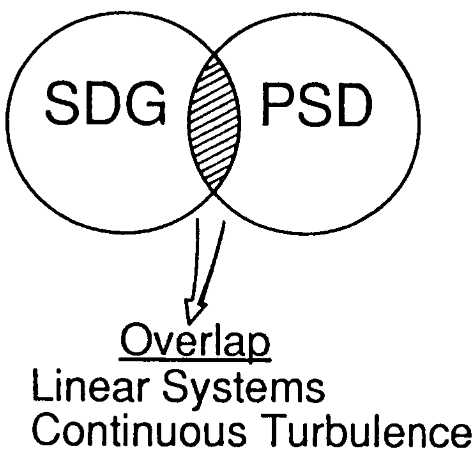
C-2

Overlap Between SDG and PSD Gust Analysis Methods Established

- FAA requested assistance from NASA to validate RAE claim that the Statistical Discrete Gust (SDG) and Power Spectral Density (PSD) methods produce essentially the same numerical results . . .

(SDG Response) = 10.4 (PSD Response)

- Advantages of SDG method are --
  - yields time-correlated gust loads
  - provides an alternate design approach



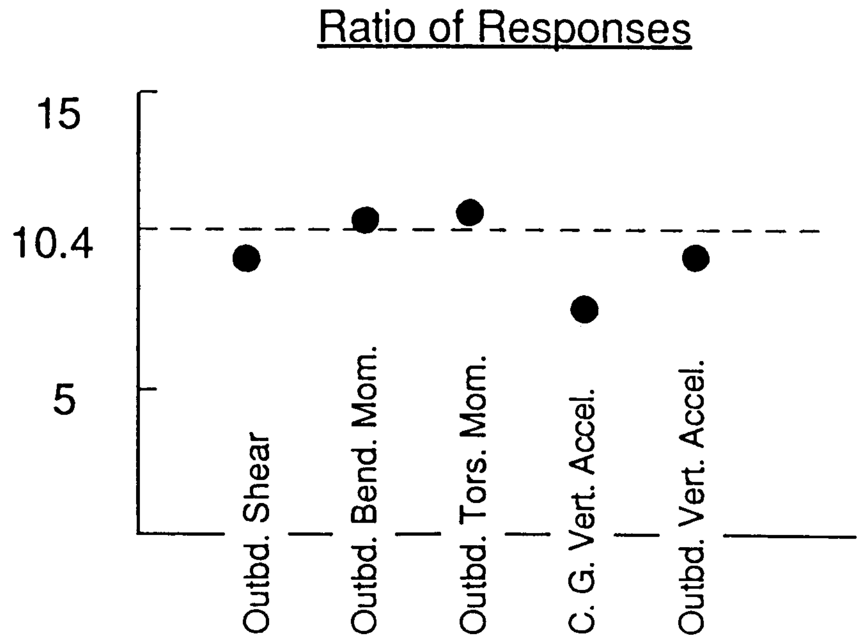


Figure 41 (b).

MATCHED FILTER AND RANDOM PROCESS THEORIES PROVIDE EFFICIENT OPTIONS FOR DETERMINING MAXIMIZED TIME-CORRELATED GUST LOADS

Anthony S. Pototzky and Dr. Thomas A. Zeiler & Boyd Perry, III
PRC Systems Services & Aeroservoelasticity Branch

RTOP 505-63-21

Research Objective - The objective is to investigate alternate gust response analysis methods that will provide maximized time-correlated gust responses of unsteady aeroelastic systems.

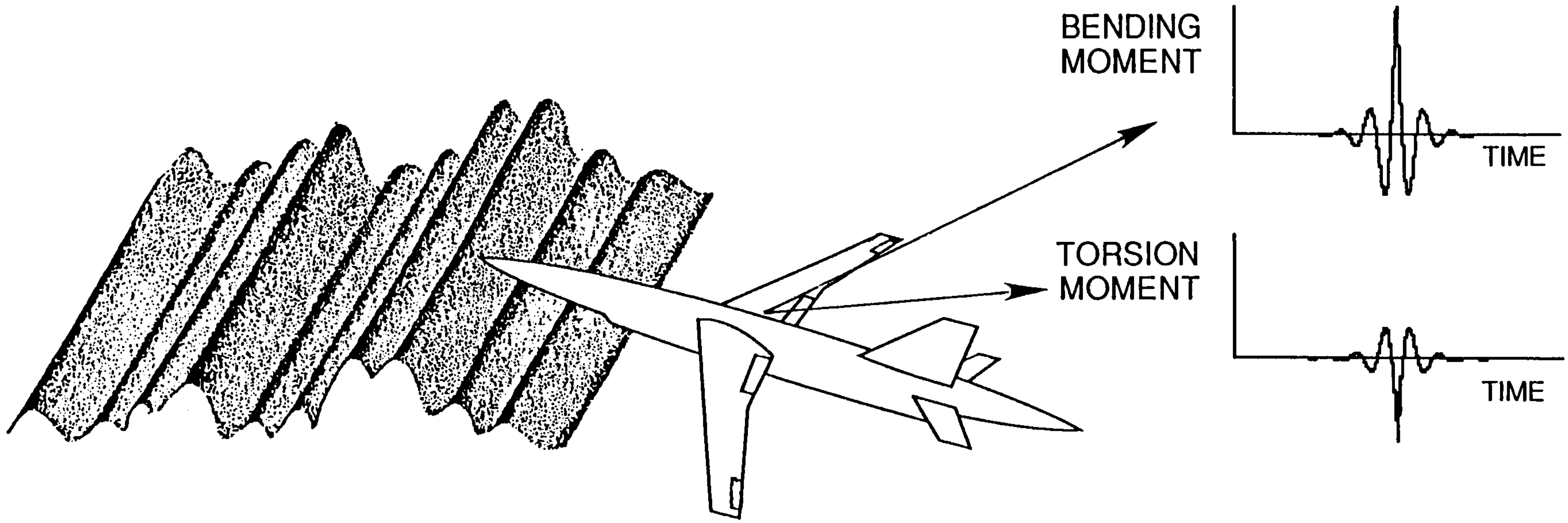
Approach - Unconventional interpretations of two existing theories are explored for computing maximized time-correlated responses that can be used for balanced-load calculations. The first is based on Matched Filter Theory, which originally was applied in radar signal detection; the second is based on a new interpretation of the results from Random Process Theory. In the case of Matched Filter Theory, the critical gust profile, the maximized time response, and other time-correlated system responses are generated. The figure gives a graphical representation of a typical application of Matched Filter Theory. In the case of Random Process Theory, the auto- and cross-correlation functions are obtained by inverse Fourier transforming the auto- and cross-power spectral density functions. The auto- and cross-correlation functions are interpreted as time-correlated system responses. Both approaches are direct and yield theoretically identical results; however, the choice of which to use depends on the intended application.

Significance - Aircraft manufactures have encouraged the FAA to investigate alternate means of compliance with Federal Aviation Regulation 25.305(d), which relates to aircraft gust loading. Matched Filter Theory and a new interpretation of the correlation functions from Random Process Theory provide new analysis options for computing time-correlated dynamic loads in aircraft. Both approaches are computationally fast. In addition to computing time-correlated gust loads, the two theories may be employed to compute time-correlated taxi, landing or maneuver loads.

Future Plans - Results from this study will be presented at the AIAA Structures, Structural Dynamics and Materials Conference to be held at Mobile, Alabama in April 1989. Further investigations to define the applicability of Matched Filter Theory for generating time-correlated gust load responses for systems with embedded nonlinearities is also planned.

Figure 42 (a).

MATCHED FILTER AND RANDOM PROCESS THEORIES PROVIDE EFFICIENT OPTIONS FOR DETERMINING MAXIMIZED TIME-CORRELATED GUST LOADS



- UNCONVENTIONAL USE OF EXISTING THEORIES
- COMPUTATIONALLY FAST AND DIRECT
- THEORETICALLY IDENTICAL RESULTS
- APPLICABLE TO WIDE RANGE OF DYNAMIC RESPONSE PROBLEMS
- NEW ANALYSIS OPTIONS FOR THE STRUCTURAL DESIGN ENGINEER

Figure 42 (b).

CORRECTION FACTOR METHODOLOGIES DEVELOPED TO IMPROVE PREDICTIONS OF UNSTEADY AERODYNAMICS

Carol D. Wieseman
Aeroservoelasticity Branch

RTOP 505-63-21

Research Objective - The objective of the present work is to develop, implement, and validate methodology for using steady experimental force or pressure data to obtain corrections for improving the prediction of both steady and unsteady aerodynamic forces and pressures. The data used to calculate the correction factors involves steady pressures and forces from wind tunnel tests, flight experiments or calculations using a higher level source code, such as a CFD code.

Approach - Three different approaches to calculate correction factors were developed and implemented. Correction factors were calculated to match steady pressure distributions at the analytical locations, by matching section aerodynamic properties or by matching aircraft total forces and moments. The approach for matching total forces uses optimization procedures. Correction factors can be applied to either box downwashes or box pressures.

Accomplishment Description - The pressure matching methodology has been fully implemented and validated using experimental pressure data from a rectangular supercritical wing tested in the NASA Langley Transonic Dynamics Tunnel. The upper left of figure 43(b) shows a planform of the wing, the experimental pressure measurement stations and the analytical box pattern for the Doublet Lattice subsonic lifting surface theory. Steady experimental pressures were interpolated using one-dimensional splines to the analytical locations corresponding to the quarter-chord and midspan location of each of the boxes. The data was then interpolated as a function of angle of attack to obtain the derivatives of the pressure coefficients with angle-of-attack. Correction factors were calculated to match pressures at Mach numbers ranging from 0.266 to 0.8. Comparisons of experimental and analytical steady pressure distributions at a Mach number of 0.4 at a midspan location are shown in the upper right of the figure. For improving unsteady aerodynamic predictions, downwash correction factors were obtained and applied to the analytical unsteady pressures for the values of reduced frequency at which unsteady experimental data were available so that direct comparisons between experiment and analyses could be made. The bottom of the figure shows comparisons of the unsteady magnitudes and phases of the experimental pressures with corrected analytical pressures at a reduced frequency of 0.309.

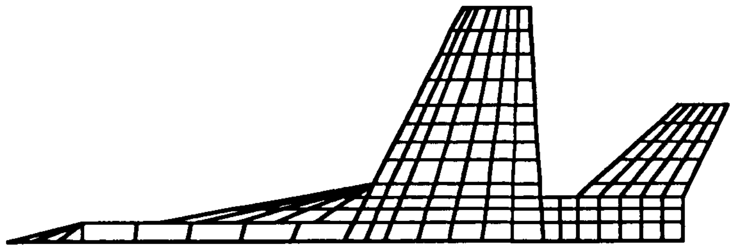
Significance - In performing aeroservoelastic analyses, it is important to have accurate predictions of the unsteady aerodynamic forces. This methodology will help improve the accuracy of the Doublet Lattice Method which is currently the state-of-the-art in calculating unsteady subsonic oscillatory aerodynamics.

Future Plans - The validation of the section property matching methodology will be completed. Validation of the total force method will be conducted using the F-18 aircraft as a test case.

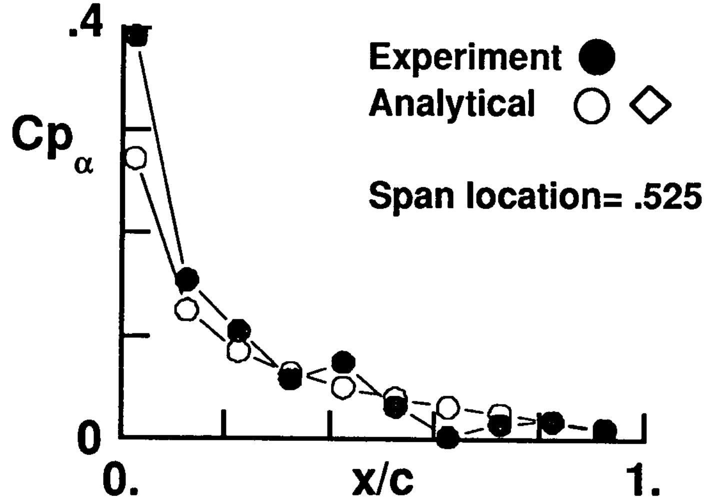
Figure 43 (a).

CORRECTION FACTOR METHODOLOGIES DEVELOPED TO IMPROVE PREDICTIONS OF UNSTEADY AERODYNAMICS

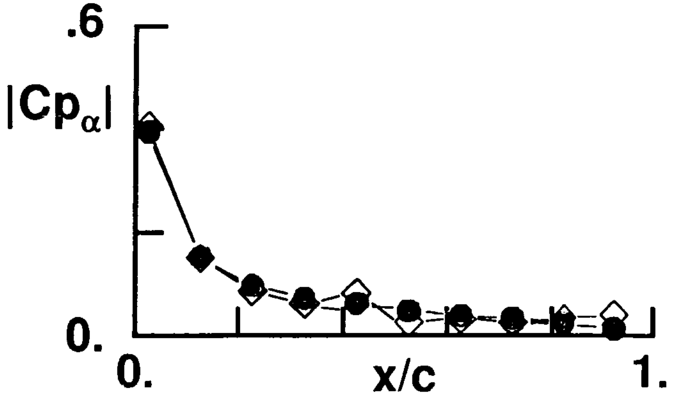
Typical Aerodynamic Box Layout



Comparison of Steady Experimental and Analytical Pressure Data, M=.4



Comparison of Experimental and Corrected Analytical Unsteady Pressure Coefficients, M=.4, k=.309, Span Location =.525



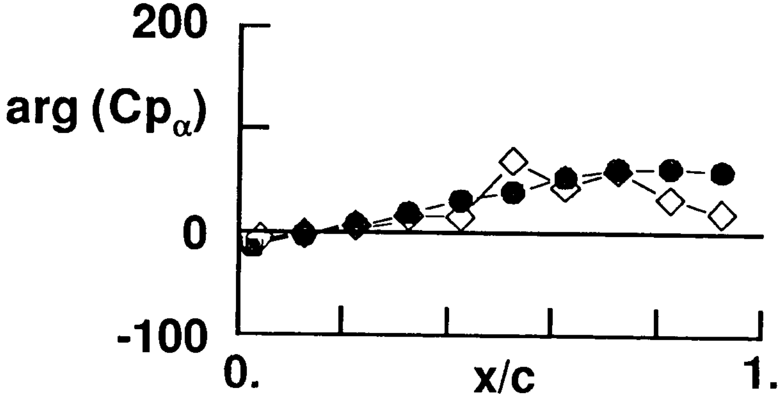


Figure 43 (b).

DESIGN OF DIGITAL MULTI-INPUT/MULTI-OUTPUT FSS OBTAINED FOR AFW MODEL

Dr. Vivekananda Mukhopadhyay (PRC)
Aeroservoelasticity Branch

RTOP 505-63-21

Research Objective - The objective is to develop preliminary digital control laws for Active Flexible Wing (AFW) wind-tunnel model for symmetric and antisymmetric flutter suppression at Mach 0.9 using the theoretical state-space system of equations.

Approach - The multi-input/multi-output (MIMO) candidate flutter suppression (FSS) system uses the leading edge outboard (LEO) and the trailing edge outboard (TEO) control surfaces as input and collocated accelerometers at LEO and TEO actuators as the sensor output shown on left of figure 44(b). The FSS system block diagram is shown in the right side of the figure. The design plant model included the first ten flexible modes, the actuator dynamics, a Dryden gust model and the antialiasing filters. The FSS control law was obtained by first developing and analyzing the full-order analog optimal control law. This control law was then reduced in order by block-diagonalization and subsequent truncation. The reduced order control law was optimized in the continuous time domain, analyzed for robustness and performance, and then transformed into the discrete-time domain and evaluated.

Accomplishment Description - Based on the preliminary theoretical state-space model (obtained from ISAC), full order LQG analog control laws and reduced order analog and digital control laws were developed for flutter suppression. The analog and digital simulation indicated that an 8th order control law could increase the closed loop symmetric flutter dynamic pressure at Mach 0.9 by nearly 100 percent. Realistic gain and phase margins at 300 psf were demonstrated in the continuous domain. No gain scheduling was required with dynamic pressure to maintain stability. Guaranteed MIMO stability margins of ± 4.5 dB and ± 25 degrees (corresponding to minimum singular value of 0.4) at the plant input and output were achieved.

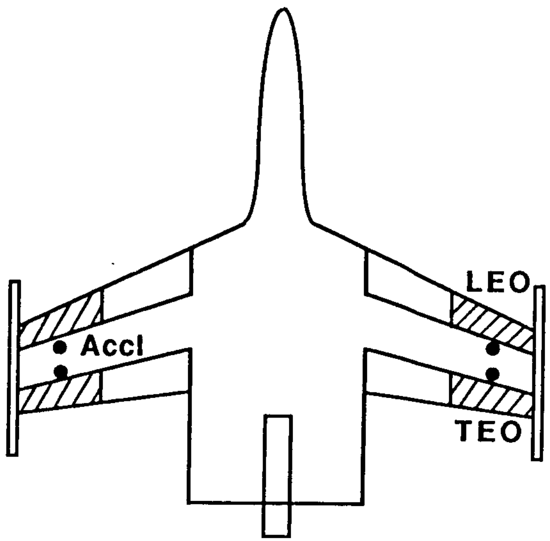
Significance - The analysis indicated that fairly low-order FSS control laws can be designed for this complex large-order system for symmetric flutter suppression. The effects of a one-cycle computational delay and the antialiasing filters were included in the design. However, the stability margins usually deteriorate when the system is discretized and the control laws will need to be reoptimized in the discrete domain.

Future Plans - The control laws are to be redesigned using the latest updated model data for symmetric and antisymmetric flutter suppression in the analog and discrete domain. Load alleviation control law development is also being planned.

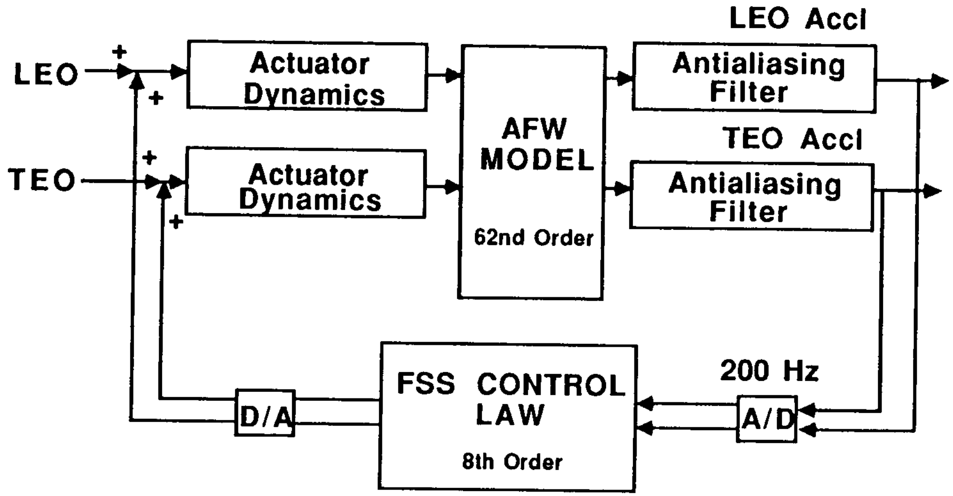
Figure 44 (a).

DESIGN OF DIGITAL MULTI-INPUT/MULTI-OUTPUT FSS OBTAINED FOR AFW MODEL

CANDIDATE CONTROLS/SENSORS



BLOCK DIAGRAM



ACCOMPLISHMENTS

- o Flutter Dynamic Pressure at M = 0.9 Increased 100%
- o Realistic Gain and Phase Margins at 44% Increase in Flutter Dynamic Pressure
- o No Gain Scheduling Required with Dynamic Pressure
- o Guaranteed MIMO Stability Margins

Figure 44 (b).

STABILITY ROBUSTNESS IMPROVED USING SINGULAR VALUE CONSTRAINTS

Dr. Vivekananda Mukhopadhyay (PRC)
Aeroservoelasticity Branch

RTOP 505-63-21

Research Objective - The objective of this effort is to demonstrate the application of an optimization technique for improving the stability robustness of a multi-input/multi-output discrete feedback control system using singular value constraints.

Approach - The constrained optimization technique was used to search for the digital control law design variables which would satisfy the design load constraints and the singular value cumulative constraints while minimizing a prescribed cost function of the design responses and control input. The singular values of the discrete domain transfer function matrices were computed over a frequency range. The cumulative constraints were defined on the lower bound of the minimum singular value of the return-difference matrix at the plant input and output. The procedure was applied to a digital gust load alleviation problem of a remotely-piloted drone aircraft shown in figure 45(b). The gust load alleviation (GLA) system uses elevator and aileron pairs as control input. Wing and fuselage mounted accelerometers are used as sensor output. The state-space model represents symmetric flight through a gust defined by a Dryden spectrum. Second order GLA control laws were designed from the full order LQG design.

Accomplishment Description - The gust load alleviation digital control law synthesis steps are shown at the top of figure 45(c). The comparison of normalized RMS loads and responses with respect to the open-loop system are shown in the middle of figure 45(c). The wing-root bending moment (WRBM) and shear (WRS) were reduced by 50% without increasing the wing-outboard bending moment (WOBM) and torsion (WOT). The stability robustness was improved by increasing the minimum singular values at the plant input and output to 0.6 (-4.6 dB) as shown at the bottom of figure 45(c). This corresponds to a simultaneous gain margin of ±8 dB and a phase margin of ±35 degrees in all the loops at the plant input and output.

Significance - In general, when a control law of a given order is discretized keeping the order constant, the discrete domain stability robustness usually deteriorates. This procedure presents a direct method of improving the robustness in the discrete domain.

Future Plans - The procedure will be used for the Active Flexible Wing (AFW) flutter suppression control law synthesis and optimization tasks.

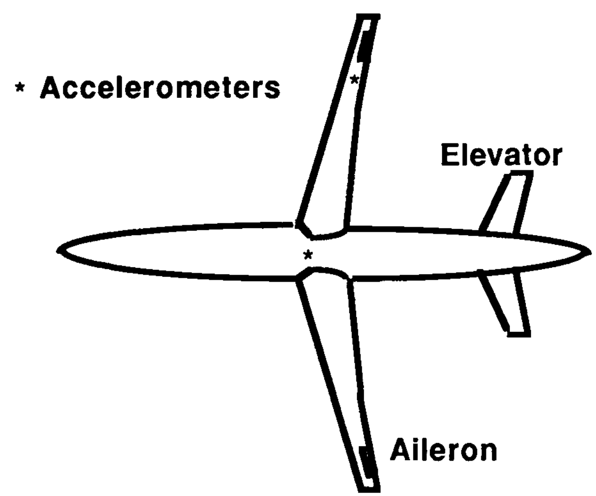
Figure 45 (a).

SYNTHESIS METHODOLOGY DEVELOPED FOR DIGITAL ROBUST CONTROL LAWS USING CONSTRAINED OPTIMIZATION

DESIGN PROCESS

1. LQG Full Order Analog
2. Analog Reduced Order
3. Discretize (Law I)
4. Optimization (Law II)
5. Apply Constraints
   a) RMS Loads (Law III)
   b) Singular Values (Law IV)

EXAMPLE OF GUST LOAD ALLEVIATION SYSTEM DESIGN



Physical Quantities	Design Objective
Wing Root Bending Moment	50% reduction
Wing Root Shear	50% reduction
Wing Outboard Bending Moment	No increase
Wing Outboard Torsion	No increase
Elevator Deflection	Within max limit
Elevator Rate	Within max limit
Aileron Deflection	Within max limit
Aileron Rate	Within max limit

Figure 45 (b).

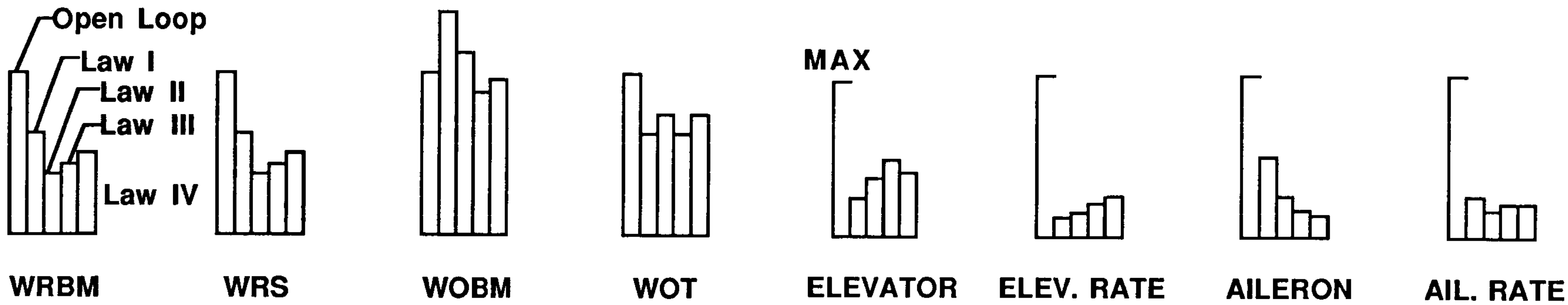
RM-88/11

STABILITY ROBUSTNESS IMPROVED USING SINGULAR VALUE CONSTRAINTS

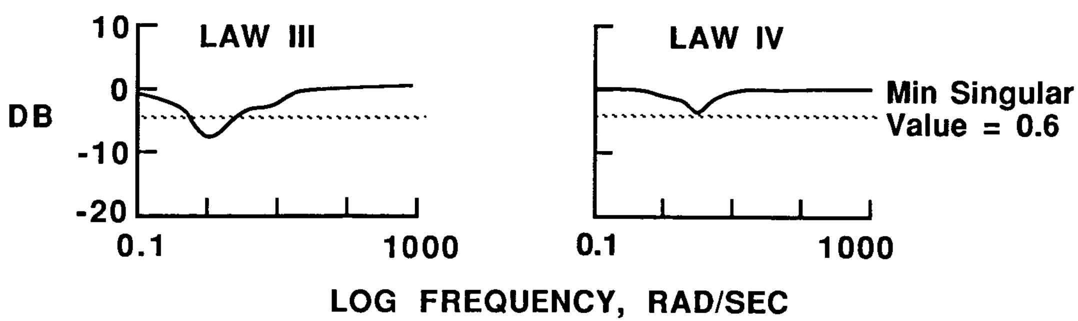
DIGITAL CONTROL LAWS

Law I	After Discretizing Reduced Order Analog System
Law II	After Optimization Without Constraints
Law III	After Applying RMS Load Constraints
Law IV	After Applying Singular Value Constraints

COMPARISON OF NORMALIZED RMS RESPONSES



OPTIMIZATION WITH SINGULAR VALUES



Gain Margins	± 8 dB
Phase Margins	± 35 degrees

Figure 45 (c).

ANALYTICAL SENSITIVITIES IMPROVE INTEGRATED STRUCTURE/CONTROL LAW DESIGN METHODOLOGY

Michael G. Gilbert
Aeroservoelasticity Branch

RTOP 505-63-21

Research Objective - Analytical sensitivity or gradient expressions can be used in numerical optimization problems to improve accuracy and reduce computational costs when compared to equivalent finite difference methods. The purpose of this research was to develop analytical expressions for the sensitivity of optimized Linear Quadratic Gaussian (LQG) control laws for use in a multilevel, integrated structure/control law design methodology.

Approach - Analytical expressions for the change or sensitivity of the solution to an optimization problem due to changes in physical parameters fixed during the optimization can be derived from the necessary conditions of optimality for the problem. This is accomplished by differentiating the necessary conditions with respect to the fixed parameter and evaluating the resulting expression at the point were the necessary conditions are satisfied.

Accomplishment Description - Analytical expressions for the sensitivity of optimized LQG control laws have been derived using the above approach. The resulting sensitivity information is then used in conjunction with other existing analytical expressions for the sensitivities of linear system eigenvalues, covariance responses, time responses, and frequency responses to predict the change in optimized system performance due to a change in a system parameter. The approach was validated using a 25^{th} order state-space mathematical model of an aeroservoelastic aircraft with an unstable short period mode eigenvalue. The control law was designed to stabilize the short period. The sensitivity of the optimized control law and aircraft responses to various inputs, including discrete and random gust environments, was computed for several aircraft structural and configuration parameters. Sensitivity results computed using the above approach were compared with more traditional control system sensitivity results which measure the change in performance due to unmodeled or undesired parameter changes for a given control law. Shown on the left in figure 46(b) is the error in predicting a change in the mean square pitch response due to a change in the wing bending stiffness of the aircraft, indicating that the current method provides more accurate sensitivity information for design purposes than the standard control system sensitivity method. Computation cost comparisons in terms of CPU time necessary to compute the sensitivity information using the analytical expressions and by using finite difference methods is shown on the right for a single parameter. Evaluation of the analytical expressions is cheaper than the finite difference method for a single parameter, with significant additional advantages for the multiple parameter case possible due to storage of partial problem solutions.

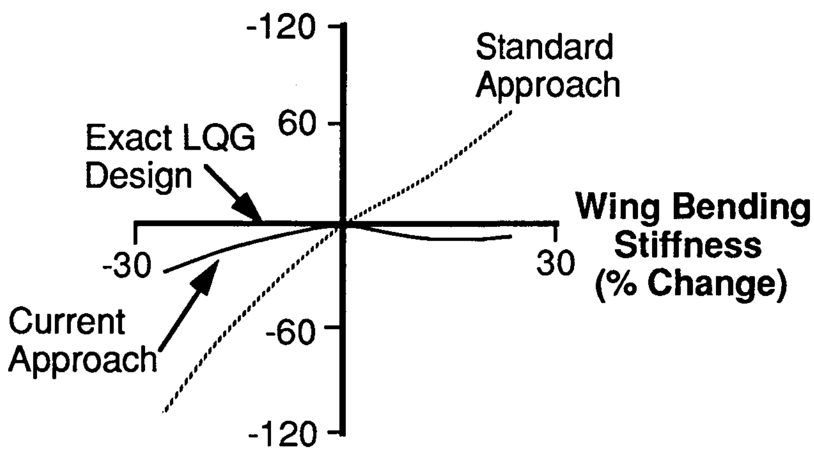
Significance - The analytical sensitivity expressions provide more accuracy at lower cost, which will improve multilevel integrated structure/control law design algorithms by reducing the number of iterations required for convergence.

Future Plans - The LQG control law design sensitivity analysis methodology will be combined with a multilevel structural optimization method to complete the development of a realistic integrated structure/control law design capability.

Figure 46 (a).

ANALYTICAL SENSITIVITIES IMPROVE INTEGRATED STRUCTURE/CONTROL LAW METHODOLOGY

- o Developed Analytical Expressions for the Sensitivity of Optimal LQG Control Laws to Physical Parameters
- o Used to Predict Changes in Controlled System Response
- o Methodology Validated on 25th Order Aeroservoelastic Model of ARW-II (Control Law Stabilizes Short Period Root)
- o More Accurate, Less Costly than Other Sensitivity Approaches



CPU Time Comparisons (Sec.)

Control Design	100.7	
Control Design with Sensitivity	133.6	Single Parameter
Control Design with Finite Difference Sensitivity	196.4	Single Parameter

Figure 46 (b).

AFW FLUTTER BOUNDARY LOWERED BY ADDITION OF "TIP MISSILE"

Boyd Perry, III
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RTOP 505-63-21

Research Objective - The Active Flexible Wing (AFW) Program is a joint NASA-LaRC/Rockwell effort with the overall objective of demonstrating, through the application of active controls technology, aeroelastic control of a sophisticated aeroelastic wind tunnel model. One of the areas of aeroelastic control is flutter suppression, and the goal is to demonstrate a significant (on the order of 100%) increase in flutter dynamic pressure. A situation existed which, had it not been corrected, would have prevented this goal from being achieved: the basic AFW flutter boundary was beyond the upper operating limits of the wind tunnel in which it will be tested. The objective of the present work was to modify the AFW wind tunnel model, thereby lowering its flutter boundary so that a significant increase in flutter dynamic pressure may be demonstrated in the wind tunnel by using active controls.

Approach - Based on joint NASA-LaRC / Rockwell discussions, the decision was made to employ the decoupler pylon principle and to build a "tip missile" which would do two things: (1) lower the flutter boundary as desired; and (2) act as a flutter stopper in the event that flutter or other aeroservoelastic instabilities are unexpectedly encountered during the wind tunnel testing.

Accomplishment Description - Rockwell has designed, built, and installed "tip missiles" on the wing tips of the AFW wind tunnel model. The sketch in the lower right-hand corner of the figure illustrates the arrangement and relative size of a "tip missile" installed on the wing. Analysis indicates that the "tip missiles" have the desired effects of (1) lowering the flutter dynamic pressure of the basic wind tunnel model by over 300 psf when they are in the coupled configuration; and (2) raising the lowered flutter dynamic pressure by over 200 psf when they are in the decoupled configuration. These effects are illustrated by the flutter boundary sketch in the lower left-hand corner of figure 47(b).

Significance - The significance of this work is that it allows an important research objective of the AFW Program to be met and it provides a valuable benefit in the form of additional safety for the wind tunnel model.

Future Plans - Wind tunnel tests are scheduled to begin in August 1989. A status report of the project will be presented at the AIAA Structures, Structural Dynamics and Materials Conference to be held at Mobile, Alabama in April 1989.

Figure 47 (a)

AFW FLUTTER BOUNDARY LOWERED BY ADDITION OF "TIP MISSILE"

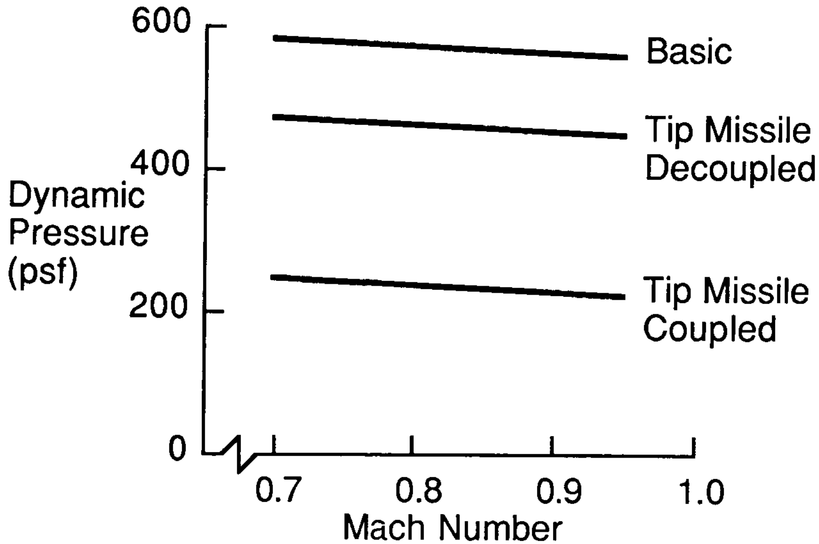
Goal: Demonstrate significant increase in flutter dynamic pressure using active controls

Problem: Basic AFW flutter boundary is beyond the TDT tunnel limits

Solution: Add "tip missile" to lower flutter boundary

Benefit: "Tip missile" designed to also be a flutter stopper if we get into trouble

FLUTTER BOUNDARIES



AFW WING

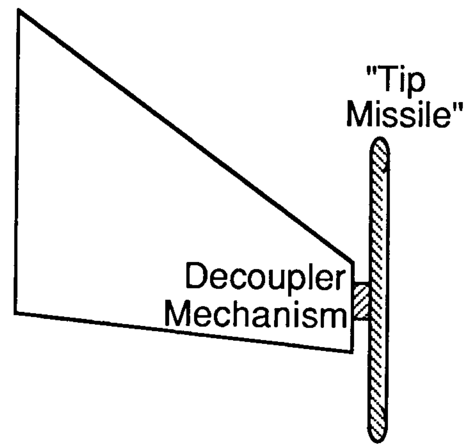


Figure 47 (b).

AFW CONTROL SYSTEM HARDWARE LAYOUT

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RTOP 505-63-21

Research Objective - The objective is to design and implement a real-time, multi-input/multi-output digital controller for the Active Flexible Wing (AFW) wind tunnel model. The hardware layout for the digital controller, including the NASA/Rockwell interface box, is shown in figure 48(b).

Approach - The AFW digital control system had to meet certain design specifications. The digital control system needed to be programmed in a high-level language so code could be developed and modified easily and quickly. The system had to handle at least 64 analog and 16 discrete inputs from the model, 16 analog and 8 discrete outputs to the model, and 16 discrete inputs and outputs from and to the user control panel. Furthermore, it had to sample data and execute the flutter suppression and rolling maneuver load alleviation systems at least 200 times per sec.

Accomplishment Description - In order to accomplish the goals set for the AFW digital controller, several additional processors were attached to the system via the VME bus: a digital signal processor (DSP), an array processor, and four Data Translation boards. The DSP provides the scheduling of the control laws and management of all signal processing. The array processor provides floating point arithmetic. The four Data Translation boards provide the analog-to-digital (ADC) and digital-to-analog (DAC) conversions needed between the model and the controller. As bus master, the DSP sends the digital control commands for the actuators to the DAC's, sends commands to the array processor which implements the desired control law, and checks the user control panel switches, sets lights, and checks for faults. The NASA/Rockwell interface box provides antialiasing and low-pass filters, and electrical isolation between the model and the digital controller.

Significance - The digital control system is VME based and is programmed in a high-level language. A VME-based system allows for hardware changes to be made with relative ease. Programming the digital control system in a high-level language allows any software changes to be made quickly. The digital control system can be modified easily and quickly for use in future projects.

Future Plans - The AFW digital controller will begin near-real time simulation in the Spring of 1989; the AFW wind tunnel model will undergo testing in the 16-foot Transonic Dynamics Tunnel in August 1989 at which time digital active flutter suppression and rolling maneuver load alleviation systems will be evaluated.

Figure 48 (a).

AFW Control System Hardware Layout

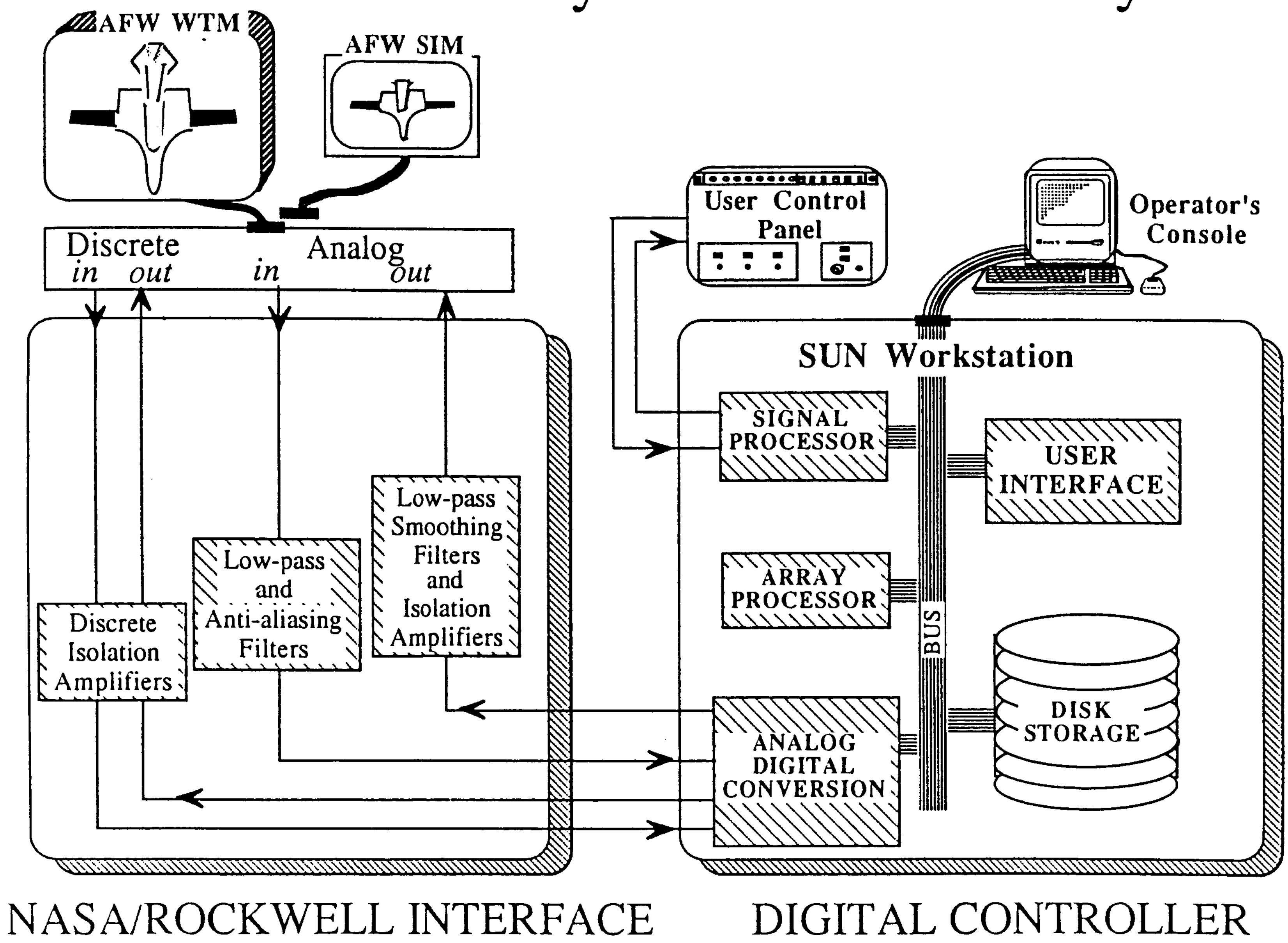

Figure 48 (b).

ANALYSIS ESTABLISHES GUIDELINES FOR AVOIDING AEROELASTIC INSTABILITIES OF X-WING AIRCRAFT

Jessica A. Woods (PRC) and Michael G. Gilbert
Aeroservoelasticity Branch

RTOP 505-63-21

Research Objective - The objective of this research was to determine the aeroelastic stability of a stopped-rotor X-Wing aircraft configuration with design parameter variations.

Approach - A symmetric aeroelastic mathematical model representing a generic X-Wing aircraft configuration in the stopped-rotor mode was developed and used for this study. A composite material beam finite element and lumped mass fuselage inertial effects were included in the model. An arbitrary untapered blade aerodynamic planform was assumed and unsteady generalized aerodynamic forces were computed using structural vibration mode data from the structural model and a Doublet Lattice aerodynamic panel method computer code. Parameters, each varied independently, define the model configuration and include vehicle center-of-gravity location, X-Wing rotor to vehicle mass ratio, the angle between forward- and aft-swept blades, and a blade composite material stiffness cross coupling parameter. For each parameter variation, stability as a function of velocity at sea-level flight conditions was determined.

Accomplishment Description - The aeroelastic stability results for each parameter variation were analyzed to determine trends and develop design guidelines for improving X-Wing aircraft aeroelastic stability characteristics. In all cases the predominant aeroelastic instability was a low frequency coupling or interaction of the first elastic rotor mode with short period motions to produce a "body-freedom" flutter condition. An illustration of the instability is presented as a function of dynamic pressure in figure 49(b). Variation of the flutter boundary as a function of mass ratio is on the lower left. A decrease in stability is observed as the vehicle mass ratio increases and the vehicle c.g. moves forward. In the middle figure, aeroelastic stability as a function of sweep angle is illustrated. A result from this parameter variation was that increased forward blade bending relative to aft blade bending has a stabilizing effect. The effect of stiffness cross-coupling induced washin and washout of the rotor blades is shown on the right. Aeroelastic washout of the aft-swept blade is the most beneficial in terms of stability for the nominal configuration.

Significance - These results show that X-Wing aircraft in the stopped-rotor flight configuration develop unstable aeroelastic interactions between the rigid-body and structural vibration modes at low velocities.

Future Plans - Results from the study will be presented at the AIAA Structures, Structural Dynamics and Materials Conference to be held at Mobile, Alabama in April 1989.

Figure 49 (a).

ANALYSIS ESTABLISHES GUIDELINES FOR AVOIDING AEROELASTIC INSTABILITIES OF X-WING AIRCRAFT

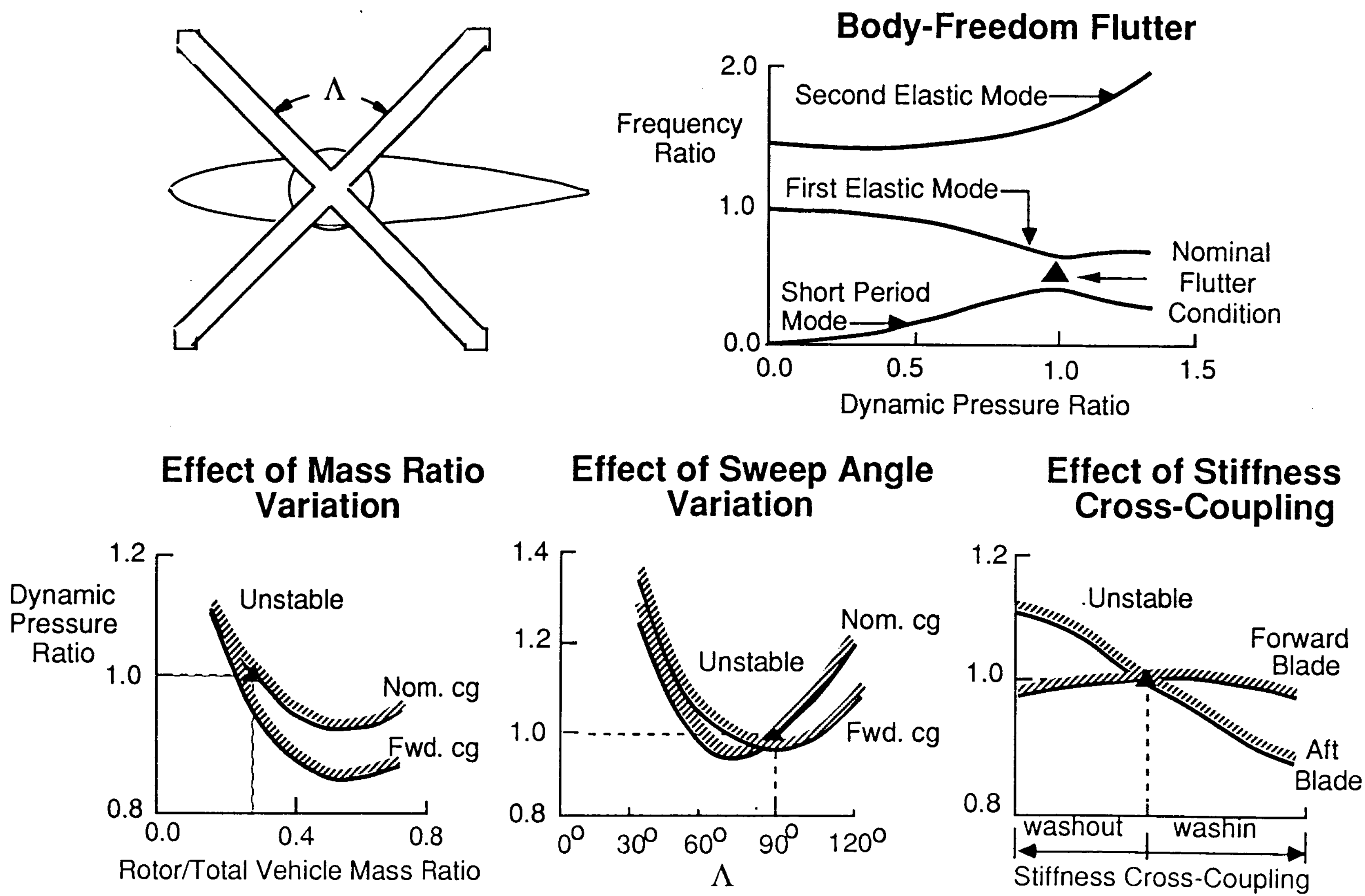

Figure 49 (b).

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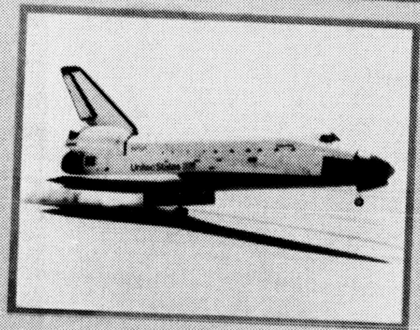
LANDING AND IMPACT DYNAMICS BRANCH



Research opportunities

- Reduce fatalities
- Improve landing gears, tires and runways
- Reduce crash loads with loading-limiting structure

Figure 50.

LANDING AND IMPACT DYNAMICS BRANCH

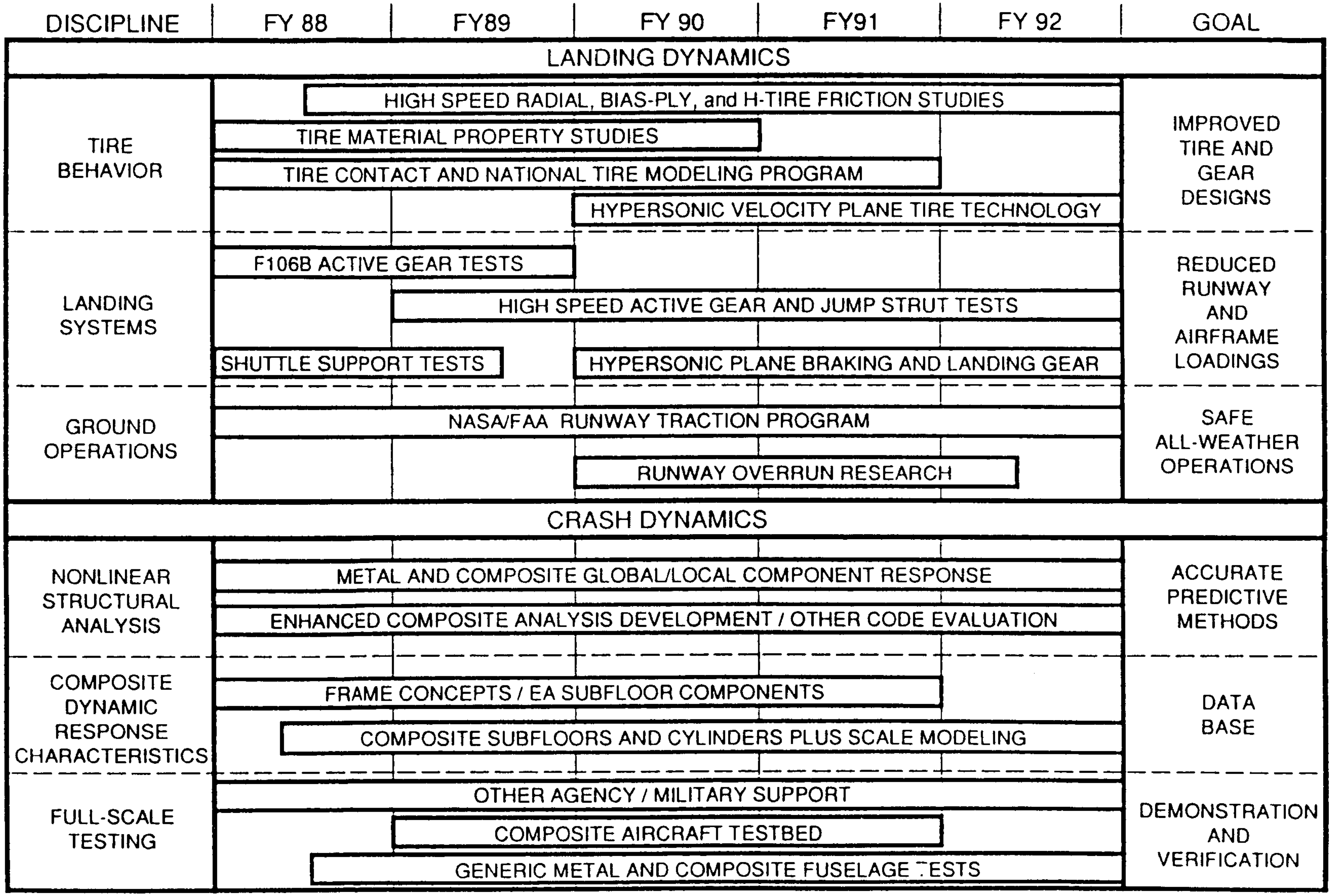


Figure 51.

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30 X 11.5 - 14.5, TYPE VIII, RADIAL AIRCRAFT TIRE PROGRAM

Pamela A. Davis and Mercedes C. Lopez
Landing and Impact Dynamics Branch

RTOP 505-63-41

Research Objective - The objective of this research is to measure the mechanical properties of 30 X 11.5 - 14.5, type VIII, radial aircraft tires and compare these properties to those of the presently used bias-ply tire.

Approach - Static and dynamic mechanical properties tests of a 30 X 11.5 -14.5 radial and bias-ply tire (figure 52(b)) at rated inflation pressure have been conducted at the NASA Langley Aircraft Landing Dynamics Facility (ALDF). Static vertical, lateral, and fore-and-aft tests and fore-and-aft free vibration tests have been conducted in order to analyze the stiffness and damping characteristics of each tire. Footprint geometrical properties tests gave an indication of tire hydroplaning characteristics.

Accomplishment Description - The static mechanical properties tests showed that the vertical and lateral stiffness characteristics of the radial tire were similar to those of the bias-ply tire indicating the landing dynamics of aircraft equipped with radial tires would be the same as aircraft with bias-ply tires. Static fore-and-aft stiffness values for the radial tire were approximately 30 per cent less than the bias-ply tire. This reduced stiffness of the radial tire could be detrimental to the dynamics of antiskid braking systems designed for bias-ply tires (figure 52(c) & 52(d)). The static hysteretic loss of the radial tire was less than for the bias-ply tire indicating that the radial tire generates less heat during normal operation and, therefore, runs cooler. The nearly circular shape of the radial tire footprint at its rated inflation pressure may interfere with the improved hydroplaning potential associated with higher inflation pressures.

Significance - In the past, the general belief in the aircraft landing-gear industry was that the mechanical properties of radial tires were unacceptable for use on aircraft. Research on radial aircraft tires is ongoing because of a need to establish a national database and thus establish the validity of radial aircraft tires. Tests on the 30 X 11.5 - 14.5 radial and bias-ply tires are being conducted to begin establishing such a database.

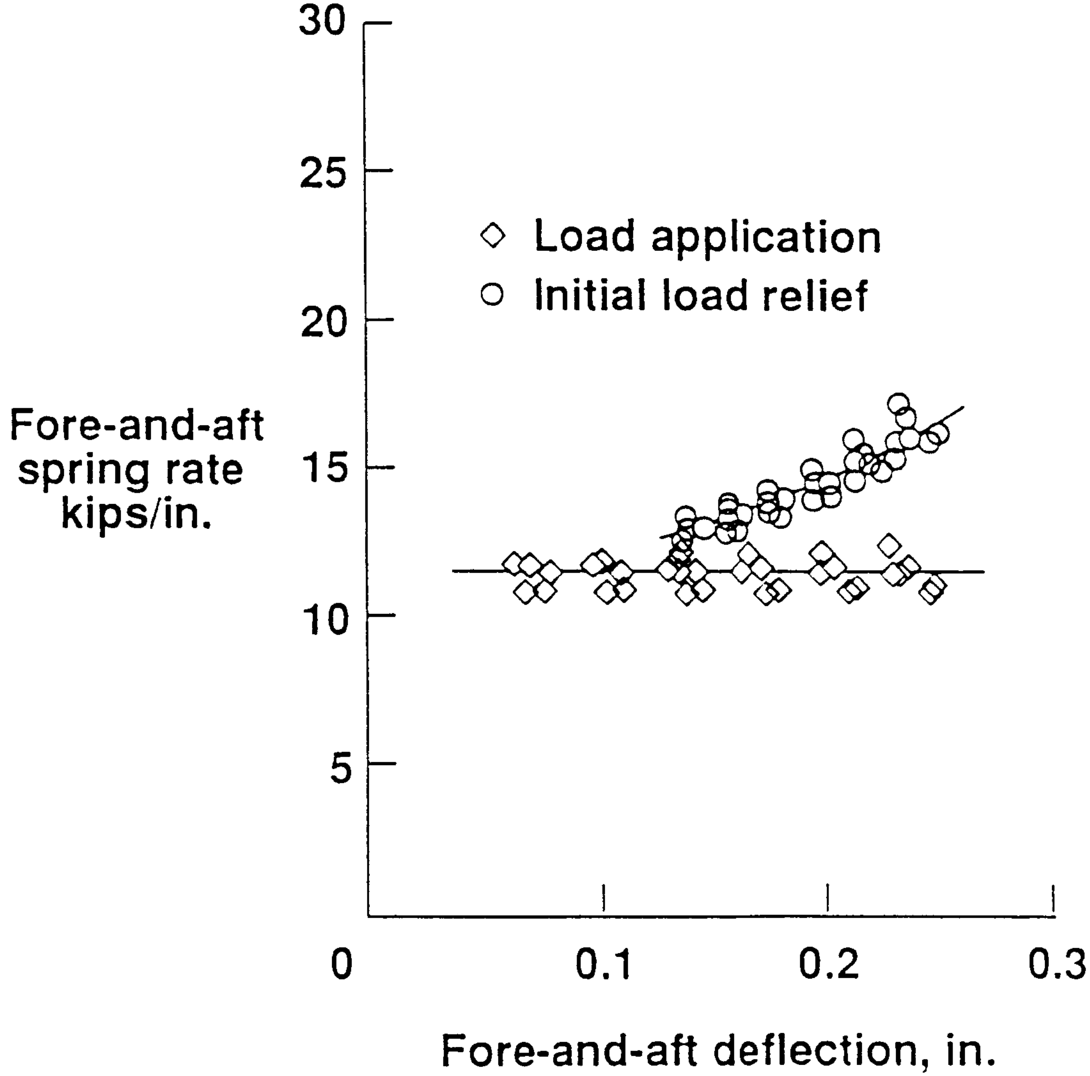
Future Plans - Lateral free-vibration, torsional stiffness, static relaxation length tests will be conducted at the ALDF. Also, free-rolling and antiskid braking tests up to 160 knots and 9° yaw angle will be conducted on the track.

Figure 52 (a).


BIAS-PLY AND RADIAL-BELTED TIRES, UNINFLATED
Bias-ply
Radial-belted

Figure 52 (b).

FORE-AND-AFT SPRING-RATE CURVE
Radial-belted tire



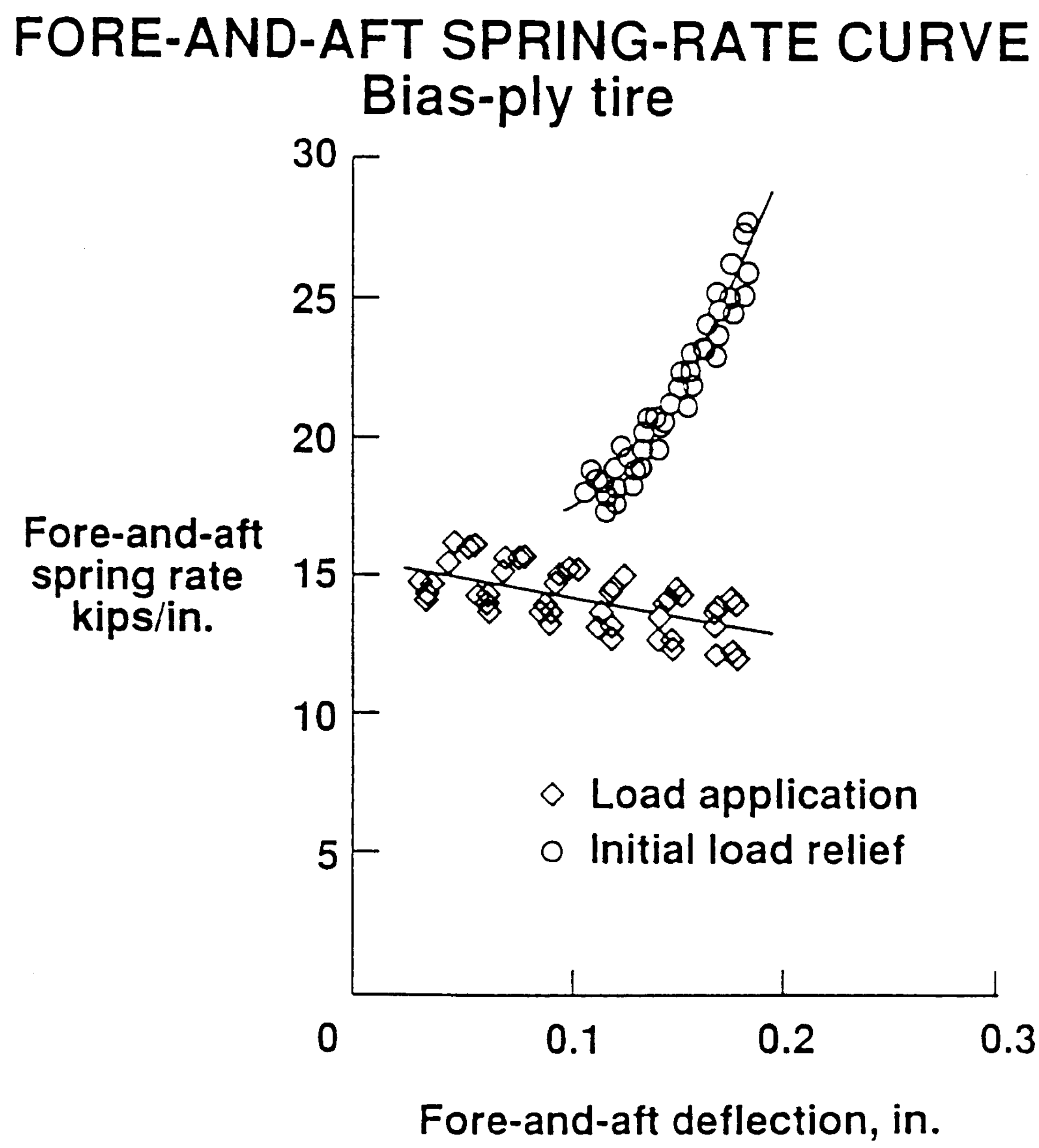

FORE-AND-AFT SPRING-RATE CURVE
Bias-ply tire
Fore-and-aft spring rate kips/in.
30
25
20
15
10
5
0
0.1
0.2
0.3
Fore-and-aft deflection, in.
Load application
Initial load relief

Figure 52 (d).

CORDUROY TEXTURE IDENTIFIED AS SOLUTION TO SPIN-UP WEAR DAMAGE PROBLEM AT THE KENNEDY SPACE CENTER SHUTTLE LANDING FACILITY

Robert H. Daugherty and Sandy M. Stubbs
Landing and Impact Dynamics Branch

RTOP 505-63-41

Research Objective - The objective of this research is to define a texture modification for the touchdown zones of the KSC Shuttle Landing Facility that reduces spin-up damage to an acceptable level while maintaining adequate cornering friction when wet.

Approach - A variety of texture modifications were investigated at the Aircraft Landing Dynamics Facility (ALDF) and on the KSC runway. These modifications included: smooth concrete; smooth-grooved; longitudinal corduroy with spacing of 4, 4 1/2, 5. and 7 blades per inch; and 4 1/2 blades per inch corduroy with transverse grooves. Tests were conducted at the ALDF to investigate the spin-up wear and cornering behavior of the smooth, smooth-grooved, and 4 1/2 blades per inch corduroy both grooved and ungrooved. The Instrumented Tire Test Vehicle (ITTV) was taken to KSC and used to evaluate the differences between 4, 4 1/2, 5, and 7 blades per inch corduroy which had been cut right into the existing runway in a test area (figure 53(b)). Then the ITTV was brought back to the ALDF and used to compare the KSC data with the 4 1/2 blades per inch corduroy texture which had been cut into the ALDF track. In addition, a simulated rollout was performed at the ALDF to predict tire wear for spin-up on the 4 1/2 blades per inch ungrooved corduroy surface and subsequent rollout on the KSC texture.

Accomplishment Description - Test results showed that spin-up damage caused by the 4 1/2 blades per inch ungrooved, corduroy surface is no worse than that caused by a smooth, ungrooved surface (the minimum possible except for lakebed landings). ITTV tests at KSC showed that the 4 1/2 blades per inch ungrooved corduroy was preferred over 4, 5, or 7 blades per inch corduroy, and the ITTV tests at Langley showed that Langley corduroy data was comparable to KSC data, thus validating Langley wear predictions for KSC landings using the modification. Wet friction on the corduroy surface is acceptable, at 70% of the dry KSC surface value. Figure 53(c) shows details of the runway lengths with the corduroy texture modification.

Significance - The wear and friction benefits of the corduroy runway surface should translate into improved safety margin at present crosswind limits, or a crosswind capability increase of 3 to 5 knots at present safety margins.

Future Plans - Evaluate the KSC modified runway friction and wear characteristics with the ITTV.

Figure 53 (a).

NASA L-88-5340

AERIAL VIEW OF KSC SHUTTLE LANDING FACILITY

APPROACH END R/W 15; MAY 1988

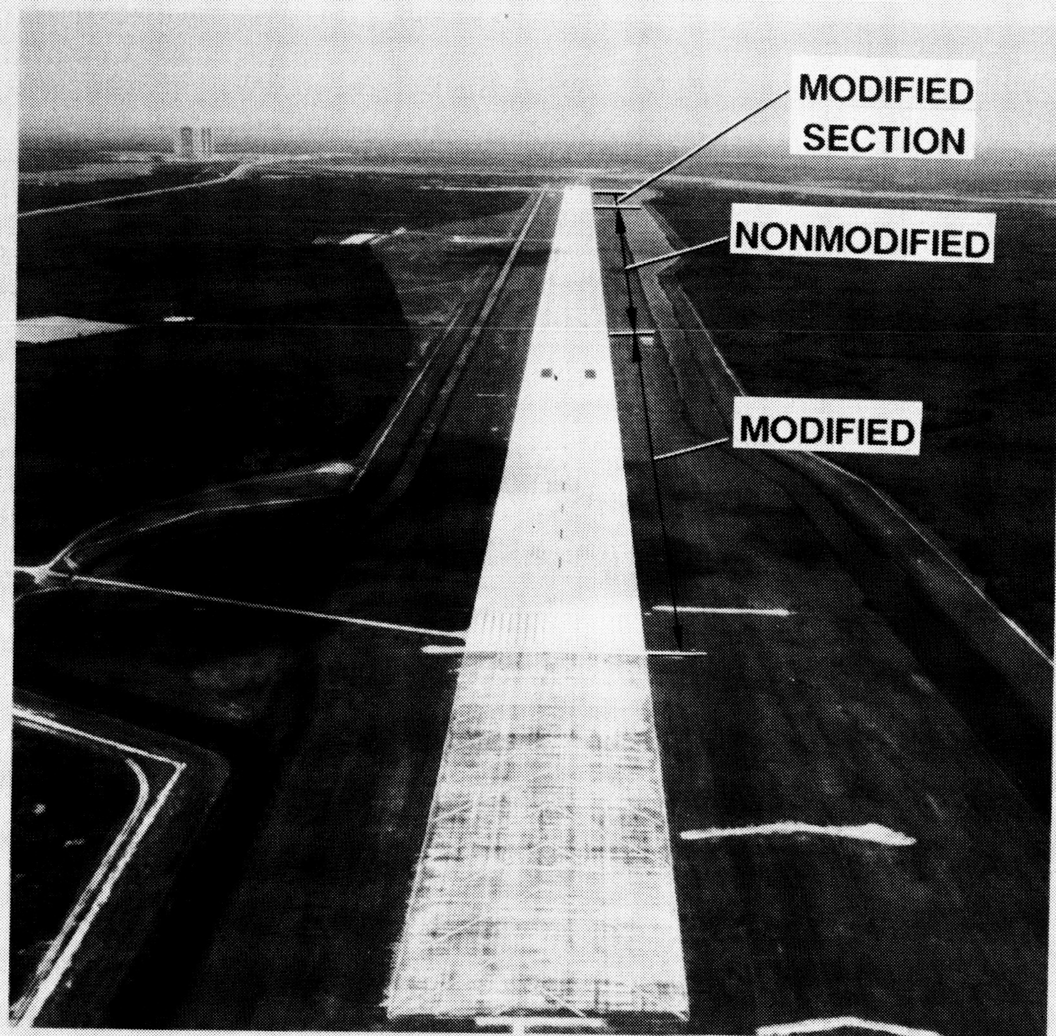
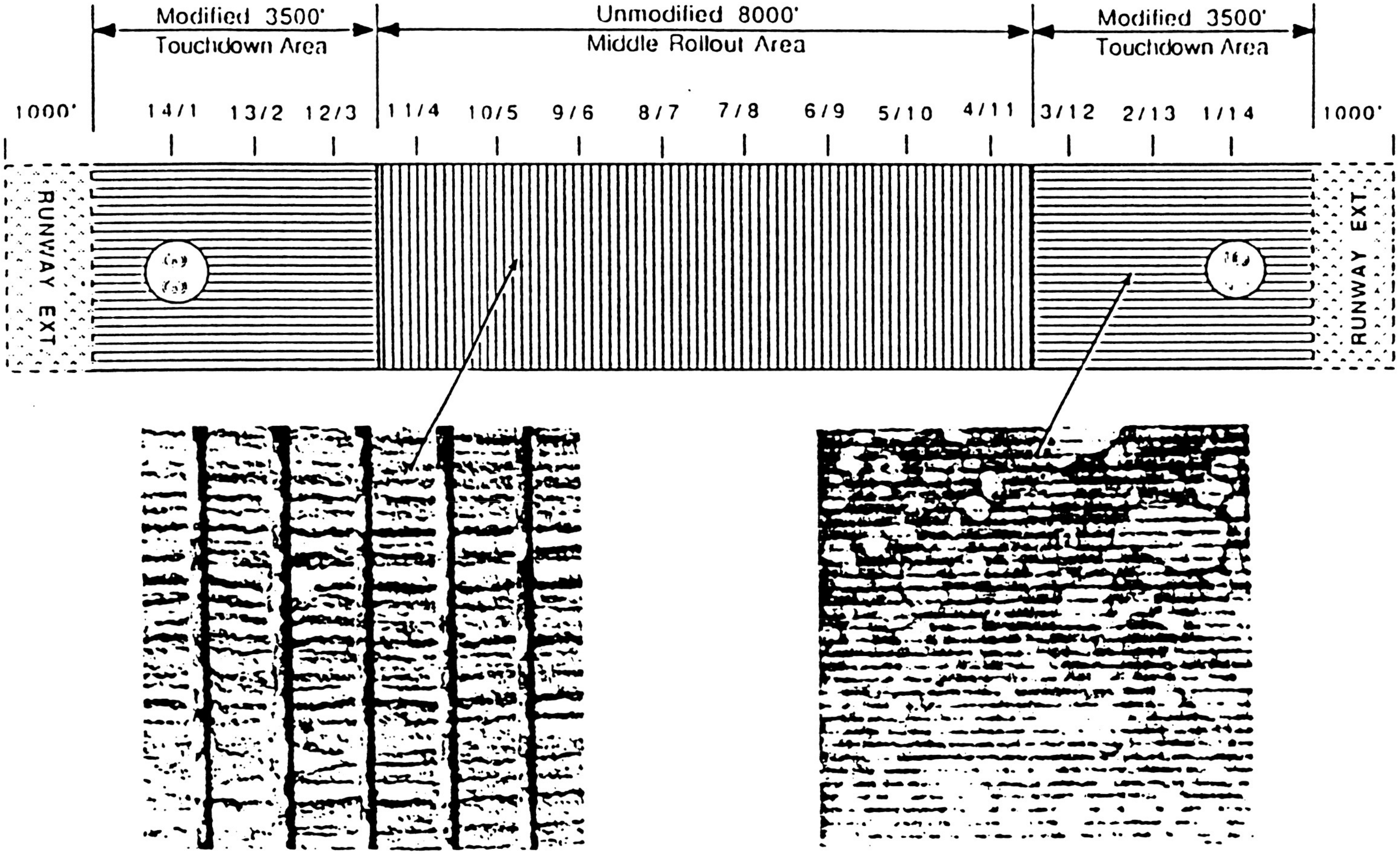


Figure 53 (b).

NASA KSC SHUTTLE LANDING FACILITY

MODIFIED SURFACE ARRANGEMENT



TRANSVERSELY GROOVED
1/4 X 1/4 X 1 1/8 IN.

LONGITUDINAL GRINDING
4 1/2 BLADES/IN., CORDUROY FINISH

Figure 53 (c).

ORBITER FLAT-TIRE LANDING MODEL IS DEVELOPED

Robert H. Daugherty
Landing and Impact Dynamics Branch

RTOP 505-63-41

Research Objective - The objective of this research is to define the drag characteristics of the Space Shuttle Orbiter main gear strut assembly during the sequential destruction of its components for a flat-tire landing on a concrete runway.

Approach - The Orbiter fleet is now equipped with strain-gaged wheels to provide a measure of tire pressure before and during a flight. The astronaut office has raised concerns about the consequences of a flat tire detected in orbit for landing. Although this type of failure is extremely remote, tests were conducted at the Aircraft Landing Dynamics Facility (ALDF) to examine the friction behavior of various components of the lower strut assembly at high speeds. Basically, landing with one deflated tire on a strut almost guarantees that the other tire will fail due to overload during the landing. The first tests conducted involved sliding various skid specimens down a simulated concrete runway at higher speeds and bearing pressure than ever tested before. This provided data relative to the skidding friction of the orbiter brake stack, axle, or strut piston. Next, tests were conducted on bare orbiter wheels to determine how far they could roll under realistic loads and speeds in the event the failed tire is ripped off the wheel during the rollout. Wheels with modified flanges were also tested in hopes of increasing the roll distance. Finally, a test was conducted involving landing a deflated tire mounted on the orbiter wheel to determine the rolling friction of this configuration.

Accomplishment Description - Test results given in figure 54(b) show that landing with a flat tire produces rolling resistance of 0.2, and after about 1000 feet the flat tire destroys itself and leaves behind the tire bead, which causes a rolling resistance of 0.1. After bead failure the wheel loads are high enough to fracture the wheel and the wheel was found to survive less than one second as shown in the figure. For the remainder of the rollout, modeled by ALDF skid material testing, relatively constant sliding friction is indicated until the speed is fairly low.

Significance - Based on these data, it was determined that landing with a flat tire is controllable on a concrete runway. The wheel is almost certain to fracture during this type of landing. A review of skid data on lakebed surfaces shows that the skidding friction on a lakebed will be too great for orbiter directional control. These data were presented at a Program Review Change Board (PRCB) and the flat-tire landing priority runway was changed from Edwards lakebed to a concrete runway at Edwards.

Future Plans - A Convair 990 aircraft is being modified to permit full-scale orbiter strut and wheel testing under failure conditions on both concrete and lakebed runways.

ORBITER FLAT-TIRE LANDING MODEL IS DEVELOPED

1 TIRE FLAT PRIOR TO TOUCHDOWN

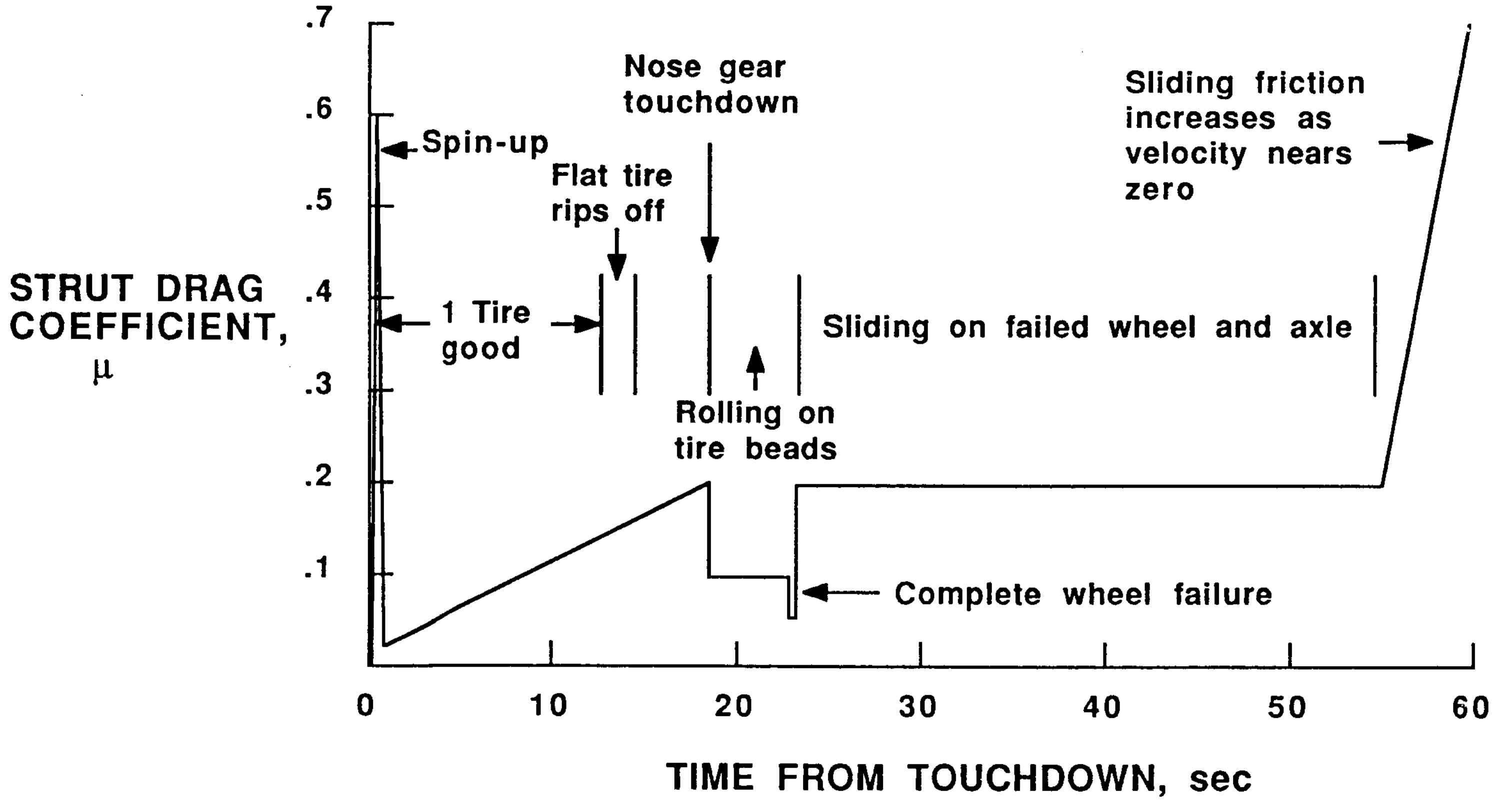


Figure 54 (b).

SCALE MODEL TEST GUIDE SHUTTLE NET ARRESTMENT SYSTEM DEVELOPMENT

Sandy M. Stubbs and Pamela A. Davis
Landing and Impact Dynamics Branch

RTOP 505-63-41

Research Objective - The objective of this research is to develop a net arrestment system to stop the Space Shuttle Orbiter with a minimum of damage if a landing anomaly resulted in a runway overrun incident.

Approach - Shuttle orbiter net arrestment tests have been conducted at the NASA Langley Aircraft Landing Dynamics Facility (ALDF) using a 1/27.5 scale model shown in figure 55(b), to analyze the net-orbiter interaction. Approximately 120 tests have been conducted at simulated speeds up to 95 knots full-scale using five nets of different geometries. Areas of interest were the effect of net height on orbiter arrestments, the effect of various net geometries on net entanglement of the nose gear and main gear, whether or not the top horizontal bundle contacts the wind screen and where it comes to rest on the cargo bay doors, and the dynamics of off-center engagements.

Accomplishment Description - Scale model testing has resulted in modifications of the layout of the energy absorbing system and the net suspension tearaway system to minimize the chances of the net falling under the wing for some slower speed engagements. Approximately 20 tests have been conducted using a net built to the geometric specifications proposed for use on the orbiter and these tests have indicated that no more than three vertical net straps will be caught by the nose gear and they will be broken for engagement speeds greater than 60 knots. 1/8 scale model test confirmed the 1/27.5 scale model results.

Full-scale pull-through tests have been conducted at Dulles International Airport using the Enterprise to determine the interface of the net and the orbiter nose gear and to determine the distribution of the net along the wings and over the payload bay doors.

Significance - There are a number of landing sites designated for use by the Space Shuttle Orbiter and they are all within the capabilities of the orbiter. However, there is the possibility of landing anomalies that could lead to a hazardous runway overrun incident. A runway overrun has the potential of significantly damaging the orbiter and the possibility of injury or loss of crew. A properly designed net arrestment system can bring the orbiter to a safe stop with a minimum of damage.

Future Plans - Install three arrestment systems at transatlantic abort runways.

Figure 55 (a).

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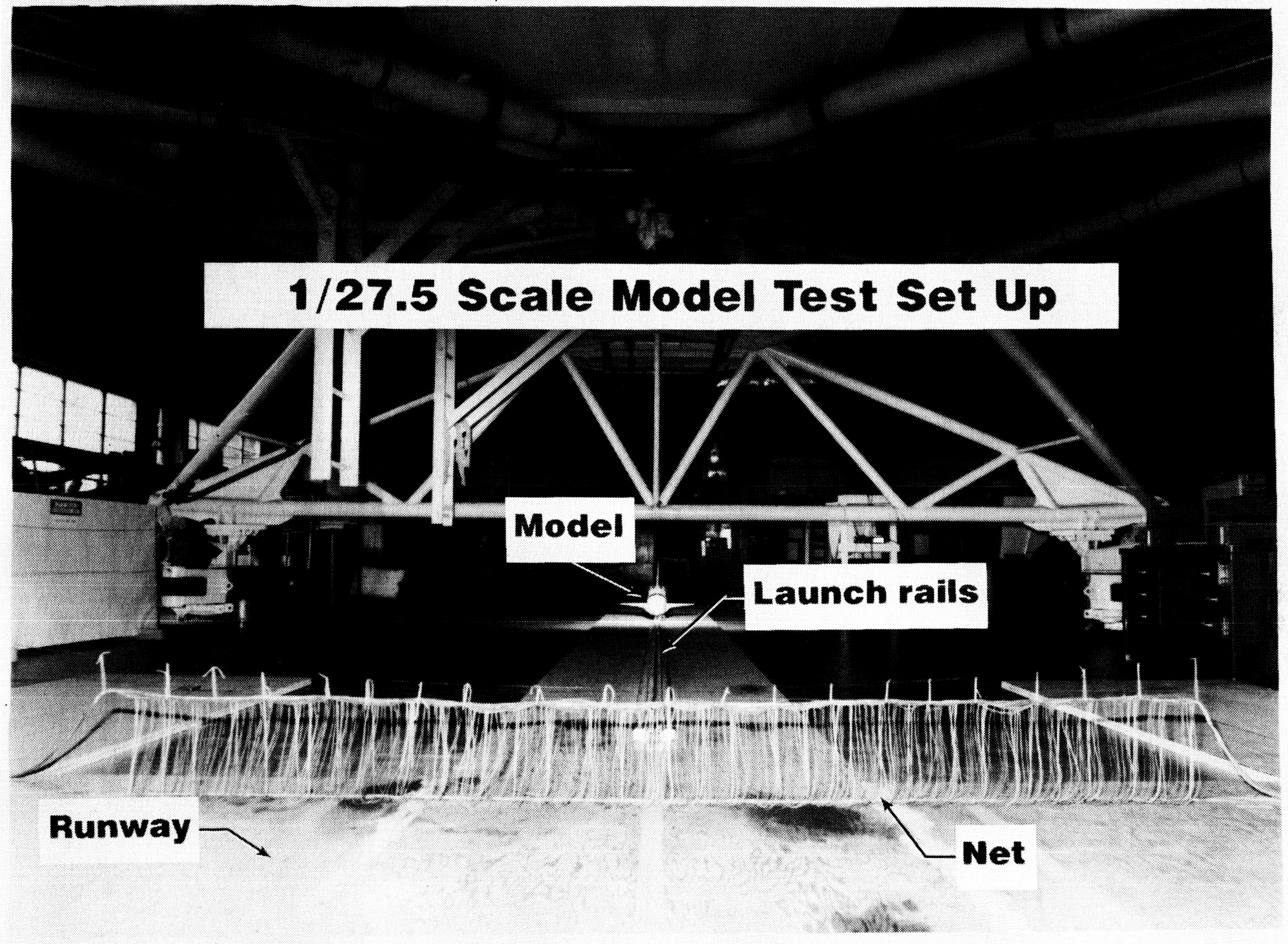


Figure 55 (b).

F-106B AIRPLANE ACTIVE CONTROL LANDING GEAR PROGRAM NOSE GEAR DROP TESTS

William E. Howell and John R. McGehee
Landing and Impact Dynamics Branch

RTOP 505-63-41

Research Objective - The objective of this research is to demonstrate the feasibility and the potential of a series-hydraulic active-control gear during aircraft landing and ground operations.

Approach - A set of F-106B landing gear have been obtained and modified to perform in either the passive or active modes. An electronic controller has been designed and fabricated and is being checked out for the gear. Vertical drop tests will be conducted in the laboratory to verify both the controller and each modified gear. Controllers (one for each of the three gears), a flight data acquisition system, and a flight hydraulic system will be used for further drop tests for final verification of all flight systems. The entire system will then be installed on the F-106B airplane for landing and taxi tests (figure 56(b). The data obtained from the drop, landing, and taxi tests for both passive and active operation modes will be compared with analytical results.

Accomplishment Description - Test results given in the figure 56(c) are for vertical drop tests conducted on the modified nose gear. The solid curve is for the accelerations experienced by the passive gear mass as a function of time when dropped from a height of 0.3 inches with no lift simulation. The dashed curve is for the same situation with the gear in the active mode with only the force control loop functioning. A 42% reduction in G-forces was achieved in this test. The three high peaks experienced during impact of the active gear indicate that adjustments still must be made to the controller.

Significance - The preliminary data indicate that a significant reduction in the forces at the landing gear -- air frame interface can be achieved by active control of the F-106 airplane nose landing gear.

Future Plans - Get the controller fully operational, continue the assembly of major flight components, complete the drop tests, and have the complete system ready to install on the F-106 as soon as the Vortex Flap Flight Test Program is completed.

Figure 56 (a).

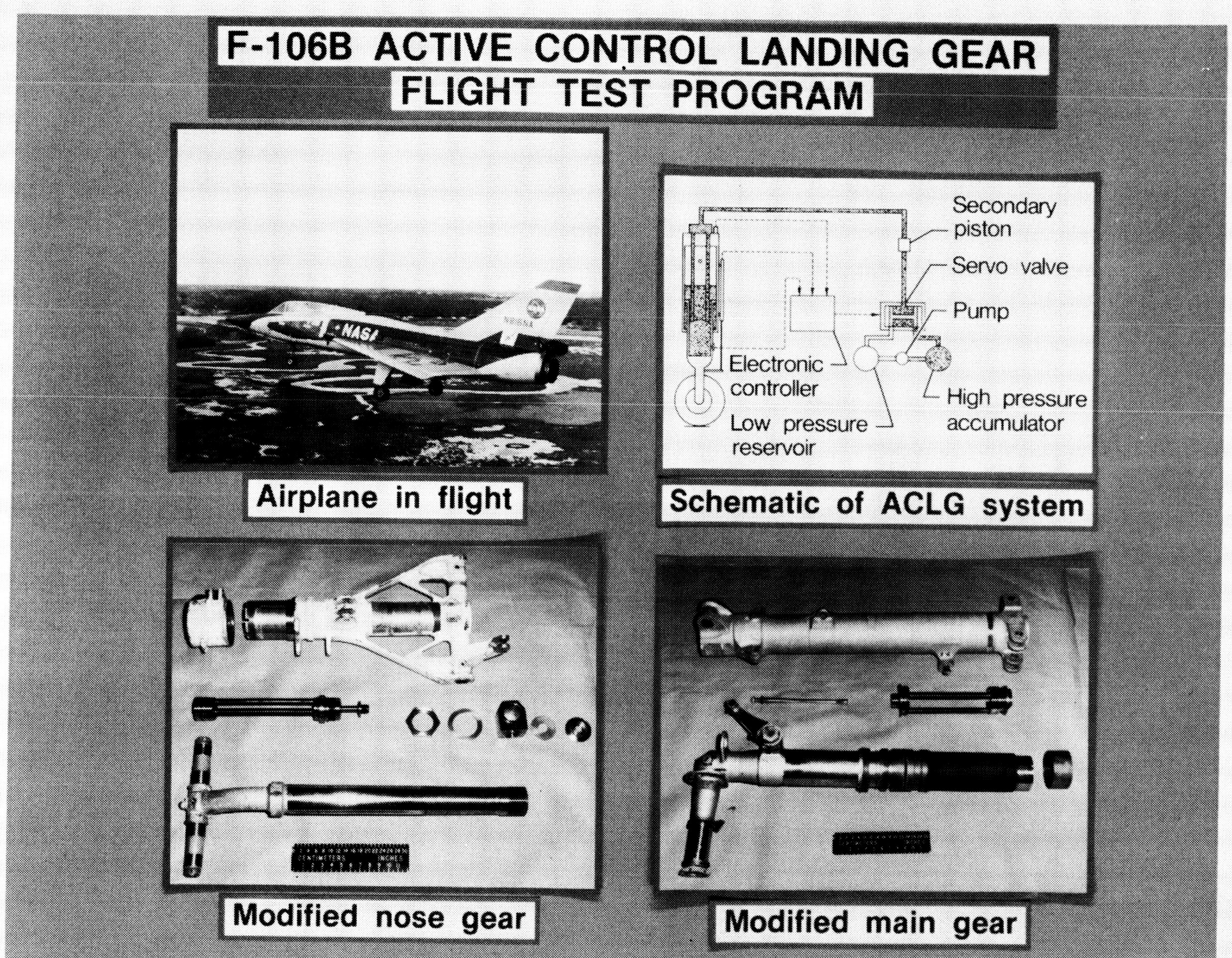
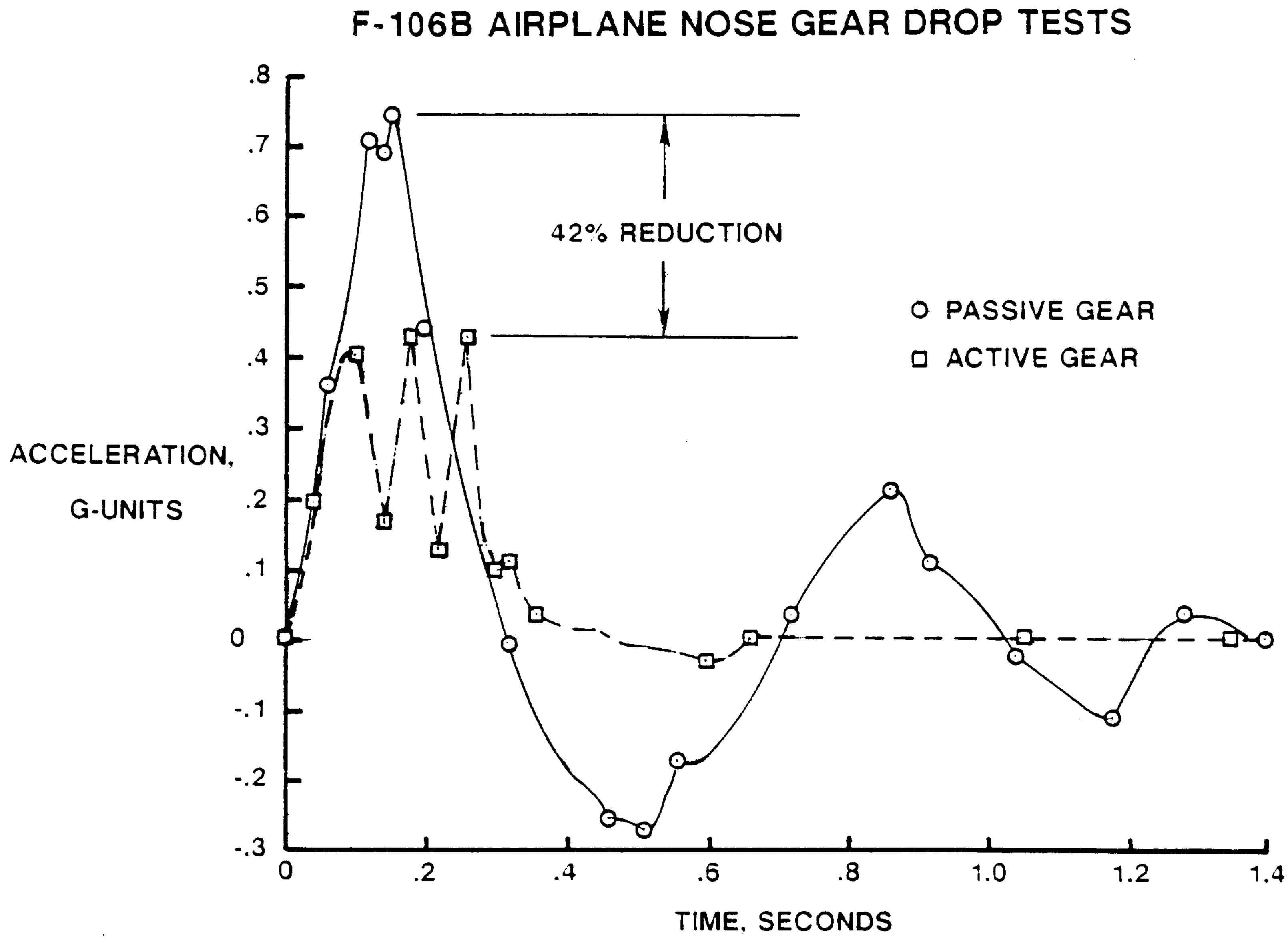

F-106B ACTIVE CONTROL LANDING GEAR
FLIGHT TEST PROGRAM
Airplane in flight
NASA
Secondary piston
Servo valve
Pump
Electronic controller
High pressure accumulator
Low pressure reservoir
Schematic of ACLG system
Modified nose gear
Modified main gear

Figure 56 (b).



F-106B AIRPLANE NOSE GEAR DROP TESTS
42% REDUCTION
O PASSIVE GEAR
□ ACTIVE GEAR
ACCELERATION,
G-UNITS
.8
.7
.6
.5
.4
.3
.2
.1
0
-.1
-.2
-.3
0
.2
.4
.6
.8
1.0
1.2
1.4
TIME, SECONDS

Figure 56 (c).

RUNWAY FRICTION WORKSHOP

T. J. Yager
NASA Langley Research Center

RTOP 505-63-41

Research Objectives -

- o Provide aviation industry with an advance review of Joint FAA/NASA Runway Friction Program test results.
- o Obtain industry's comments concerning test results and future test plans.

Approach -

- o 18 formal presentations were made to approximately 80 attendees.
- o Organizations represented included U.S., Canadian, and Swedish government agencies, airlines and pilots, airframe manufacturers, airport managers, ground test vehicle manufacturers/suppliers, and tire companies.
- o Runway friction work conducted in Sweden, England, France, Japan, and Canada was presented.
- o Joint FAA/NASA program test results were presented by NASA, FAA, PRC Kentron, Boeing, and retired NASA consultants
- o New ground vehicle test equipment and use of liquid chemicals to control snow/ice contaminants were also discussed.

Accomplishments -

- o Workshop attendees given complete copies of all presentation figures and summary of workshop proceedings prepared and distributed to over 180 aviation industry personnel.
- o Joint program draft reports have been modified and improved based upon workshop discussions.

Significance -

- o Findings discussed at workshop could lead to improved safety of aircraft ground operations during adverse weather/runway conditions.

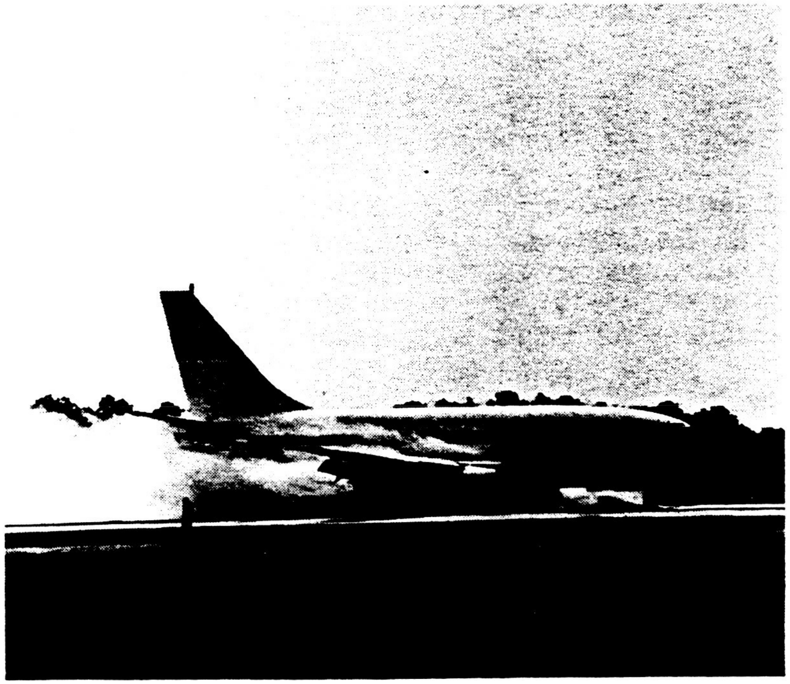
Future Plans -

- o Proceed with Joint NASA/FAA surface traction program using ALDF to evaluate radial constructed transport aircraft tires.
- o Collect copies of ground vehicle/aircraft friction data forms currently used at U.S. and foreign airports in order to design new standardized form for use at all U.S. airports.
- o Participate in FAA Headquarters meetings next year to determine how test findings impact existing advisory circulars, standards, and regulations.

Figure 57 (a).

RUNWAY FRICTION WORKSHOP

October 26 & 27, 1988



National Aeronautics and Space Administration
Langley Research Center
Hampton, Virginia

Department of Transportation
Federal Aviation Administration

NASA/DOT/FAA SPONSORS

- 2–DAY MEETING HOSTED AT LANGLEY RESEARCH CENTER
- REVIEW WITH AVIATION COMMUNITY THE TEST RESULTS FROM THE JOINT FAA/NASA AIRCRAFT/GROUND VEHICLE RUNWAY FRICTION PROGRAM
- 150 INVITATIONS
- 18 PRESENTATIONS
- 80–85 ATTENDEES (FROM 10 COUNTRIES)
- VERY SUCCESSFUL WORKSHOP

Figure 57 (b).

STATIC RESPONSE OF COMPOSITE FUSELAGE FLOOR SECTIONS

Richard Boitnott
Landing and Impact Dynamics Branch

RTOP 505-63-01

Research Objective - The objective of this research is to determine the behavior of composite aircraft structure under crash conditions through detailed experimental and analytical studies of structural subcomponent tests.

Approach - Composite materials are presently being investigated as the primary structural material for the next generation of aircraft. A program has been initiated at NASA-Langley Research Center to address the behavior of composite aircraft structure under crash conditions using structural subcomponent tests.

Accomplishment Description - Circular six-foot diameter graphite-epoxy fuselage floor sections were statically tested as an initial step in assessing the response of such structural subcomponents to vertical impacts. Two fuselage floor sections have been tested. Both have an identical skeletal framework consisting of three graphite-epoxy six-foot diameter semi-circular Z cross-section frames (Z-frames), three aluminum floor beams, and fifteen graphite-epoxy stringers. Notches were machined into the outside flanges of the Z-frames to accommodate stringers which are attached to the frames with aluminum shear clips and rivets. One specimen had a graphite-epoxy skin bonded and riveted to the frames and stringers. The other specimen without skin (referred to as the skeleton specimen) originally 39 inches in height was statically crushed approximately 12 inches (see figure 58(b)). The load-deflection response for the skeleton floor section (figure 58(c)) was reduced to one-third of its value to compare with the load-deflection data obtained from testing a single frame. For the single frame test, the frame was sandwiched between two fences which limited the amount of twisting and out-of-plane bending along the entire length of the beam. The floor section load for one frame agrees very well with load-deflection curve obtained from the test of the single frame. On a test of the skinned specimen (figure 58(d)), the load-deflection stiffness was approximately four times larger than the stiffness measured on the skeleton specimen. The DYCAST (DYnamic Crash Analysis of STructures) finite element computer code was used to predict the load-deflection response using a simpler model similar to the detailed idealization shown in figure 58(e). Analyses were conducted with different boundary conditions on the out-of-plane and twisting deformation of the specimen. The analysis which yielded the best correlation for the skeleton section was obtained with boundary conditions which allowed the frame to bend out-of-plane but not to twist. These assumption agreed with experimental observations. The best correlation for the skinned section was obtained by not allowing the frames to bend out-of-plane or twist.

Significance - Tests of such specimens provide a data base and understanding of the crash behavior of composite structures and will allow development of innovative concepts to improve energy absorption characteristics.

Future Plans - Future dynamic tests and analyses will be conducted on both a skeleton and skinned floor specimen.These tests will be compared to the static floor section tests and dynamic single frame tests.

Figure 58 (a).

Figure 58 (b).

STATIC RESPONSE OF COMPOSITE FLOOR SECTIONS

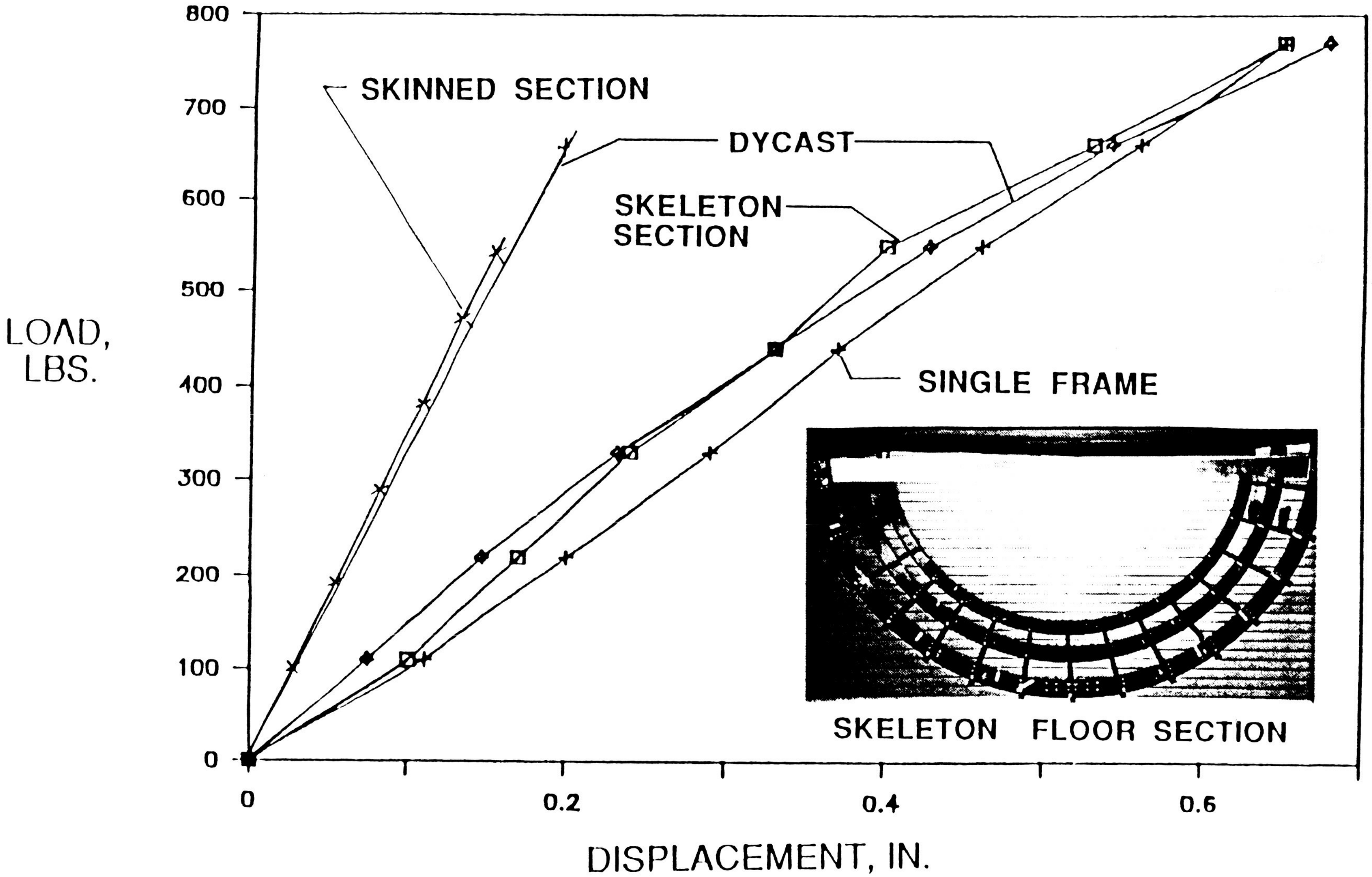
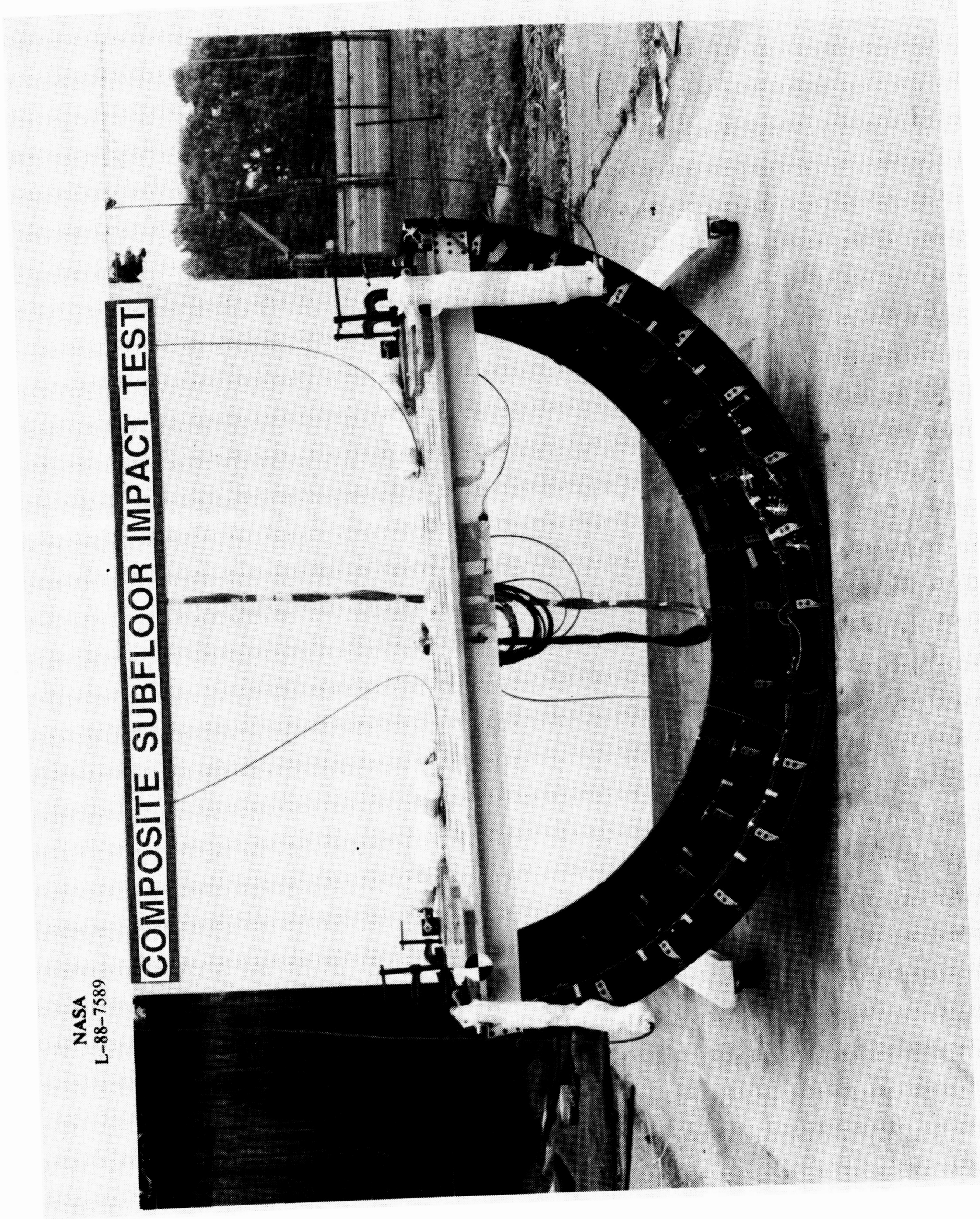


Figure 58 (c).



NASA
L-88-7589

Figure 58 (d).

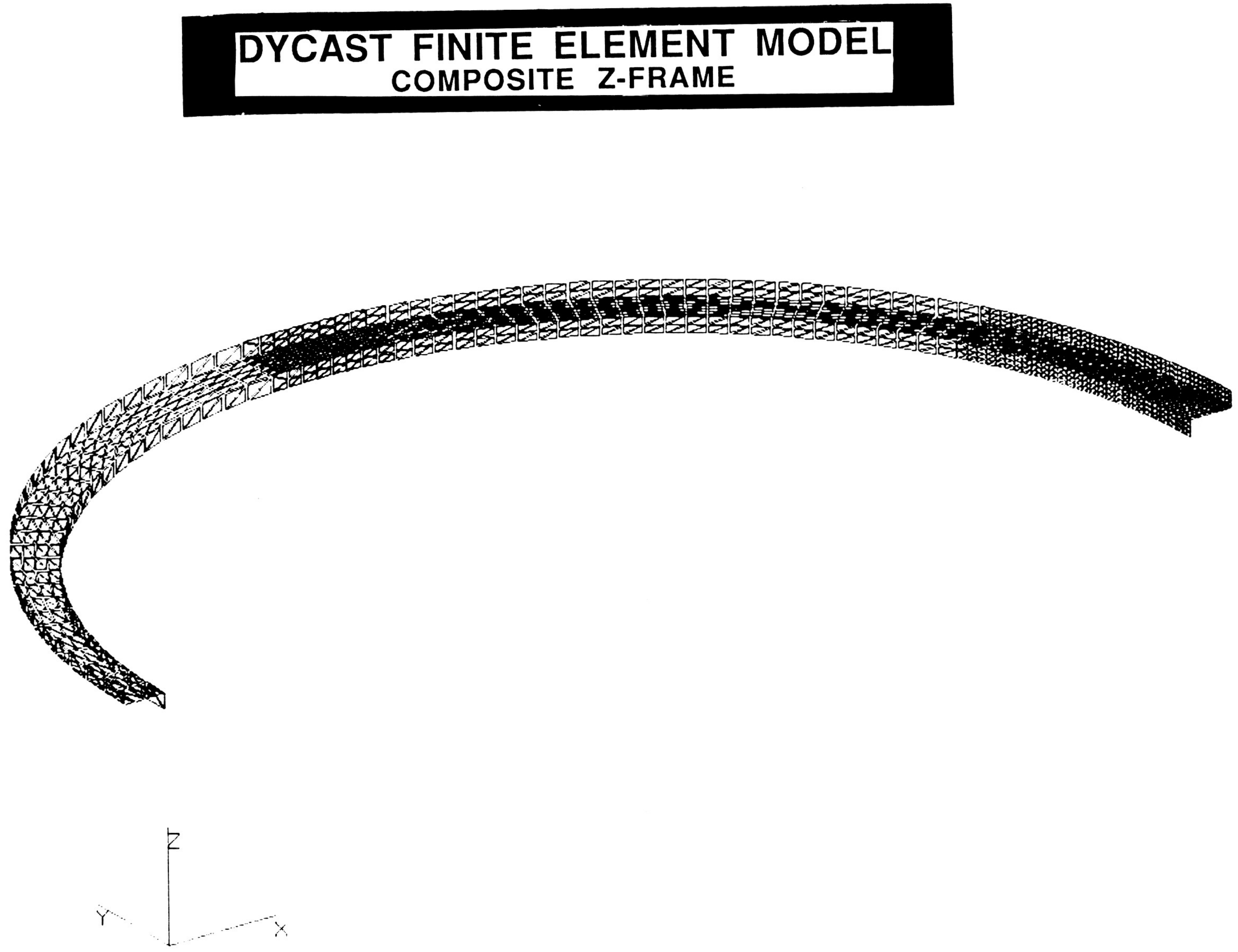


Figure 58 (e).

MODEL DEVELOPMENT FOR ANALYSIS OF THIN-WALLED BEAMS

Dr. Ahmed K. Noor, Dr. B. J. Min and Huey D. Carden
George Washington University & Landing and Impact Dynamics Branch

RTOP 505-63-01

Research Objective - The objective of this research is to develop simple models and an effective computational strategy for simulating the dynamics response of composite structures during impact.

Approach - A simple one-dimensional finite element model is developed based on Vlasov's type thin-walled beam theory. The model accounts for the warping of the cross-section, the transverse shear deformation as well as the anisotropic material response. A perturbed Lagrangian formulation is used to reduce the continuity requirements at interelement boundaries and to simplify the element development. The model has been applied to free vibration problems and is being extended to nonlinear dynamic problems. In the critical regions of the structure, the predictions of the one-dimensional model will be post-processed using two-dimensional plate/shell equations.

Accomplishment Description - The finite element model developed has been tested by comparing its predictions for free vibrational response to those of two-dimensional plate/shell elements. For the semicircular frame with Z cross sections shown in figure 59(b), the maximum error in the lowest four frequencies was less than 4%.

Significance - The one-dimensional model results closely approximate the predictions of the considerably more expensive two-dimensional models. With the post-processing of the one-dimensional results in the critical zones, considerable savings in computer time can be achieved.

Future Plans - Assess the performance of the model developed and the proposed post-processing approach for nonlinear dynamic problems.

Figure 59 (a).

MODEL DEVELOPMENT FOR ANALYSIS OF THIN-WALLED BEAMS

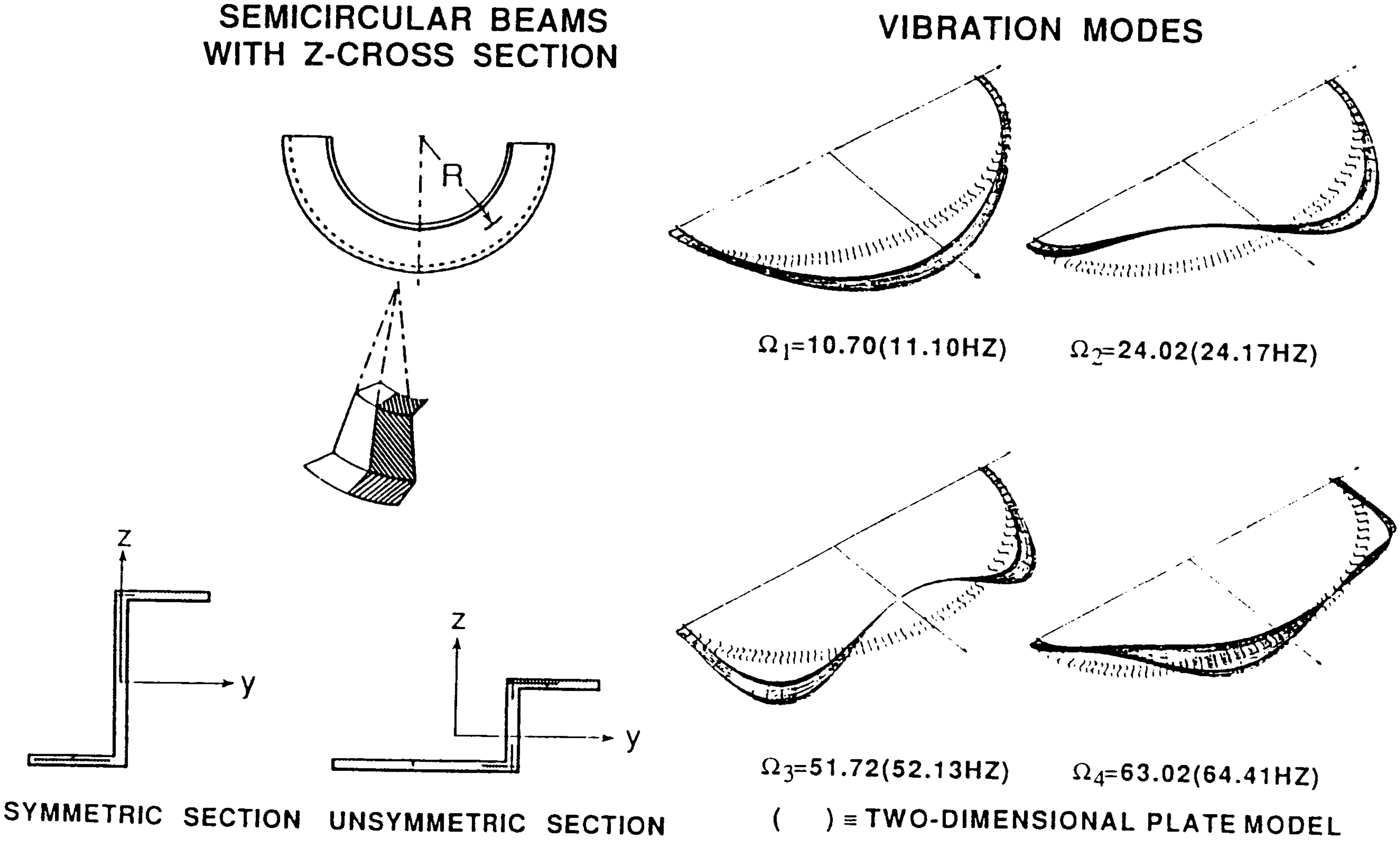


Figure 59 (b).

ENERGY ABSORBING CHARACTERISTICS OF COMPOSITE SUBFLOOR INTERSECTIONS

Lisa E. Jones (PRC), Edwin L. Fasanella (PRC) and Huey Carden
Landing and Impact Dynamics Branch

RTOP 505-63-01

Research Objective - An experimental program has been underway to study collapse behavior, crushing strength, structural integrity, and energy absorbing properties of composite subfloor intersections (cruciforms) for possible application to energy absorbing aircraft subfloor structures which potentially can help limit impact loads to occupants in a crash.

Approach - Crashworthiness of an aircraft is based upon its ability to protect the occupants from serious and/or fatal injury in a crash situation. In all aircraft, there are structures such as landing gear, the seat, and subfloor that can absorb energy. If the over-all energy absorption is optimized by the aircraft designer, the loads transmitted to the occupants can be significantly reduced in the event of a crash. The energy absorption performance of composite subfloor structures is being investigated at the Impact Dynamics Research Facility. Forty-one cruciform specimens constructed of twelve ply ($[\pm 45]_6$) laminates of either Kevlar® 49/934 or AS-4/934 graphite/epoxy with heights of 4, 8, and 12 inches were crushed quasi-statically. Four intersection angle attachment concepts were compared: 1) tapered with cutouts (Kevlar® only), 2) tapered without cutouts, 3) straight with cutouts (Kevlar® only), and 4) straight without cutouts. All specimens were crushed to approximately 25 percent of the original height. The effect of restraining or allowing the ends of the specimens to twist was also studied.

Accomplishment Description - Preliminary data analysis based on energy absorption, loads, and post crush structural integrity indicates that the Kevlar® specimens with tapered intersection angle attachments and no cutouts may be the best concept to limit peak loads. The initial failure loads for these specimens were essentially the same as the sustained crushing loads and were considerably lower than the initial failure loads of the straight intersection specimens. Additionally, the energy absorption increased gradually corresponding to the increased load carrying capability with the widening of the taper, and post crush structural integrity was maintained (See figure 60(b)). The graphite/epoxy specimens generally lost structural integrity and energy absorbing capabilities after crush initiation. Restraining the ends of the specimens appear to have little, if any, effect on the energy absorbing capabilities of the specimens.

Significance - Such research will provide data base of potential designs that meet structural performance, integrity and energy absorption requirements.

Future Plans - A comparative study of the cruciform concepts based upon the energy absorbed, average load, specific energy, specific stress, material response, post crush structural integrity, and peak load will be used to rank the performance of the intersection concepts for possible use in load limiting subfloor structures. The data may also be useful to verify DYnamic Crash Analysis of STructures (DYCAST) and NIKE3D models of such specimens.

Figure 60 (a).

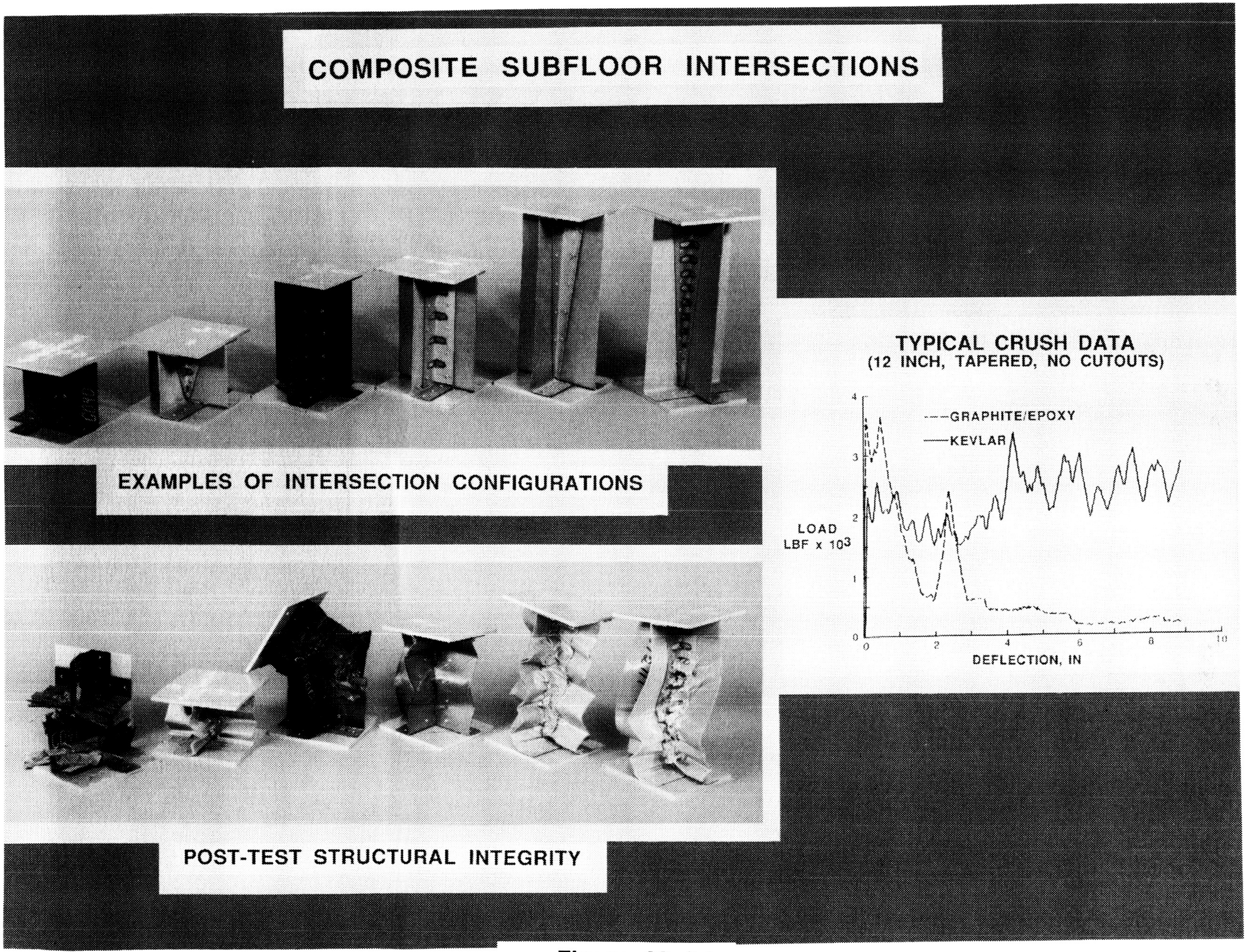

COMPOSITE SUBFLOOR INTERSECTIONS
EXAMPLES OF INTERSECTION CONFIGURATIONS
POST-TEST STRUCTURAL INTEGRITY
TYPICAL CRUSH DATA
(12 INCH, TAPERED, NO CUTOUTS)
GRAPHITE/EPOXY
KEVLAR
LOAD
LBF x 10³
DEFLECTION, IN
0
1
2
3
4
6
8
10

Figure 60 (b).

SCALING EFFECTS IN THE LARGE DEFORMATION BENDING RESPONSE OF COMPOSITE BEAMS

Karen E. Jackson (Army)
Landing and Impact Dynamics Branch

RTOP 505-63-01

Research Objective - An experimental program has been developed to determine the effectiveness of scale model testing for predicting the static large deflection response and failure of composite beams. One of the main goals of the research is to characterize scaling effects in the behavior of the beams such that measurements made on subscale models will accurately predict prototype response.

Approach - Static tests have been conducted at the Impact Dynamics Research Facility on 1/6, 1/4, 1/3, 1/2, 2/3, 3/4, 5/6, and full scale replica model beams of AS4/3502 graphite-epoxy composite material. Beams having lay-ups of unidirectional, angle ply, cross ply, and quasi-isotropic were tested under an eccentric axial compressive load to failure, as shown in figure 61(b). This testing configuration was chosen since it promotes large bending deformations and failures occur in the center of the beam away from the hinged end supports. Test output included vertical load, end deflection, and tensile and compressive strains. Comparisons of the test results should verify the model analysis and indicate whether full-scale composite beam behavior can be determined through inexpensive scale model testing.

Accomplishment Description - Static testing has been completed for graphite-epoxy scale model beams. The figure shows a photograph of the failed quasi-isotropic beams (1/6 through full scale) which indicates that failure mechanisms are the same between the model and prototype beams. For the quasi-isotropic beams failure occurred through a combination of matrix cracking, delamination, and fiber fracture. The normalized load versus end displacement plot shown in the figure shows that the beam response scales in the elastic, small deflection region. Deviation from scaled response is observed as the beam undergoes large deformations and the response becomes nonlinear. A significant scale effect in strength is observed. Failure of the full scale beam occurs at a lower load and substantially lower end displacement than that of the 1/6 scale beam. The trends exhibited by the quasi-isotropic beams are typical of the unidirectional, angle ply, and cross ply beams, even though these laminates have much different failure mechanisms. Stress and strain based failure criterion such as maximum stress, maximum strain, Tsai-Hill, or Tsai-Wu could not predict this observed scale effect in strength.

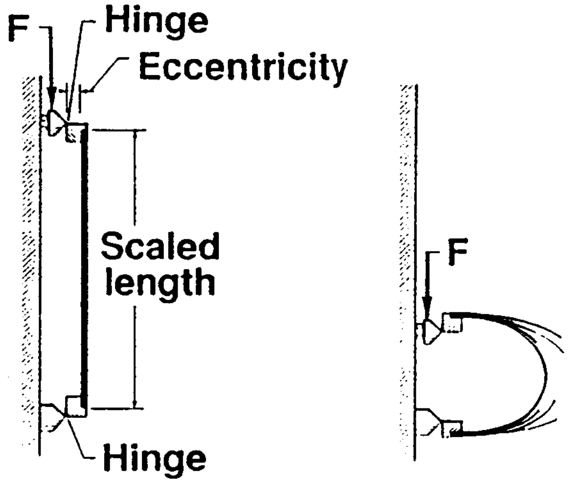
Significance - It is anticipated that the results of this project will encourage the use of scale model tests in studies of the crashworthiness of transport aircraft and helicopters, particularly at the subcomponent level.

Future Plans - Load versus deflection data generated from the static tests will be used to develop a test matrix for impact testing of scaled composite beams. The impact tests will be performed to investigate scaling effects in the beam response and failure under dynamic loads. Some statistical approaches and strength theories based on fracture mechanics are being investigated to explain the scale effect in failure.

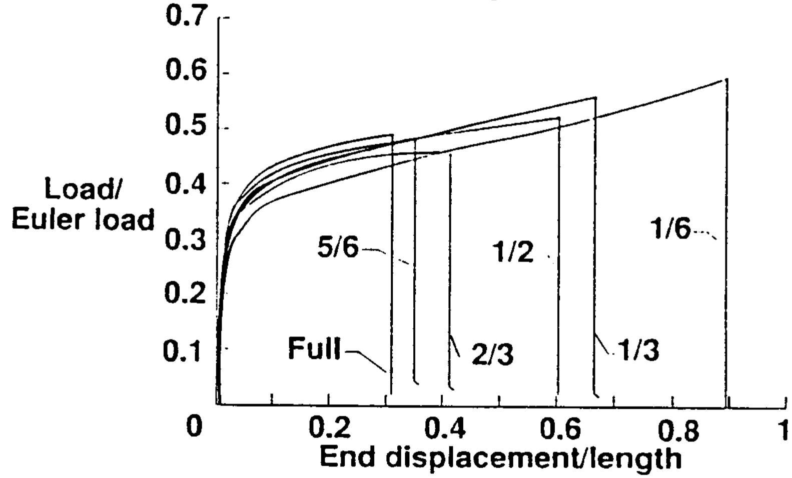
Figure 61 (a).

STATIC TEST RESULTS FOR SCALE MODEL QUASI-ISOTROPIC GR-EP BEAMS

SCHEMATIC DRAWING OF TEST CONFIGURATION



NORMALIZED LOAD VERSUS END DISPLACEMENT



FAILED BEAM TEST SPECIMENS

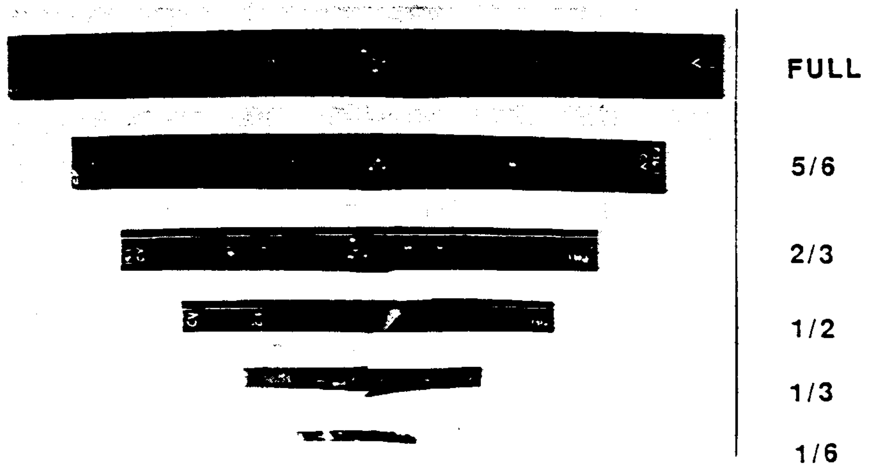


Figure 61 (b).

INTERIM TRANSPORTATION OVERPACK CONTAINER (ITOC) EXPERIMENTS AND ANALYSIS

Edwin L. Fasanella (PRC) and Lisa E. Jones (PRC) and Huey D. Carden
NASA Langley Research Center

RTOP 505-63-01

Research Objective - An experimental and analytical program was conducted on the ITOC system being developed by the U.S. Army Armament Research, Development and Engineering Center (ARDEC). The ITOC is a steel vault-like structure (6 ft long, 4 ft wide and 3 ft high) designed to house three artillery projectiles for security and protection in the event of a helicopter crash and postcrash fire. DYnamic Crash Analysis of STructures (DYCAST) pretest models were constructed to predict responses of the ITOC.

Approach - Drop tests of fully instrumented ITOC packages mounted inside a CH-47 helicopter section were performed at 100 ft/s (197 ft drop height) onto a 5 feet depth of soil over concrete impact surface to test the integrity of the ITOC (figure 62(b)). To limit loads on the ITOC and the interior projectiles, the ITOC was mounted on a balsa cargo pallet containing crushable aluminum honeycomb. Accelerations were expected to exceed any previously measured values on test articles at the IDRF and required the development of special lead mounts for the accelerometers to prevent high frequency overranging. DYCAST models of the ITOC substructure such as the fork lift stirrups were created to develop nonlinear force-deflection curves. The soil, honeycomb, and helicopter floor forces were determined and input into DYCAST as nonlinear springs. In the pretest model the ITOC was treated as a lumped mass. This interactive model ran quickly on the IDRF MicroVax II computer.

Accomplishment Description - In spite of the severity of the impact (over 200 G's), accelerations were successfully measured on the helicopter section, the ITOC, the containers inside, and on the projectiles. Structural integrity of the ITOC was also maintained. The experimental acceleration on the ITOC floor is shown in the figure and is compared with the pretest DYCAST finite element model. The DYCAST model slightly overpredicted the acceleration but can be made more accurate by adjusting the effective planform area used in determining the soil force.

Significance - Static results from detailed models of substructure were successfully used to form hybrid elements for input into a more complex system. The good pretest DYCAST prediction of the experimental ITOC accelerations support the validity of this approach. The validated model predicted that the honeycomb would be effective in limiting the load to the projectiles for velocities up to 150 ft/s.

Future Plans - The results of this study will be documented.

Figure 62 (a).

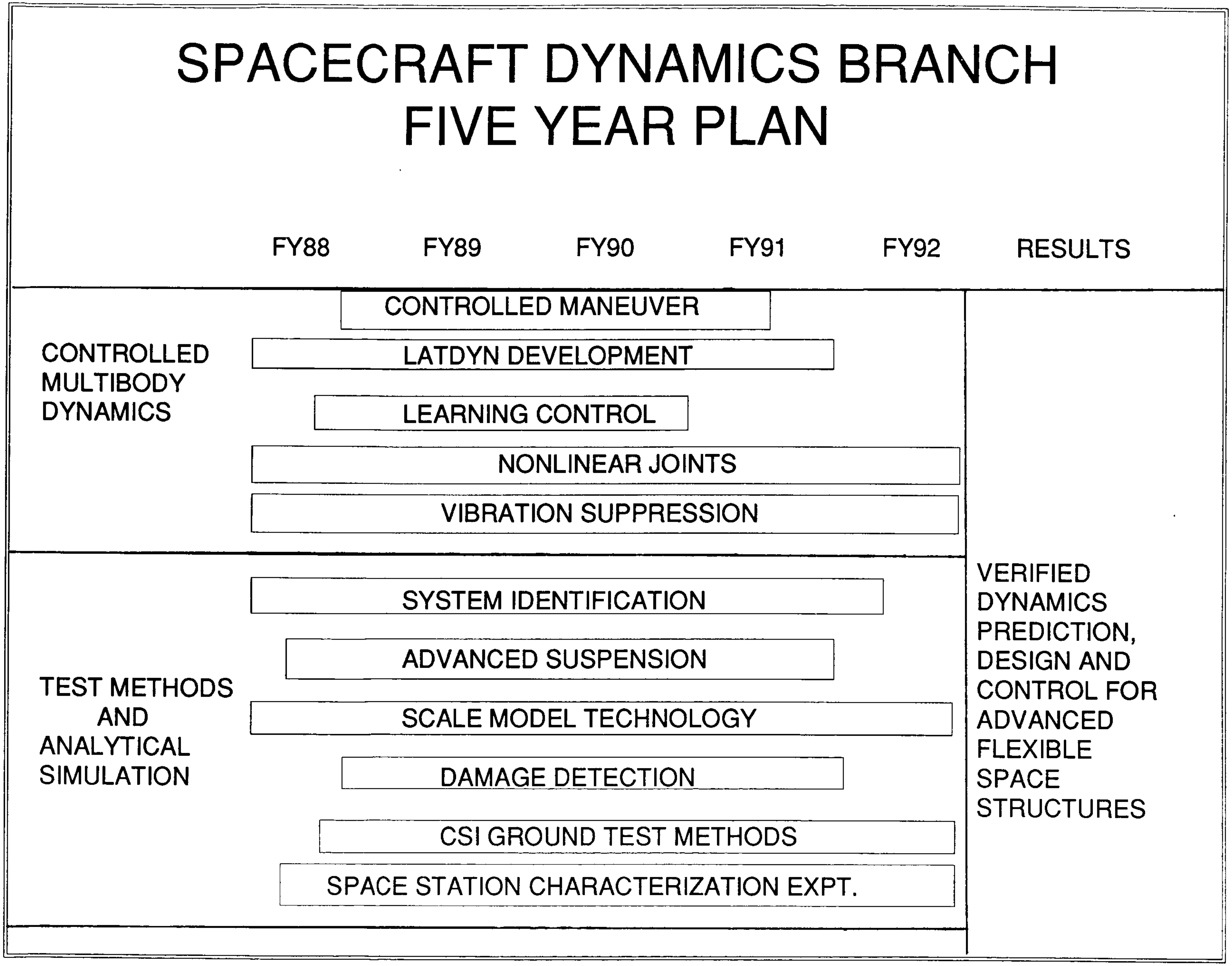

SPACECRAFT DYNAMICS BRANCH
FIVE YEAR PLAN
FY88
FY89
FY90
FY91
FY92
RESULTS
CONTROLLED MULTIBODY DYNAMICS
CONTROLLED MANEUVER
LATDYN DEVELOPMENT
LEARNING CONTROL
NONLINEAR JOINTS
VIBRATION SUPPRESSION
TEST METHODS AND ANALYTICAL SIMULATION
SYSTEM IDENTIFICATION
ADVANCED SUSPENSION
SCALE MODEL TECHNOLOGY
DAMAGE DETECTION
CSI GROUND TEST METHODS
SPACE STATION CHARACTERIZATION EXPT.
VERIFIED DYNAMICS PREDICTION, DESIGN AND CONTROL FOR ADVANCED FLEXIBLE SPACE STRUCTURES

Figure 64.

RESULTS OF 20 METER MINI-MAST GROUND TEST PROGRAM

Richard S. Pappa
Spacecraft Dynamics Branch

RTOP 585-01-61

Research Objective - The objective of this research is to develop analytical and experimental techniques for structural and dynamic characterization of large space structures using representative ground test articles.

Approach - Because of their high efficiency, truss beams will be used as the framework for many future large space structures, such as Space Station Freedom. Accurate analytical models of the truss-beam elements are required to ensure acceptable structural and dynamic characteristics of the overall system, including avoidance of adverse interaction with the spacecraft's control system. To examine potential problem areas in developing and verifying these models, and 18-bay, 20-meter-long, deployable/retractable truss beam was designed and built, and is being used at LaRC as a ground testbed (figure 65(b)). The structure is referred to as "Mini-Mast" because it is a shortened version--one-third the length-- of the truss beam planned to be used in the now-cancelled COFS-I MAST flight experiment. It was manufactured to flight-quality specifications by Astro Aerospace Corporation (Carpinteria, California) using high-modulus graphite/epoxy tubes for all strut members and precision titanium parts for all hinges. A statically-determinant design (3 longerons) with mid-diagonal hinges was selected.

Accomplishment Description - Baseline static and dynamic tests of Mini-Mast in its initial testing configuration (cantilevered vertically from the floor with a 70-lb tip plate) have been completed. Static tests were performed under both bending and torsional tip loads. Overall, the static results correlated well with analytical predictions (within 10%). In torsion, however, significant friction and deadband were observed. These effects are attributed primarily to localized bending and sliding of the diagonal members at the end-fittings. Dynamic (modal) tests were conducted using two shakers located at Bay 10. A typical frequency response function measured in these tests is shown in the figure. The identified modes up to 4th bending, at approximately 50 Hz, agree well with predictions. The cluster of modes near 20 Hz, resulting from 1st bending of the diagonal members, is also well-predicted by analysis. The precise mode shapes in this frequency range have not yet been identified, however, because individual truss members were not instrumented. A second cluster of modes, near 70 Hz, is attributed to 2nd-bending of the diagonal members, but this behavior is still unconfirmed. Consistent with the observed nonlinear static torsion results, the dampong of the torsional modes increased significantly with increased dynamic force level. Damping ratios (i.e., modal damping as a fraction of critical damping) of approximately 0.5% at low force levels increased to 3% - 5% at higher levels. Several test challenges were encountered, including the identification of repeated eigenvalues (all bending modes), highly sensitivity of bending direction with slight asymmetries, and inconvenient access to the test article because of its vertical orientation.

Significance - This project is one of the first combined analytical/experimental programs conducted using a flight-quality, representative large space structure. In-house capabilities, both analytical and experimental, were exercised significantly and improved as a result of the work. These resources and skills will help ensure the development of accurate analytical models in future flight programs.

Future Plans - Mini-Mast will serve as the first ground testbed at LaRC for the Controls-Structures Interaction (CSI) program. Torque actuators will be mounted at the tip, and used to evaluate the performance of active-vibration-control techniques. Eight CSI Guest Investigators (5 Universities, 3 Aerospace Companies), as well as individuals from DFVLR (W. Germany), will participate over the next two years in additional research activities with Mini-Mast.

Figure 65 (a).

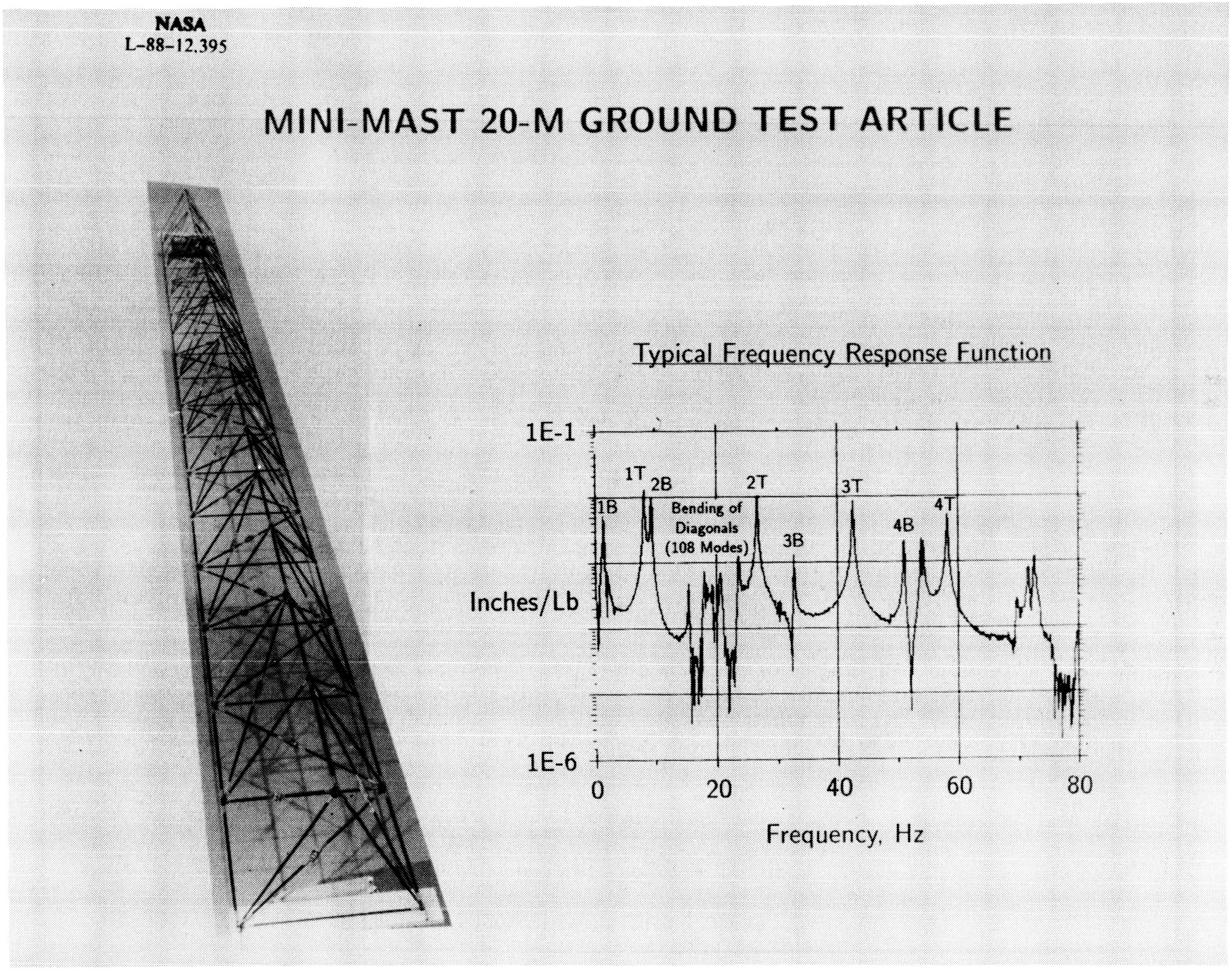

NASA
L-88-12,395
MINI-MAST 20-M GROUND TEST ARTICLE
Typical Frequency Response Function
1E-1
Inches/Lb
1E-6
1B
1T
2B
Bending of
Diagonals
(108 Modes)
2T
3B
3T
4B
4T
0
20
40
60
80
Frequency, Hz

Figure 65 (b).

NONLINEAR ANALYSIS CORRECTS CONVENTIONAL SLEWING PREDICTIONS

Paul E. McGowan and Jerrold M. Housner
Spacecraft Dynamics Branch

RTOP 585-01-51

Research Objective - The objective of this research is to examine the nonlinear effects involved in the dynamic analysis of slewing flexible booms.

Approach - A fundamental dynamic slewing analysis is used to investigate the effects of various nonlinear terms arising from different analysis assumptions. Nonlinear equations were formulated for the planar slewing of flexible booms elastically connected to a rigid body through rotational springs. Numerical solutions to these equations including various levels of nonlinearities were compared to a converged solution from a large angle, nonlinear multi-body dynamics code (LATDYN).

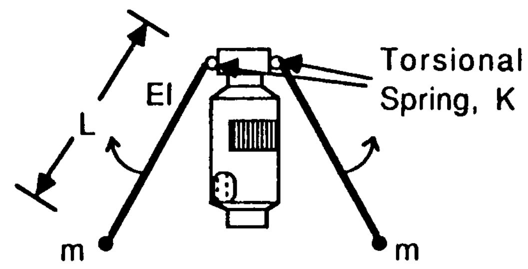
Accomplishment Description - The conventional approach to dynamic slewing analysis has been to use a precise nonlinear kinematics model in along with an imprecise linear deformation model, usually given by a summation of linear modes. As shown on the left of figure 66(b), this imbalance of nonlinear and linear modeling can lead to predictions of erroneous behavior. The imprecise deformation (linear) model, forces the boom tip to move normal to the undeformed boom shape which is then interpreted by the precise kinematics as constituting an increase in the boom's radial arm distance from the center of rotation. In turn, the increased arm results in a Coriolis force acting in the direction of motion (non-restoring) and thus the ingredients for predicting a destablized condition are set in place. The right side of the figure illustrates correct and incorrect predictions in lock-up time as the flexible deploying booms are simultaneously slewed until the boom root rotates through 90 degrees. The linear deformation model erroneously predicts a dramatic increase in lock-up time at a certain value of a nondimensional flexibility parameter. However, a converged balanced analysis reveals the physically expected decrease in lock-up time as the boom is made more flexible or the stiffness of the driving root rotational springs is increased. A rigid boom solution is also shown for comparison purposes.

Significance - The accurate analysis of slewing of flexible booms is critical to on-orbit construction (robotics), payload pointing, and deployment. As spacecraft, their components and appendages become more flexible or on-orbit operations are performed faster, accurate modeling procedures become more critical to performance, stability, and active control. Several existing research and commercialized computer codes do not presently have the required accuracy for these applications.

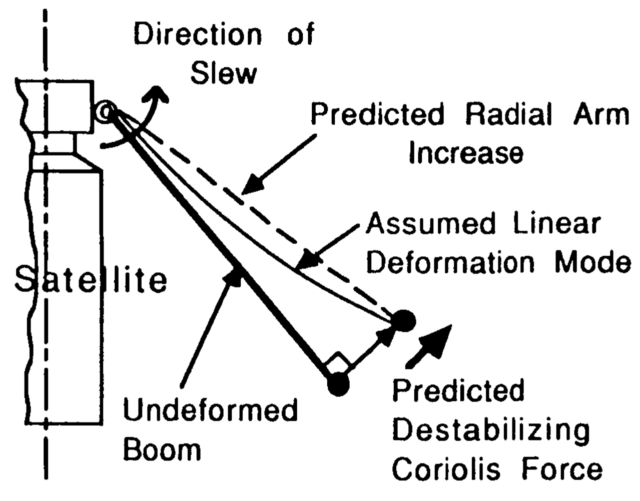
Future Plans - Utilize this example as a benchmark for future analytical developments and extend to active vibration suppression of slewing booms.

Figure 66 (a).

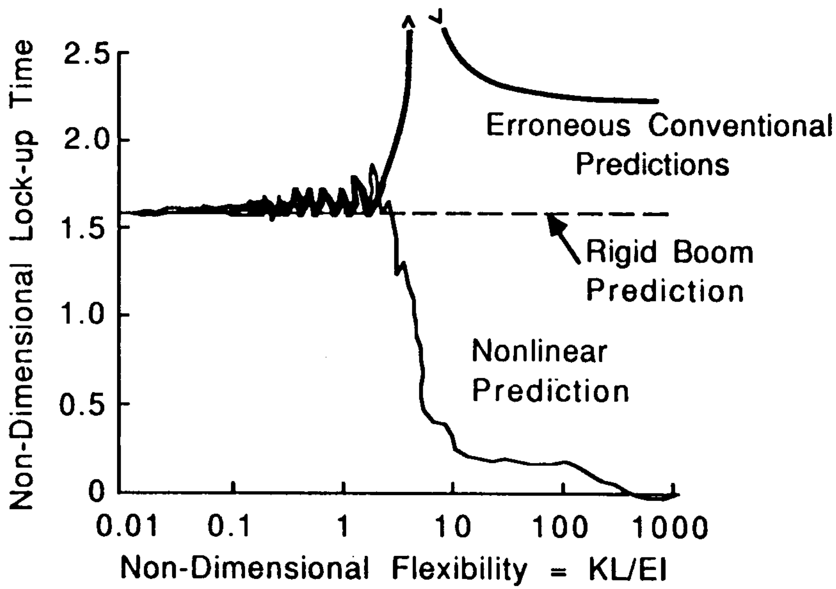
NONLINEAR ANALYSIS CORRECTS CONVENTIONAL SLEWING PREDICTIONS



Spring Drives Boom Root To Horizontal Lockup



Conventional Analysis Predicts Destabilizing Coriolis Force



Flexibility Effect On Boom Root Lock-up

Figure 66 (b).

STRUCTURAL TAILORING OBJECTIVE IDENTIFIED FOR MINIMIZATION OF CONTROLLER ENERGY IN ACTIVE STRUCTURES

W. Keith Belvin and K. C. Park
Spacecraft Dynamics Branch and University of Colorado

RTOP 585-01-21

Research Objective - Performance requirements for future Spacecraft necessitate the use of active controls to augment vibration damping. Unlike conventional control system design, active structure design allows the freedom of changing both the controller and the structure(plant). The objective of this research is to develop physical insight into the structural tailoring process which will maximize the system efficiency. Namely, identification of a tailoring objective which would meet performance requirements while minimizing the expenditure of controller energy was sought.

Approach - An interdisciplinary design was conducted by embedding structural weighting matrices into a conventional multivariable control synthesis method known as Linear Quadratic Regulator (LQR) design. Embedding of the structural matrices into the LQR control design enabled the closed loop cost functions to be expressed in terms of only structural quantities and two scalar gains. Hence, tailoring of the structure implicity tailors the control law. This permits active structure design to be performed using only structural optimization.

Accomplishment Description - Independent Modal Space Control (IMSC), whereby the number of actuators equals the number of controlled modes, was used to derive a closed from solution which minimized an energy cost function. The closed form solution provided substantial physical insight into the optimal tailoring objective. For minimization of the structures kinetic and potential energy and the work of the controller, the appropriate tailoring objective is to maximize the modal stiffness of the structure. Figure 67(b) shows a 54 bay truss-beam excited by external forces. A seven mode controller was designed for the nominal structure and for a tailored design which maximized the modal stiffness while maintaining the same weight as the original design. The lower left portion of the figure shows the design variables (outside tube diameters) for the nominal and tailored structure. The closed loop performance of both designs met the performance requirement of attenuating the vibration amplitude to a threshold level within 10 seconds after the controller was activated. However, the tailored design consumed 56 percent less energy than the nominal design.

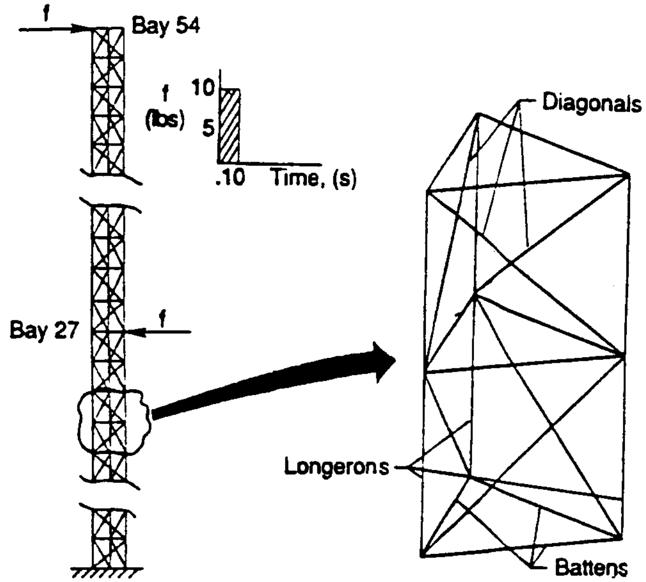
Significance - Physical insight provided by the closed form solution for the control law enables derivation of an optimization objective which is simple enough to be applied early in the design process.

Future Plans - Studies with more general control laws are planned to obtain physical insight into the proper choice of objectives and constraints for integrated design of controls and structures.

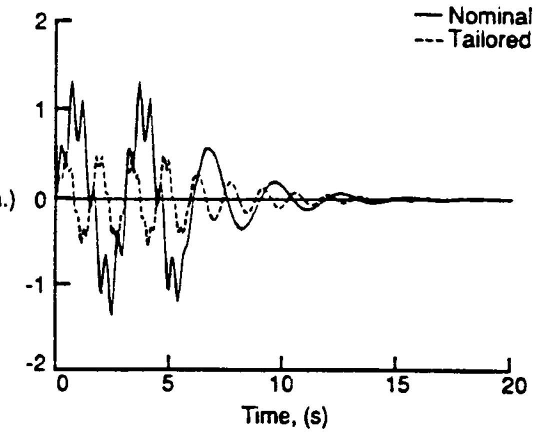
Figure 67 (a).

STRUCTURAL TAILORING OBJECTIVE IDENTIFIED FOR MINIMIZATION OF CONTROLLER ENERGY IN ACTIVE STRUCTURES

TRUSS BEAM



CLOSED-LOOP RESPONSE



DESIGN VARIABLES

Outside Tube Diameter (in.)	Nominal	Tailored
longeron	0.789	1.717
diagonal	1.707	1.284
batten	0.918	0.640

ACTUATOR WORK

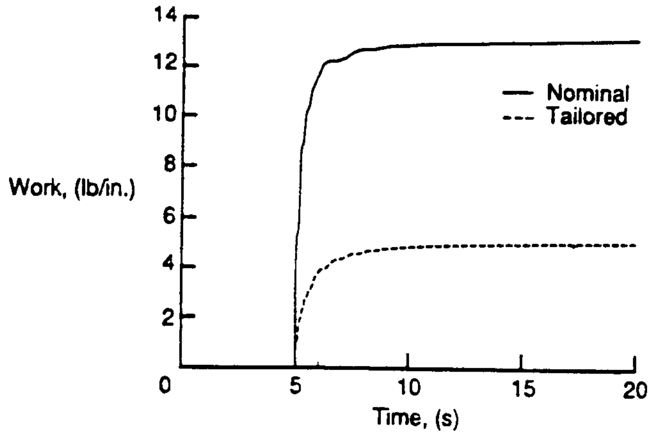


Figure 67 (b).

IMAT CONTRIBUTIONS TO SPACE STATION LEVEL 2 NOVEMBER REFERENCE DATA BOOK

Paul A. Cooper
Spacecraft Dynamics Branch

RTOP 483-32-33

Research Objective - The objective was to support the space station system engineering and integration effort by developing and analyzing structural and attitude control models of several build sequence configurations of the space station

Approach - The IMAT office developed finite-element models of the November Reference Phase 1 configuration (see the left side of figure 68(b)) and several intermediate build sequence configurations supplied by Space Station Level 2.

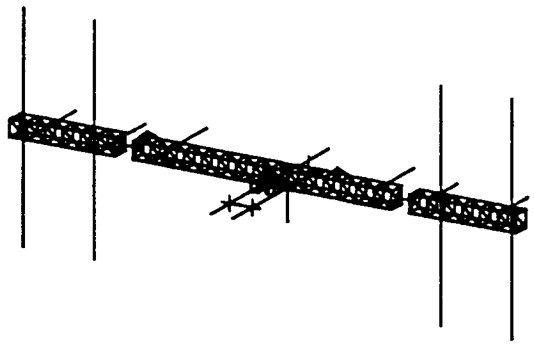
Accomplishment Description - The models were used to investigate the expected low frequency characteristics of the space station (see right side and bottom of figure 68(b)) and to investigate possible structural response interaction with attitude control systems. A simple proportional - derivative control system was developed with gains set to provide rigid body performance characteristics desired by the space station program. A basic lag compensation system was added to the control circuit to provide stability margins sufficient to allow for variations in predicted structural parameters. Saturation amplifiers which limited maximum control torques provided by the controllers (control moment gyros) were also added to the control circuits. A series of simulated disturbances and maneuvers, both open-loop with the controls inactive and closed-loop with the attitude controls responding, were investigated by IMAT personnel. Results from some of these studies and frequency response studies were used by Level 2 in their Engineering Data Book distributed to industry.

Significance - Since the major work package contractors were not funded during FY 88 to develop structural models of the station, the models developed at LaRC were used as baseline models by contractors and the various work package centers. The models were used by Lewis Research Center, Johnson Space Center, TRW, Ford Aerospace, McDonnell Douglas, Draper Labs, and Grumman. The computers and control codes in the IMAT complex were available to Grumman, the support contractor for Level 2, during their evaluation of computer codes.

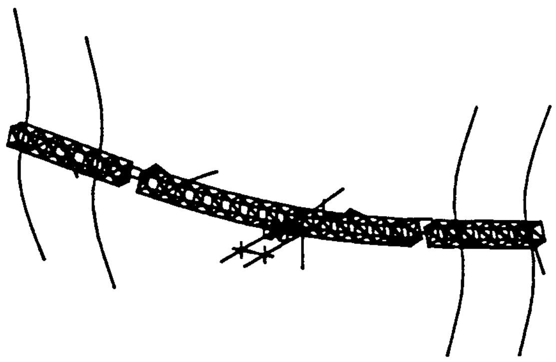
Future Plans - Updated models prepared by the space station contractors for Level 2 will be obtained. and structures/control interaction with a basic attitude control system and a nonlinear reboost procedure will be evaluated.

Figure 68 (a).

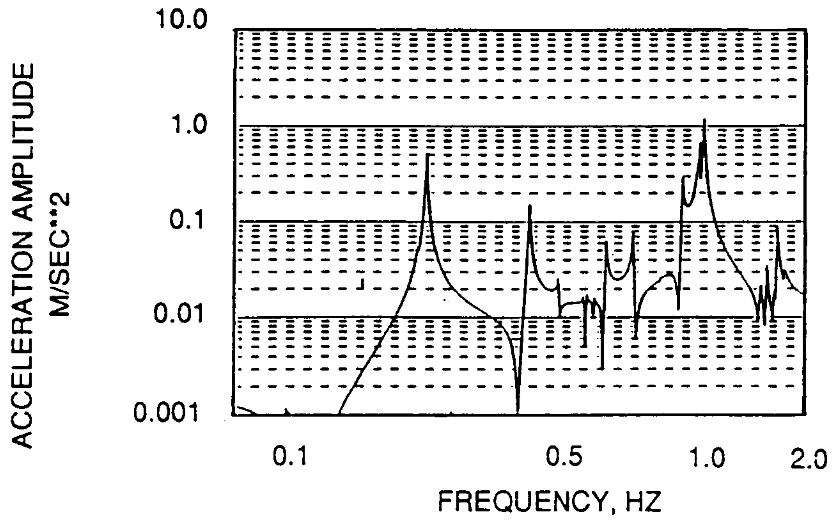
IMAT CONTRIBUTIONS TO SPACE STATION LEVEL 2
NOVEMBER REFERENCE ENGINEERING DATA BOOK



FINITE ELEMENT MODEL



FUNDAMENTAL FRAMEWORK MODE



RESPONSE OF STBD TRANSVERSE BOOM TIP
TO HARMONIC EXCITATION AT RCS LOCATIONS

Figure 68 (b).

COMPLETION OF INTEGRATED MULTIDISCIPLINARY RESEARCH TOOL (IMAT)

Paul A. Cooper
Spacecraft Dynamics Branch

RTOP 585-01-31

Research Objective - The object of the IMAT effort was to provide researchers and analysts with an efficient capability to analyze satellite control systems influenced by the satellite structural dynamics.

Approach - IMAT is a system of computer codes which links a relational database to input and output data of commercial structural and controls analysis codes (See figure 69(b)). The system, using a menu-driven interactive executive program, provides procedures for storage, retrieval, transfer, and inspection of engineering data.

Accomplishment Description - Computer processors were developed which transfer information among computer-aided-design codes, finite-element modeling codes, finite-element analysis codes, and control design and analysis codes using a common database and a common graphics interface (See figure 69(c)). The system now operates totally in the DEC VAX/VMS computer environment. Several MicroVAX computers dedicated to the IMAT system are in place and available to the researchers at Langley.

Significance - The system of codes and computer resources provide the Langley researchers and analysts with state-of-the-art codes to model and evaluate the structural dynamics of satellites, design closed-loop control of satellites and perform simulations of the closed-loop dynamic response of systems to external disturbances or commanded maneuvers. The system of computers and codes has been providing support to approximately fifty researchers per month.

Future Plans - The system of codes will be maintained and both signal analysis and system dynamics capability will be added to IMAT. An additional VAX computer will be added to the cluster of computers to provide a dedicated controls analysis and simulation capability. Plans are under way to couple the interactive capabilities in IMAT with the high speed number crunching capabilities of the CONVEX and CRAY2 supercomputers at Langley.

Figure 69 (a).

SCHEMATIC OF IMAT

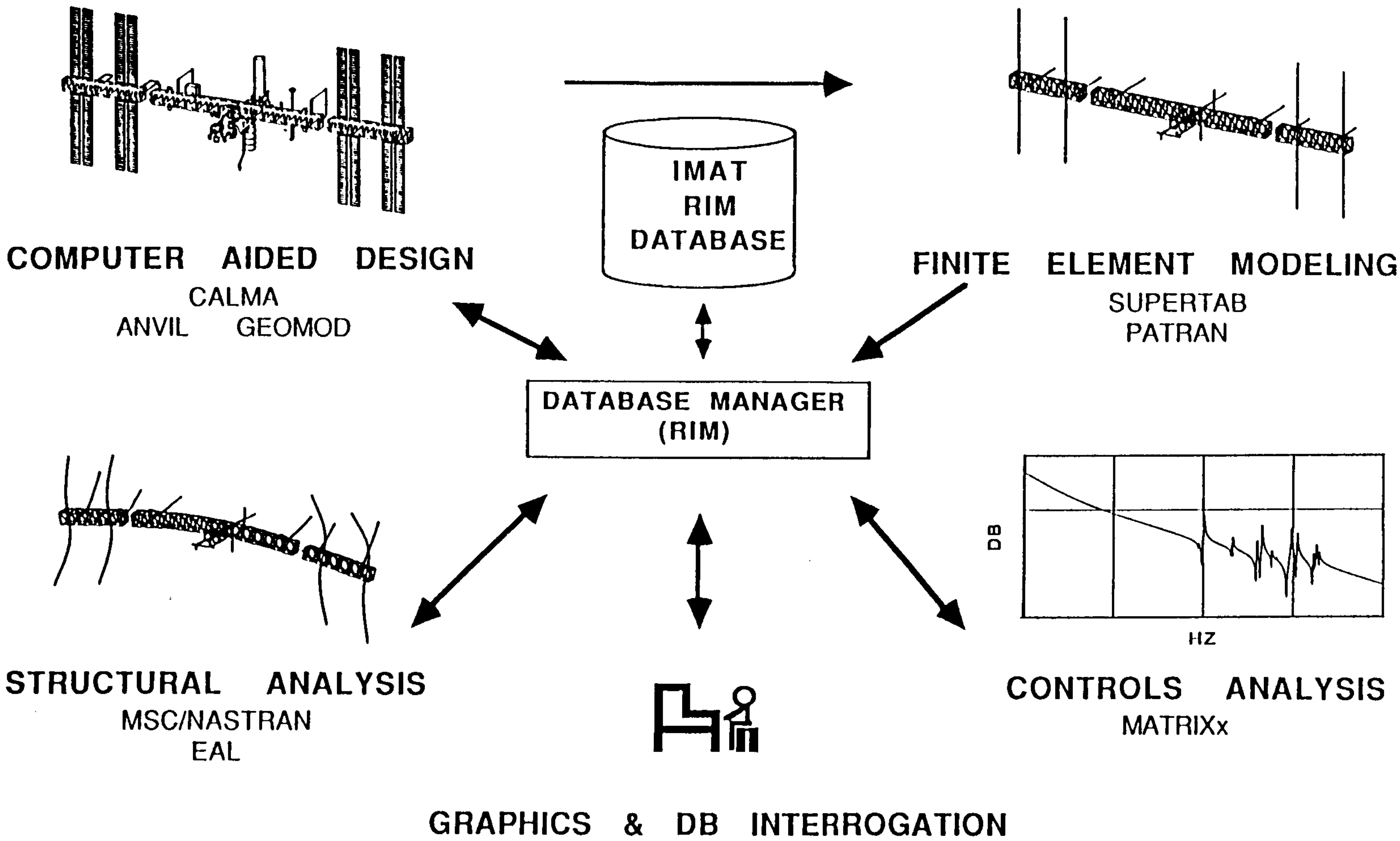


Figure 69 (b).

STATUS OF IMAT RESEARCH TOOL

- o COMPLETED CDC- AND VAX-BASED VERSIONS

- o YEARLY MAINTENANCE AND UPGRADE AT $100K

- o CLUSTER OF THREE VAXES AVAILABLE TO SUPPORT LaRC RESEARCHERS WITH COMMERCIAL CODES:

  MSC/NASTRAN .... FINITE ELEMENT ANALYSIS
  SUPERTAB .... FINITE ELEMENT MODELLING
  MATRIXx .... CONTROLS ANALYSIS & SIM.
  GEOMOD .... SOLID MODELLING
  DI-3000/PICSURE........GRAPHICS

- o COMPUTATIONAL FACILITY WITH COMMERCIAL CODES PROVIDES DIRECT SUPPORT TO SSSCE, DSMT AND CSI

- o HEAVY USAGE
  66 DIFFERENT USERS FOR MONTH OF OCTOBER '88
  A TOTAL OF 4,615 INTERACTIVE AND BATCH LOGINS

Figure 69 (c).

HYBRID SCALING LAWS DEVELOPED AND VALIDATED FOR DYNAMICALLY SCALED SPACE STATION MODEL

Paul E. McGowan and Marc J. Gronet
Spacecraft Dynamics Branch & Lockheed Missiles and Space Co.

RTOP 585-01-31

Research Objective - The objective of this research is to establish a validated design capability for scale models of large space structures that can be tested in existing facilities and which employ realistic hardware to provide low frequency dynamic characteristics.

Approach - Fundamental analyses were performed to study the use of hybrid scaling to provide a dynamically scaled model of Space Station. Hybrid scaling employs classical distorted scaling techniques. It has historically been used extensively for wind tunnel models, when used for large space structures it permits the use of different scale factors for the truss structure components, appendages and payloads while sacrificing local dynamic behavior, but retaining overall global dynamics behavior. In this study, the hybrid-scale model truss structure bay size and truss joint components were selected to be 1/10 and 1/5 scale, respectively (See figure 70(b)). This results in a model which is small enough (50' x 30' planform) to be assembled and tested in an existing LaRC facility. Furthermore, the 1/5 scale joints are essentially the minimum size at which erectable joints can be fabricated without incurring large manufacturing costs or compromising joint performance. Properly distorting the truss component stiffnesses and masses yields a model which possesses the same global dynamic properties as would a fully 1/5 scale replica model.

Accomplishment Description - Hybrid scaling laws were developed and applied to the design of a Space Station scale model. All Space Station components were scaled, and a finite element model of the hybrid-scale design was developed. Emphasis was placed on preserving the dominant strain energy content of each component vibrating within a given mode. In this fashion, truss members were scaled to preserve axial strain energy, whereas appendages and structural interconnects were scaled to preserve bending strain energy in the lower right of the figure shows that the resulting frequency errors for the hybrid-scale model were all less than 1% for the first 10 system vibration modes. In this context, a system mode is defined as a mode in which the dominant strain energy resides in the truss structure.

Significance - Hybrid scaling is a high potential approach for developing a dynamically scaled Space Station model such that a realistic test article can be obtained for developing test and suspension techniques required for verifying analytical models. This model should also provide a test-bed to support the verification of Space Station on-orbit dynamics.

Future Plans - Complete the design of hybrid-scale Space Station components using the current baseline Space Station design. Fabricate, assemble and test the scale model in various configurations to verify analytical predictions.

Figure 70 (a).

HYBRID SCALING LAWS DEVELOPED AND VALIDATED FOR DYNAMICALLY SCALED SPACE STATION MODEL

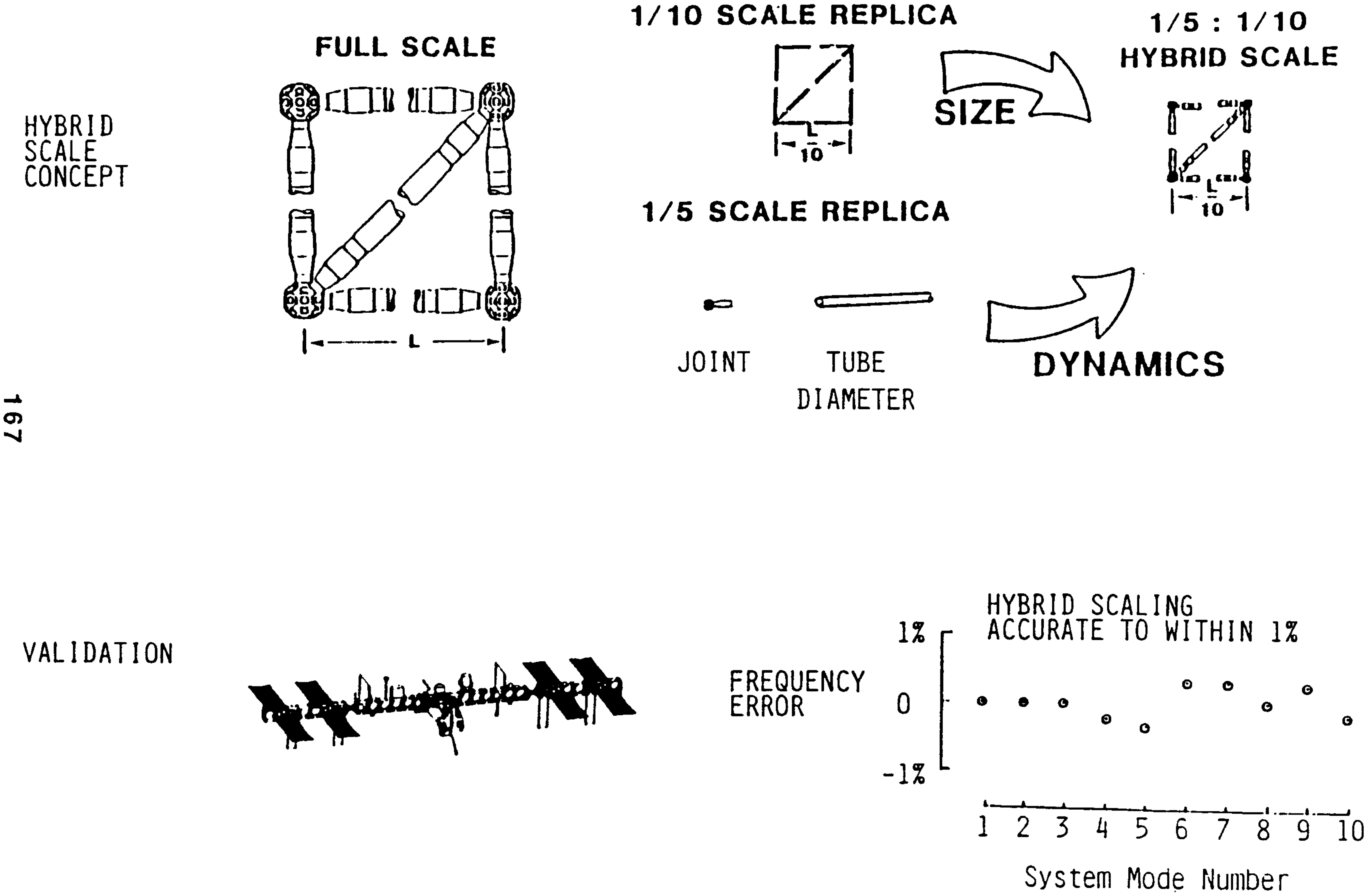


Figure 70 (b).

TERMINAL CONTROL OF MULTIBODY MANEUVER

Jer-Nan Juang
Spacecraft Dynamics Branch

RTOP 585-01-21

Research Objective - The objective of this research is to develop feedback control laws to perform large angle nonlinear maneuvers for articulated flexible arms.

Approach - A challenging controlled multibody system is defined for which nonlinear control systems are designed and then evaluated based upon analytical simulation and laboratory tests. The defined multibody system consists of a translational cart to which is attached a motorized pair of articulating flexible beams. The beams are connected to one another, end-to-end through a motor. One free end of the pair is connected to the cart through a second motor. The controller for the system is then designed, based on various methods, to meet specified performance requirements on the system state at the termination of the maneuver. This is referred to as terminal control. A laboratory setup for this system is nearly complete.

Accomplishment Description - An illustrative terminal maneuver example, as shown in figure 71(b), has been carried out in which the cart moves 1.5 meters from an initial angular motion. The maneuver is to be accomplished in 4 seconds with beam vibration suppressed at the end of the maneuver. To accomplish this design, the nonlinear dynamics of the articulated flexible manipulator are derived using Lagrange's equations along with classical vibration theory. A transformation matrix is formulated to localize the nonlinearities within the system inertia matrix to locations associated with rigidbody motion. Then a feedback linearization scheme is introduced to linearize the dynamic equations for controller design. Through a pole placement technique, a robust controller design is obtained. Analytical simulations are then carried out, as shown in figure 71(c), to assess the system's performance in meeting the terminal requirements. The design based on this linearization methods works well. As expected, it is easier to suppress vibration on the inboard arm than the outboard arm.

Significance - Multibody maneuver is important to the future space missions. Deployment/assembly of large space structures and robotic operations are areas needing this technology which requires an extension of the linear state-of-the-art controller design capability to the nonlinear maneuver case.

Future Plans - Perform laboratory experiments to validate designs and analytical simulations, and examine/compare other promising nonlinear design methods.

Figure 71 (a).

HARDWARE BEING ASSEMBLED FOR TERMINAL CONTROL OF MULTIBODY MANEUVER

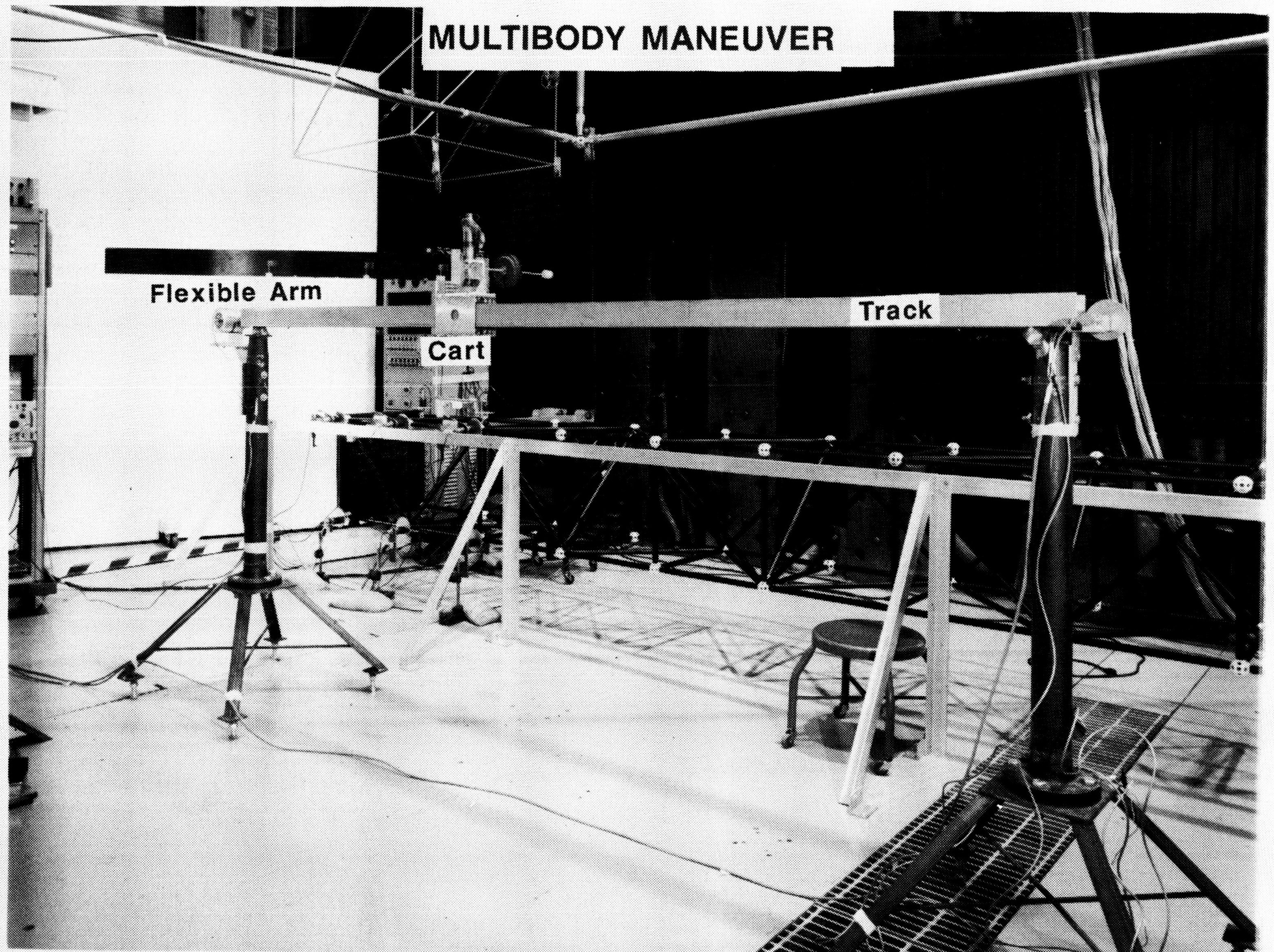


Figure 71 (b).

Terminal Control of Multibody Maneuver

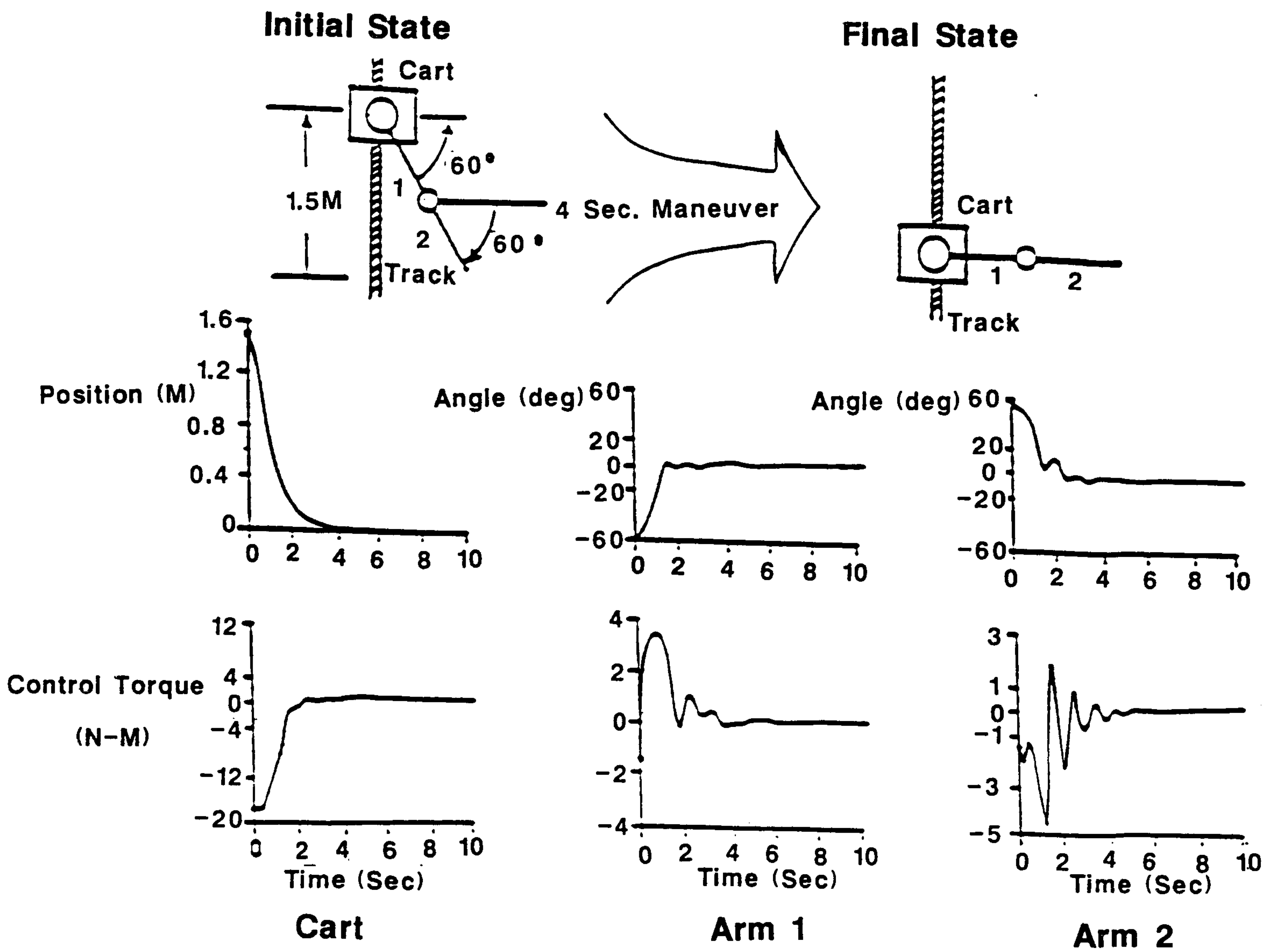


Figure 71 (c).

EFFECT OF ACTIVE TRUSS-BAY LOCATION ON VIBRATION SUPPRESSION

William W. Clark, Harry H. Robertshaw, and Garnett C. Horner
Virginia Polytechnic Institute & State University and Spacecraft Dynamics Branch

RTOP 585-01-61

Research Objective - The objective of this research is to compare the effectiveness of the location of an active truss bay in controlling the vibrations of a cantilevered truss beam.

Approach - A lumped-mass approach was used to model the active bay with physical constants taken from an experimental active bay at VPI&SU. The beam properties and geometry are identical to the mini-mast at NASA LaRC. A basis for comparison was derived and motor constants were chosen to make the comparison. Computer simulations were carried out to determine the response of the two systems. One system had the active bay placed at the base of the beam the other had the active bay at the tip of the beam along with a proof mass (figure 72(b)). Initial condition inputs were used with a full state feedback LQR optimal control law for each case.

Accomplishment Description - Simulations of these linearized models have shown that the active truss bay in the base of the beam has more authority over controlling the beam than the active truss bay at the tip. In addition to a schematic of the active truss bay there are plots of the tip deflection of the beam in response to initial condition inputs for each beam/active bay system (See the figure). As a performance index of vibration attenuation, the time integral of total system energy was calculated during the simulations (note that this is not the same as the LQR performance index). For 160 Joules (J) of strain energy in the initial conditions (this is 20 J of strain energy in each modelled mode) this number was found to be 8.4 J's for the active truss bay proof mass at the beam tip. The base located active truss bay was shown to be particularly effective at controlling first mode vibrations of the beam--a task for which inertial-type actuators are not well suited.

Significance - The concept of active bay actuation via the active truss bay actuator in the base was shown to be a more effective means of controlling vibrations in a flexible beam than locating the active truss bay at the tip with a proof mass.

Future Plans - This analysis will be extended to study the active truss bay actuator's effectiveness in controlling a free-free beam. Three-dimensional active truss bay actuators will also be studied, using linear and nonlinear techniques. The results of these analyses will be compared with results from existing experimental active truss bay set-ups.

Figure 72 (a).

EFFECT OF ACTIVE TRUSS BAY LOCATION ON VIBRATION SUPPRESSION

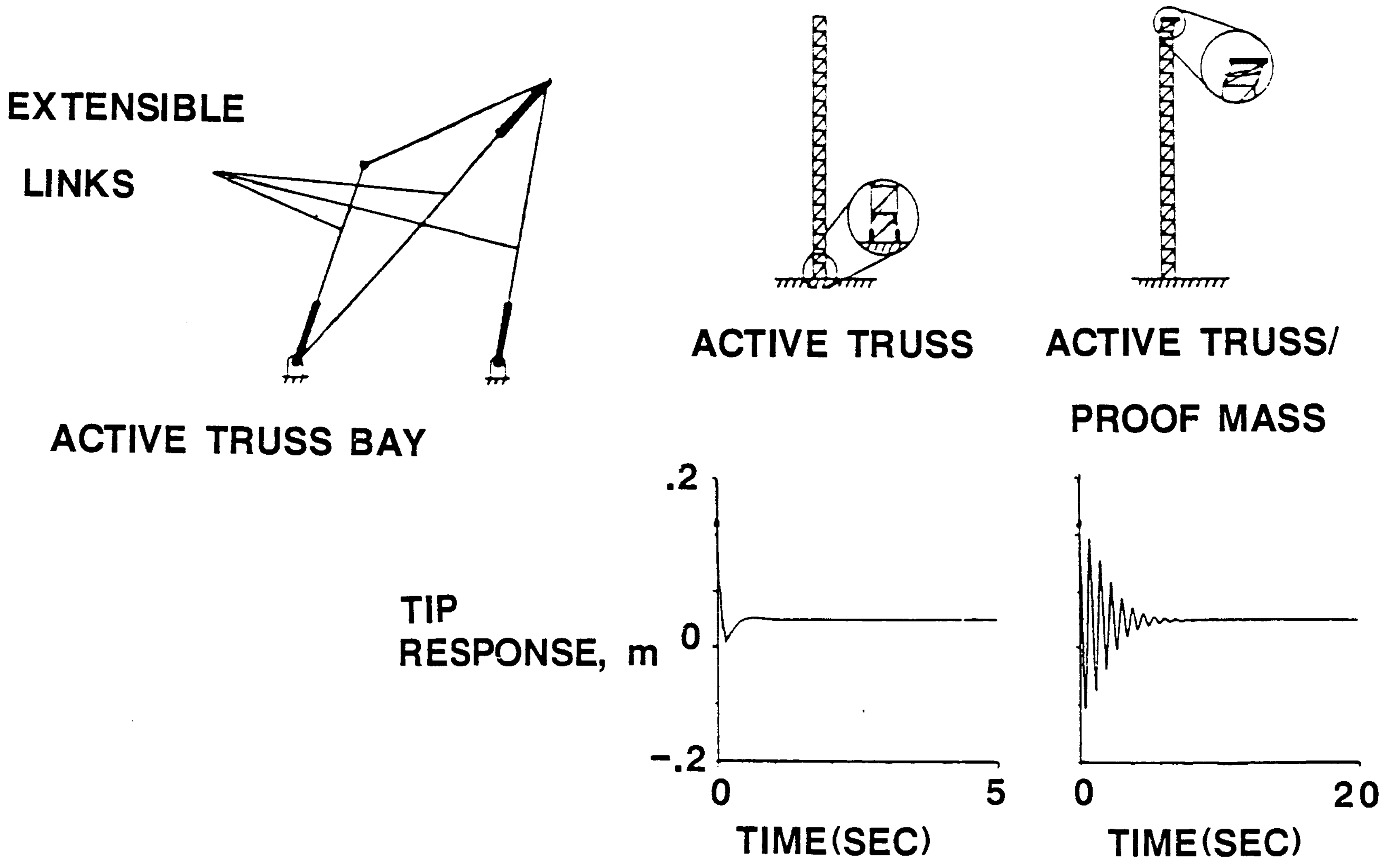


Figure 72 (b).

CONFIGURATION AEROELASTICITY

FY89 PLANS

- Complete TDT Initial Tests of AFW with Active Flutter Suppression System
- Complete Tests of Statically Unstable Cable-Mounted Model in TDT
- Complete Parametric Tests of Growth Blackhawk Rotor Blades in TDT
- Complete Initial Tests of ARES II in Hover Facility
- Complete Study of Extension-Twist-Coupled Non-Circular Composite Tubes
- Complete GVT of Bell ACAP Difficult-Components Airframe

Figure 73.

AIRCRAFT AEROELASTICITY

Maynard C. Sandford
Configuration Aeroelasticity Branch

RTOP 505-63-21

Research Objective - The objectives in the aircraft aeroelasticity technical area are (1) to determine and solve the aeroelastic problems of current designs, and (2) to develop the aeroelastic understanding and prediction capabilities needed to apply new aerodynamic and structural concepts to future flight vehicles.

Approach - The types of research included in the aircraft aeroelasticity area as illustrated in figure 74(b). This research is a combination of experimental and complementary analytical studies. The experimental work focus on the use of the Langley Transonic Dynamics Tunnel (TDT) which is specifically designed to meet the unique needs of aeroelastic testing. On occasion flight research programs are undertaken when it is necessary to simulate important parameters that cannot be accurately accounted for in ground-based facilities. Often the research is a cooperative effort with other government agencies and/or industry.

Status/Plans - Work for the coming year includes a variety of activities. Several will be mentioned here by way of illustration. The Active Flexible Wing (AFW) will be tested in a program incorporating active roll control of a flexible wing with active flutter suppression and load alleviation. Flutter clearance studies will be conducted on a statically unstable cable-mounted research plane with a stability augmentation system. Flutter clearance studies in support of the development of the advanced tactical fighter (ATF) will be initiated. In the long term this work will require several test series in the TDT using two different designs. Continued testing of unmanned launch vehicles will be performed this year with the ground winds loads testing of the Atlas II launch vehicle, launcher, and umbilical tower configuration.

Figure 74 (a).

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OF POOR QUALITY

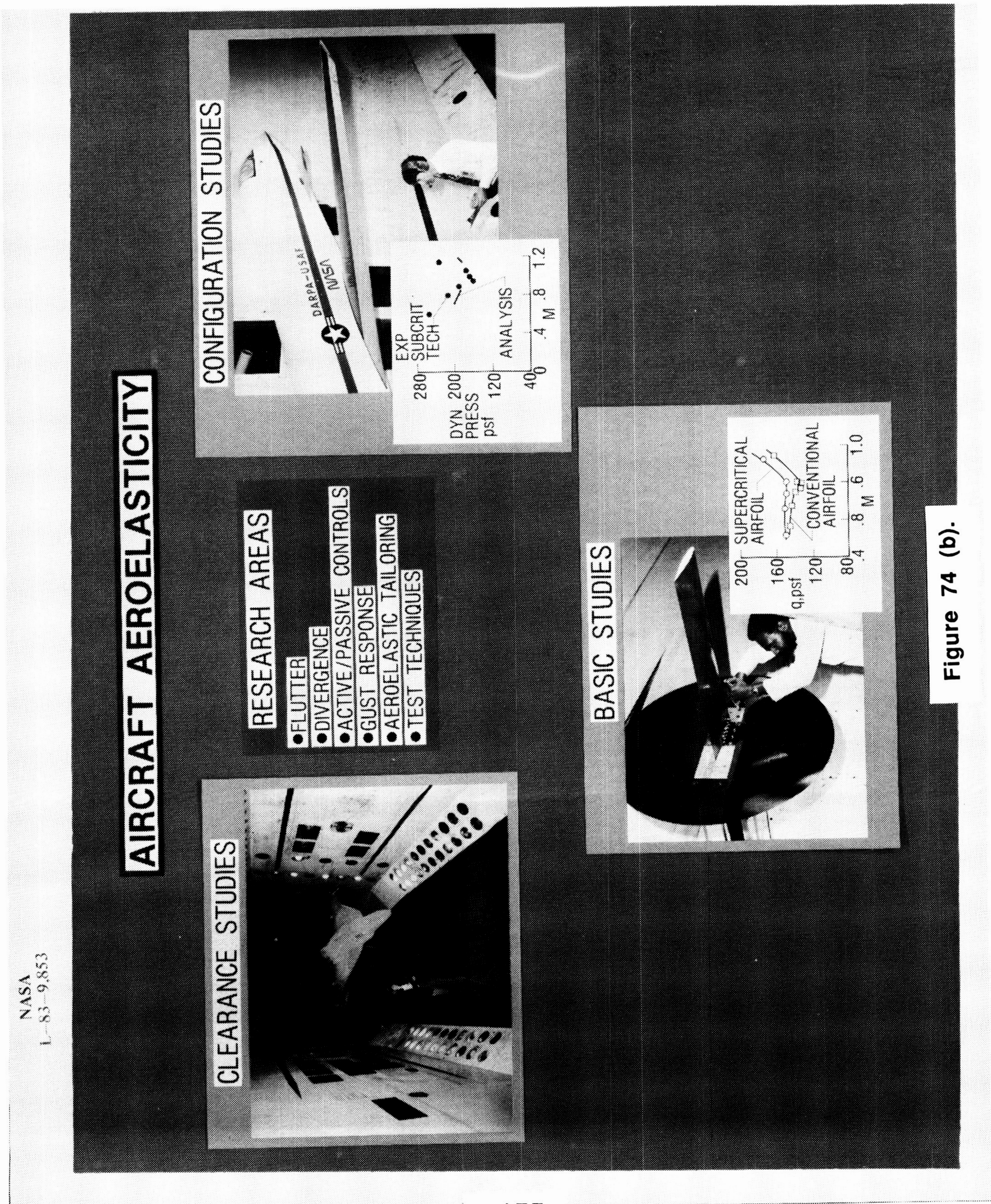


Figure 74 (b).

ROTORCRAFT AEROELASTICITY

William T. Yeager, Jr. (Army)
Configuration Aeroelasticity Branch

RTOP 505-63-51

Research Objective - The objectives in this technical area are to (1) conduct research in the aeroelastic, aerodynamic, and dynamic characteristics of rotors; (2) support design of advanced performance rotorcraft in the areas of loads, vibration, and aeroelastic stability; and (3) develop the experimental and analytical techniques necessary to extend wind tunnel and laboratory capabilities to future research requirements and opportunities.

Approach - This research area is a joint effort of SDyD and the U.S. Army Aerostructures Directorate which is co-located at Langley. The in-house civil service research is supplemented by industry contracts and university grants. The work is a combination of experimental studies conduced in the TDT and the General Rotor Aeroelastic Laboratory (GRAL), and analytical studies that include the application of existing methods for correlation with experimental results and the development of new and improved methods. The Aeroelastic Rotor Experimental System (ARES) is a key test bed in the experimental studies. This system which has drive mechanisms, force balance, and other equipment housed in a generic fuselage shape provides a means for studying a variety of rotor systems in simulated forward flight in the TDT and in hover in the GRAL. Two advanced versions of the ARES have been developed which will make it possible to better model the coupling of the rotor and the body. The first enhanced ARES design, ARES 1.5 mounts the metric section of the existing ARES model on a static gimbal or "soft mount" (see figure 75(b)). This mount will allow the model fixed system stiffness and damping characteristics in both pitch and roll to be adjusted. The second design, ARES II, mounts the metric section of ARES on a platform which is supported by six hydraulic actuators. These actuators will be computer controlled to obtain the desired body roll, pitch, yaw, side, normal and axial motion.

Status/plans -All parts for the ARES 1.5 should be completed by early 1989. The model will be assembled and system frequency and damping characteristics determined. Hover tests will be conducted, and then the model will be used for a test in the TDT. The ARES II will also be assembled in 1989. A wind-tunnel test of a tailored Growth Black Hawk (GBH-T) rotor will be conducted this year. The purpose of this test is to evaluate blade modal shaping as a means of reducing fixed-system vibratory loads.

Figure 75 (a).

NASA
L-88-11,734

505-63-51

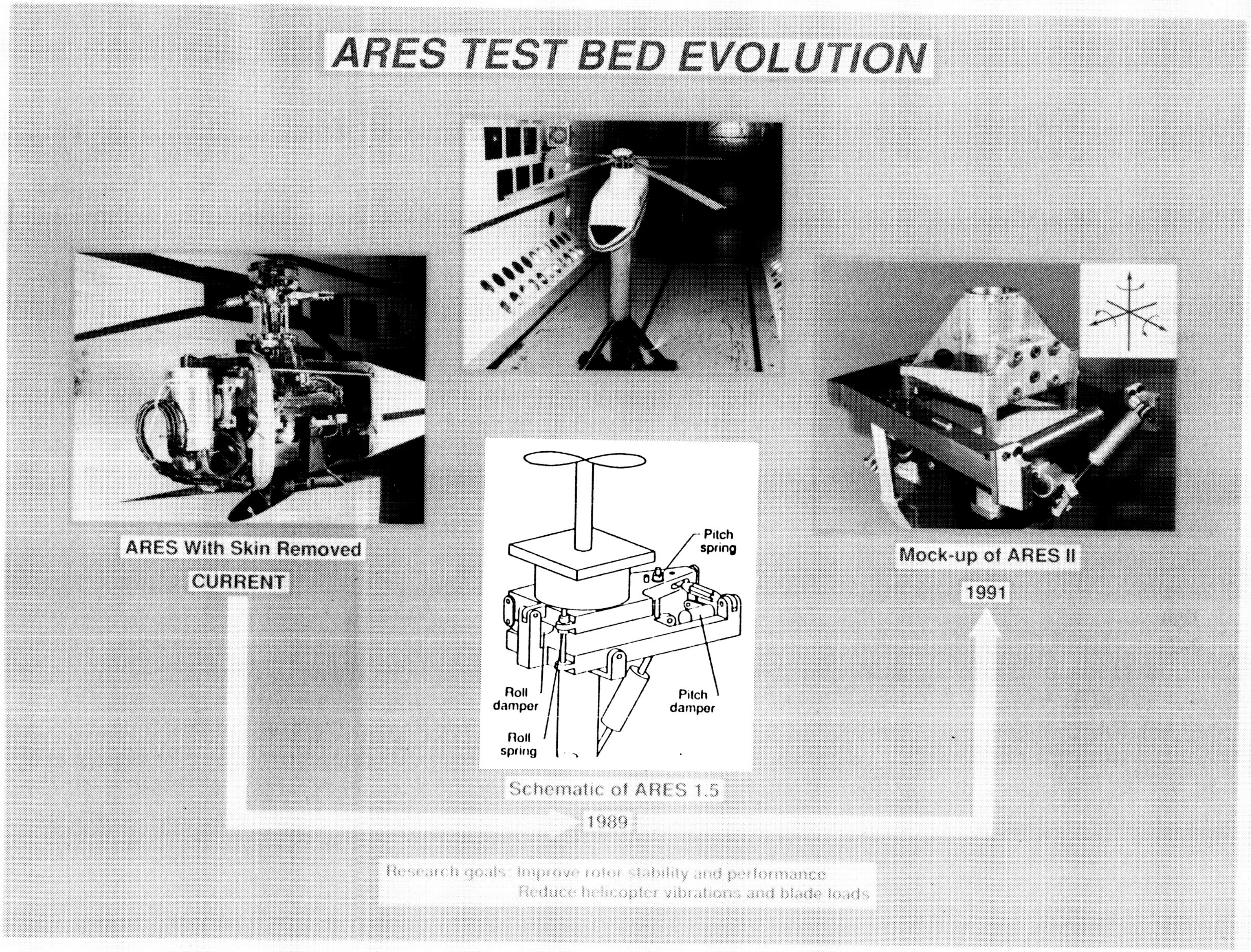


Figure 75 (b).

ROTORCRAFT STRUCTURAL DYNAMICS

Raymond G. Kvaternik
Configuration Aeroelasticity Branch

RTOP 505-63-51

Research Objective - Helicopters are prone to vibrations which can seriously degrade both service life and ride quality. With only a few exceptions, vibration problems have not been identified and attacked until the flight test and operational stages of a new helicopter. There is now a recognized need to account for vibrations during the analytical phases of design. The advent of modern methods of computer analysis has provided the opportunity to achieve such a capability. The objective is to emplace in the United States a superior capability for design analysis of helicopter vibrations.

Approach/Status - The rotorcraft structural dynamics program is commonly described by the acronym DAMVIBS, for Design Analysis Methods for VIBrationS. This research is being accomplished through a combination of efforts that involves the four major U. S. manufacturers of helicopter airframes, Bell Helicopter Textron, Boeing Helicopter Co., McDonnell Douglas Helicopter Co., and Sikorsky Aircraft. The contractor studies are complemented by in-house research and university grants. The industry participants, working under task-type contracts, have and are forming NASTRAN finite-element models of metal and composite airframes with companion ground vibration measurements and correlations, and have carried out coupled rotor-airframe vibrations analysis of a common vehicle. Four annual meetings have been held where the participants reported on the status of their work. Figure 76(b) illustrated the elements receiving current and future emphasis in the program.

Plans - Several areas are mentioned as examples of the worked planned for FY 89. A ground vibration test of the Bell D292 ACAP is planned to be conducted in the vibration test facility at the Army Aviation Applied Technology Directorate, Fort Eustis. This test would be a continuation of the difficult components studies to better identify the components of an airframe that are contributing to the lack of correlation between test and analysis at higher frequencies. In-house static and dynamic testing of non-circular composite tubes employing extension-twist coupling will be performed. Development of better finite elements such as those for describing non-linear characteristics of tapered beams will be continued. Studies on improved representation of damping will also continue.

Figure 76 (a).

D A M V I B S FUTURE EMPHASIS

Airframe structural optimization

Coupled rotor-airframe vibrations

Structural damping

Finite element modeling

Difficult components

Composite rotor blades

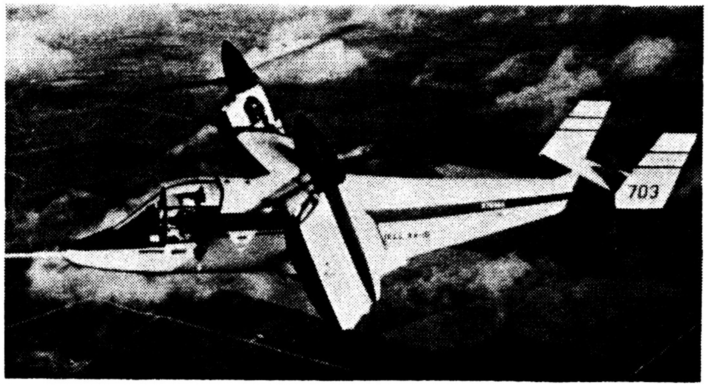


Figure 76 (b).

UNSTEADY AERODYNAMICS

FY 89 PLANS

- CAP-TSD CODE APPLICATION AND SUPPORT
  - CONTINUE APPLICATIONS TO VERIFY RANGE OF ACCURACY: INHOUSE AND COOPERATIVE EFFORTS
  - CONTINUE TO PROVIDE PROGRAMMING SUPPORT
  - DOCUMENTATION OF CODE
- DEVELOP EULER AND NAVIER-STOKES CAPABILITIES FOR AEROELASTIC ANALYSIS
  - STRUCTURED GRID IMPLEMENTATION, CFL2D/3D
  - UNSTRUCTURED GRID IMPLEMENTATION
- DEVELOP CAPABILITY FOR AEROELASTIC ANALYSIS OF VORTEX DOMINATED AND BUFFETING FLOWS

Figure 77.

AEROSERVOELASTICITY BRANCH

FY-89 PLANS

- UNSTEADY TIME-DOMAIN AERODYNAMIC METHODS TO EXPAND EXISTING ASE ANALYSIS CAPABILITIES
- REDUCED-ORDER DYNAMIC CONTROLLER DESIGN BY MULTILEVEL OPTIMIZATION
- INTEGRATED AEROSERVOELASTIC DESIGN BY MULTILEVEL OPTIMIZATION
- MULTI-YEAR JOINT NASA/ROCKWELL AEROSERVOELASTIC WIND TUNNEL PROGRAM UNDERWAY
- REUSABLE FLYBACK BOOSTER AEROSERVOELASTIC CHARACTERISTICS TO BE DETERMINED
- COMPLETE DEVELOPMENT OF AERODYNAMIC CORRECTION FACTOR METHODOLOGY
- DEFINE AEROELASTIC AND ASE CHARACTERISTICS OF "HOT" HYPERSONIC VEHICLES

Figure 78.

UNSTEADY TIME-DOMAIN AERODYNAMIC METHODS TO EXPAND EXISTING ASE ANALYSIS CAPABILITIES

Walter A. Silva (PRC) and Jessica A. Woods (PRC)
Aeroservoelasticity Branch

RTOP 505-63-21

Research Objective - Current aeroservoelastic (ASE) analysis methods are limited to the use of unsteady linear subsonic or supersonic aerodynamic forces and moments computed for harmonic motion in the frequency domain. Modern computational fluid dynamics (CFD) methods compute nonlinear transonic unsteady aerodynamic forces and moments in the time domain. An increasing need for ASE evaluations at transonic speeds requires the integration of nonlinear time domain unsteady transonic aerodynamic forces into existing ASE analysis tools.

Approach - Linear subsonic or supersonic unsteady aerodynamic forces computed for harmonic motion in the frequency domain are modeled using either Rational Function Approximation (RFA) or Minimum State (MS) methods in current ASE studies (See figure 79(b)). These linear models of the unsteady aerodynamics are coupled with the aircraft equations of motion, the structural dynamics equations, and the control system dynamics for stability and performance analyses using state-space techniques. The approach for integrating the time domain aerodynamics into a ASE stability analysis capability involves the development of modeling techniques equivalent to the RFA or MS methods used with frequency domain aerodynamics. To model the highly nonlinear transonic aerodynamic forces, nonlinear modeling techniques such as generalized frequency response methods or Volterra-Wiener series submodels will be used. These nonlinear modeling techniques linked with nonlinear system stability analysis methods will provide a complete transonic ASE stability analysis capability. Linear methods will be developed concurrently to model linear or nearly linear aerodynamic behavior for direct incorporation into existing linear ASE stability analysis tools. The linear method will also be used for later verification of the nonlinear aerodynamic approximations at the subsonic and supersonic boundaries of the transonic region.

Status/Plans - The Computational Aeroelasticity Program - Transonic Small Disturbance (CAP-TSD) code developed by the Unsteady Aerodynamics Branch is currently being used to generate linear and nonlinear aerodynamic forces for flutter analyses. Time domain aerodynamic forces for impulsive motion of rigid-body, structural vibration, and control surface modes will be used to initiate the aerodynamic modeling development efforts.

Figure 79 (a).

UNSTEADY TIME-DOMAIN AERODYNAMIC METHODS TO EXPAND EXISTING ASE ANALYSIS CAPABILITIES

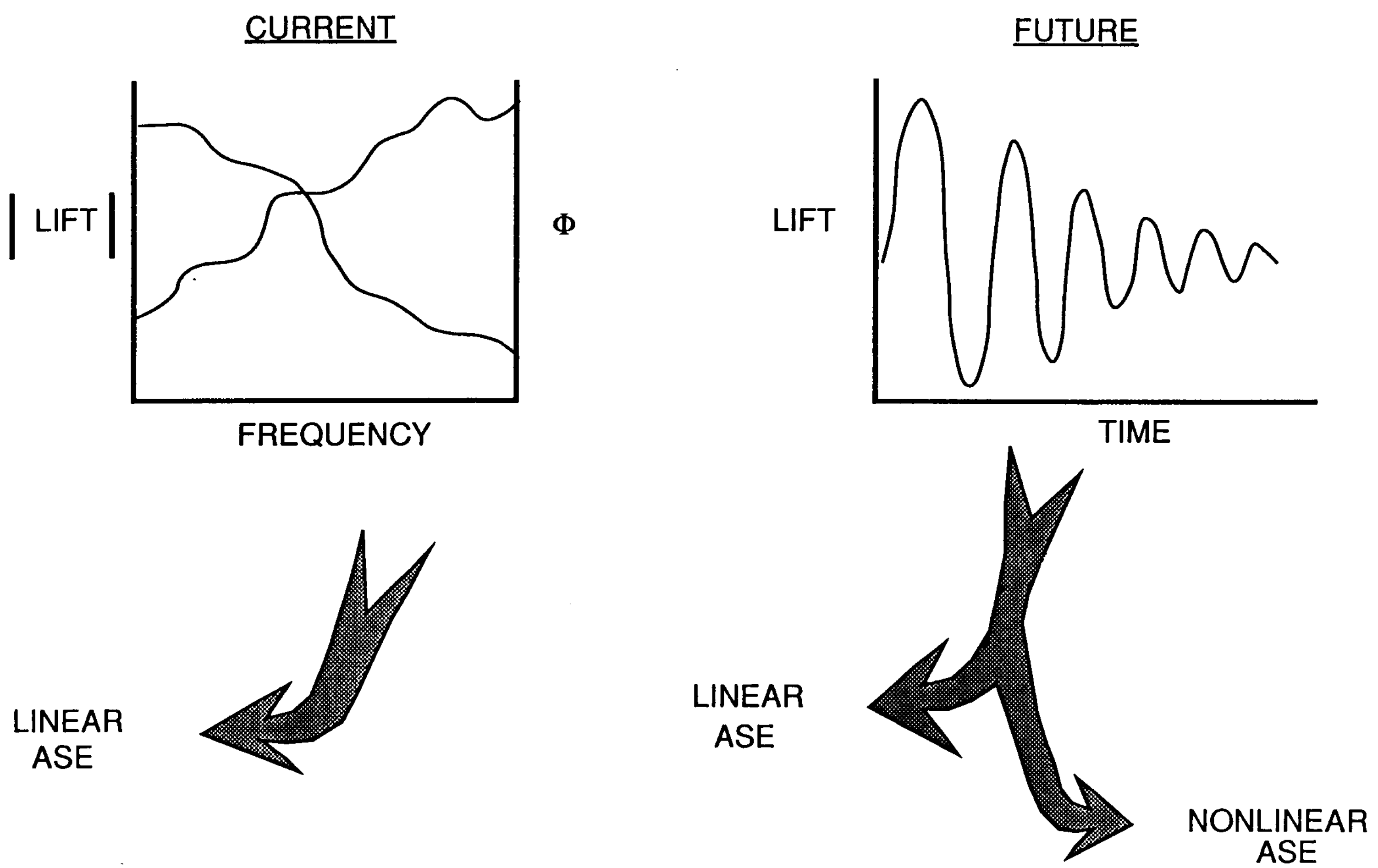


Figure 79 (b).

REDUCED-ORDER DYNAMIC CONTROLLER DESIGN BY MULTILEVEL OPTIMIZATION

Michael G. Gilbert
Aeroservoelasticity Branch

RTOP 505-63-21

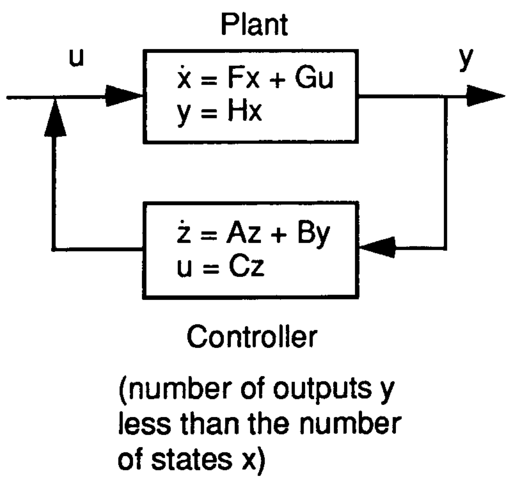
Research Objectives - Standard nonlinear programming (NLP) methods for numerical optimization are often used to design reduced-order dynamic controllers for linear systems. These methods are currently limited by the number of mathematically independent free design variables in the controller model. The objective of this research is to apply hierarchal problem decompositions and mulitlevel optimization techniques to the design of reduced order dynamic controllers to increase the problem design freedom and more easily satisfy multiple design requirements and specifications.

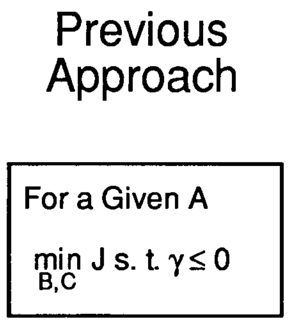
Approach - Current numerical optimization methods for the design of reduced-order dynamic controllers seek to find elements of the controller matrices (shown on the left of figure 80(b)) which cause a cost function J to be minimized, subject to a set of constraints ($\gamma \leq 0$). Only a subset of the elements of the three matrices A, B, or C are mathematically independent and can be freely selected during the optimization. One commonly used subset is all the elements of the B and C matrices. These elements are then selected such that J is a minimum for the given A matrix. Application of hierarchal decomposition and multilevel optimization to the problem (on the right of the figure) removes the mathematical dependence of the elements of the A matrix so that an additional cost function K is optimized, subject to an additional constraint set ($\eta \leq 0$). It is a requirement that the optimized value of the cost function J is at least as good as the previous approach. The sensitivity of the optimized cost function J^* to elements of the A matrix is required for this methodology. This approach provides an increase in the problem formulation and design freedoms, improving current design methods to include additional design objectives and constraints.

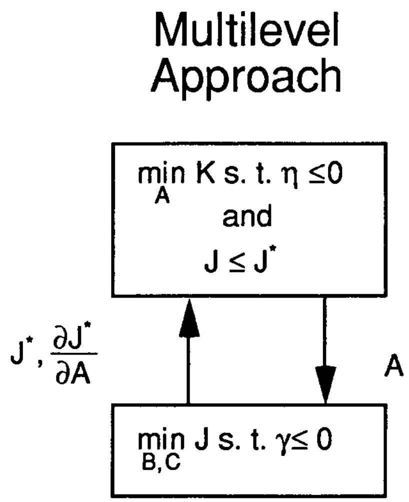
Status/Plans - Analytical sensitivity expressions for the sensitivity of the lower level optimized cost function to elements of the controller matrices are complete. The application of the multilevel approach to the design of a reduced-order controller for the Active Flexible Wing (AFW) wind tunnel model is beginning.

Figure 80 (a).

REDUCED ORDER DYNAMIC CONTROLLER DESIGN BY MULTILEVEL OPTIMIZATION







Multilevel Approach Development:

- Increases problem formulation and design freedoms
- Requires iteration and the sensitivity of J^* to changes in A
- Analytical sensitivity expressions currently being formulated

Figure 80 (b).

INTEGRATED AEROSERVOELASTIC DESIGN BY MULTILEVEL OPTIMIZATION

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RTOP 505-63-21

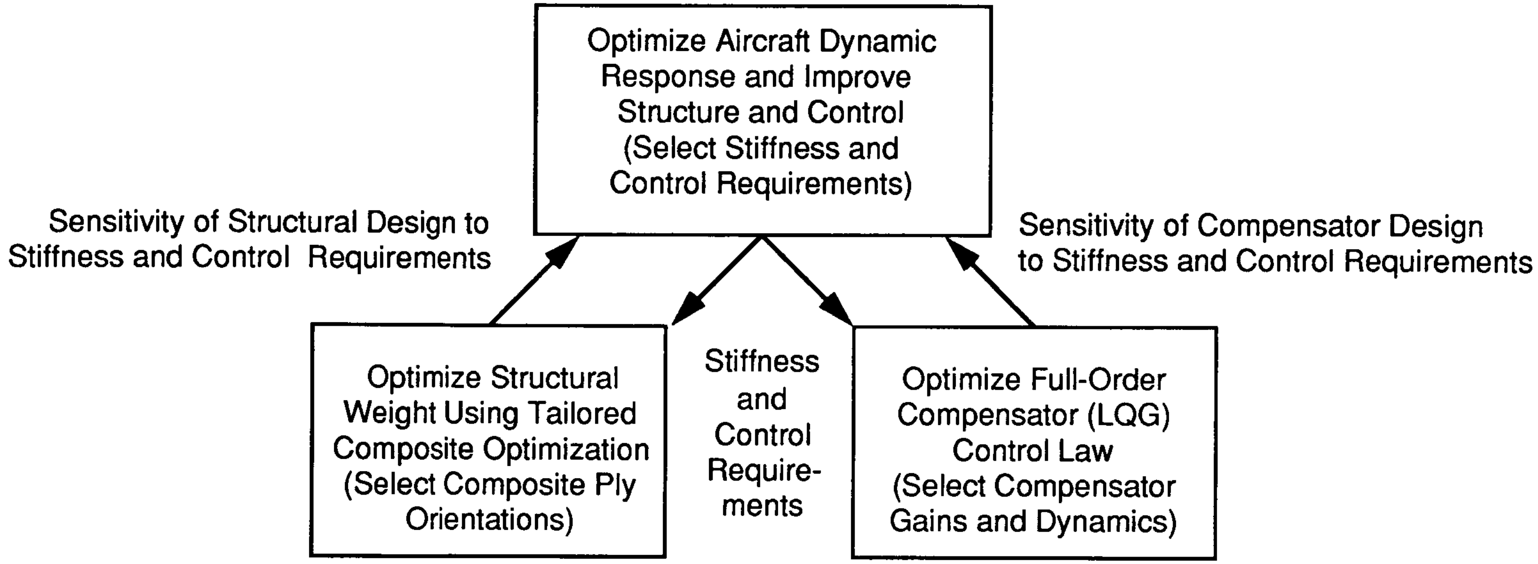
Research Objective - Existing aircraft design methodologies do not take maximum advantage of multidisciplinary interactions. The present research objective is to develop a design methodology that integrates structural design and active control design.

Approach - The integrated aeroservoelastic design methodology is based on hierarchal multilevel problem decomposition and optimization techniques. The aeroservoelastic system is decomposed along disciplinary boundaries into structural and control subsystems, as shown in figure 81(b). At the topmost, integrated level, a measure of vehicle performance or response is optimized with structural and control parameters as design variables. The structural and control subsystems are optimized using design variables and methods suitable to their respective disciplines but with those parameters specified at the top level held constant. Sensitivity derivatives, with respect to the parameters, of these optimum solutions are used in the top level optimization as gradient information. Thus, design decisions are made at the top level with knowledge of their impact on optimal subsystem designs thereby providing a rational means of performing subsystem performance tradeoffs to the benefit of the integrated system.

Status/Plans - Analytical sensitivity expressions for Linear Quadratic Gaussian (LQG) controllers now exist. Development of a module for performing multilevel structural optimization of a wing composed of composite material box beams and computation of sensitivities is in progress. At the top level of the structural decomposition, beam stiffness and inertial properties are optimized to minimize weight while matching modal frequencies and generalized masses that would be specified at the integrated level and satisfying failure constraints from the bottom structural level. At the bottom structural level, skin thicknesses and a fiber orientation are optimized to minimize a cumulative measure of failure under load while matching the beam stiffness and inertial parameters specified at the next higher level (just described). Options for response criteria and modeling at the topmost, integrated level are identified. Work will continue on the structural module in parallel with development of the top level module with single sublevels for the structural subsystem (i.e. the topmost level of the structural decomposition) and control subsystem.

Figure 81 (a).

INTEGRATED AEROSERVOELASTIC DESIGN BY MULTILEVEL OPTIMIZATION



Development Status:

- Analytical Sensitivity Expressions of LQG Controllers Exist
- Module for Structural Optimization and Sensitivity Nearing Completion
- Top Level Dynamic Response Criteria Options Identified

Figure 81 (b).

MULTI-YEAR JOINT NASA/ROCKWELL AEROSERVOELASTIC WIND TUNNEL TEST PROGRAM UNDERWAY

Boyd Perry, III
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RTOP 505-63-21

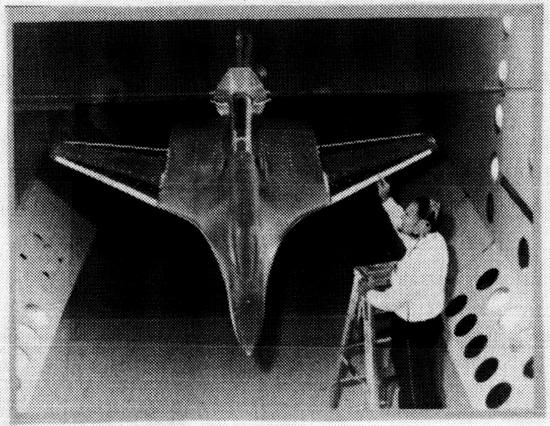
Research Objective - The Active Flexible Wing (AFW) Program is a joint NASA-LaRC / Rockwell effort with the objective of developing and validating the analysis, synthesis and test methodologies required to apply active controls technology for improving aircraft performance and stability. This program is providing an opportunity to design control systems, to develop simulation techniques, and to gain experience with digital multi-input/multi-output (MIMO) control law implementation procedures.

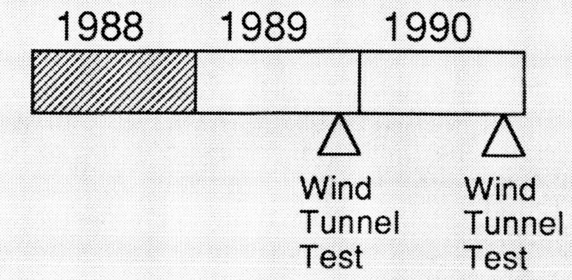
Approach - The approach selected to accomplish the program objectives is to demonstrate various MIMO control concepts on the flexible full-span wind tunnel model shown in the figure 82(b). The major tasks associated with the program include: 1) the design and development of a microprocessor digital controller and its associated hardware; 2) the derivation of the aeroelastic equations of motion using minimum state s-domain unsteady aerodynamics and state-space techniques; 3) the synthesis of Active Flutter Suppression (FSS) and Rolling Maneuver Load Alleviation (RMLA) control laws in the digital domain; 4) the simulation of the digital controller including the effects of structural flexibility and unsteady aerodynamics; 5) the ground testing of the model to define its structural, dynamic and control system characteristics for validating sections of the "truth model"; and 6) the wind tunnel tests. The joint program spans approximately three years, and will involve testing the AFW aeroelastic model in the 16-foot Transonic Dynamics Tunnel on two separate occasions. During the first tunnel entry, FSS and RMLA systems will be investigated separately; in the second test, multifunction digital control law design procedures will be validated by demonstrating FSS and RMLA systems simultaneously.

Status/Plans - The time line indicates that the joint program has been underway for about one year. The items at the bottom of the figure indicate the major accomplishments to date. The first wind tunnel test entry is scheduled to begin in August 1989.

Figure 82 (a).

MULTI-YEAR JOINT NASA/ROCKWELL AEROSERVOELASTIC WIND TUNNEL TEST PROGRAM UNDERWAY





Accomplishments to Date

- o Initiated Memorandum of Agreement with Rockwell
- o Awarded Contract to Rockwell
- o Formed Interdirectorate Team at LaRC
- o Designed Preliminary Flutter Suppression System
- o Designed Preliminary Near-Real-Time Simulation
- o Designed and Built Digital Controller
- o Transferred Technology Directly to Industry

Figure 82 (b).

ORIGINAL PAGE IS OF POOR QUALITY

REUSABLE FLYBACK BOOSTER AEROSERVOELASTIC CHARACTERISTICS TO BE DETERMINED

Michael G. Gilbert
Aeroservoelasticity Branch

RTOP 946-01-00

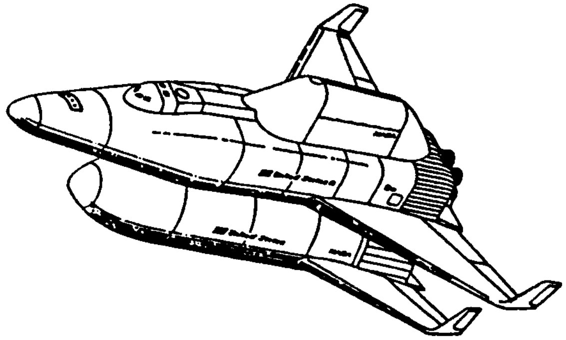
Research Objective - System architecture studies for the Advanced Launch System (ALS) vehicle have led to reusable, flyback booster concepts as possible candidates to meet the low-cost-to-orbit requirements of the system. The flyback booster concepts are tank vehicles with rocket propulsion, wings with winglets, but no vertical tails. The flyback booster would be used to launch the main vehicle, separating at about Mach 3.0 and flying back to the launch site for normal landing on a runway for reuse. The objectives of this research program are the determination of the aeroelastic stability characteristics of the flyback booster concepts, the examination of the potential design load and fatigue payoffs of closed-loop structural and gust load alleviation systems, and the early prediction of any adverse aeroservoelastic (ASE) problems.

Approach - The approach to the aeroservoelastic validation of proposed ALS flyback booster concepts is a multi-year, analytical (figure 83(b)) and experimental program geared to the research objectives and the ALS Phase II Design/Demonstration schedule. Initial efforts will be analytical studies of the various flyback booster concepts involving the development of mathematical structural and dynamic models of the vehicles, aeroelastic stability analyses, and the design and development of a semispan aeroelastic wind tunnel model of a representative configuration. This model would be tested in FY 91 to verify the aeroelastic analyses and to obtain dynamic gust load information for use in the design of a closed-loop load alleviation system. A second, full span, free-flying aeroservoelastic wind tunnel model with actuated control surfaces would then be designed, built, and tested to verify the performance of the load alleviation system design. This model would also be programmed with the vehicle flight control laws to check for adverse ASE interactions between the flight control system and structural dynamics.

Status/Plans - Resource and manpower requirements for the initial analytical studies of the various flyback booster concepts have been determined. These studies will proceed as the concept configuration and structural data becomes available.

Figure 83 (a).

REUSABLE FLYBACK BOOSTER AEROSERVOELASTIC CHARACTERISTICS TO BE DETERMINED



SCHEDULE FOR AEROELASTIC AND ASE CERTIFICATION

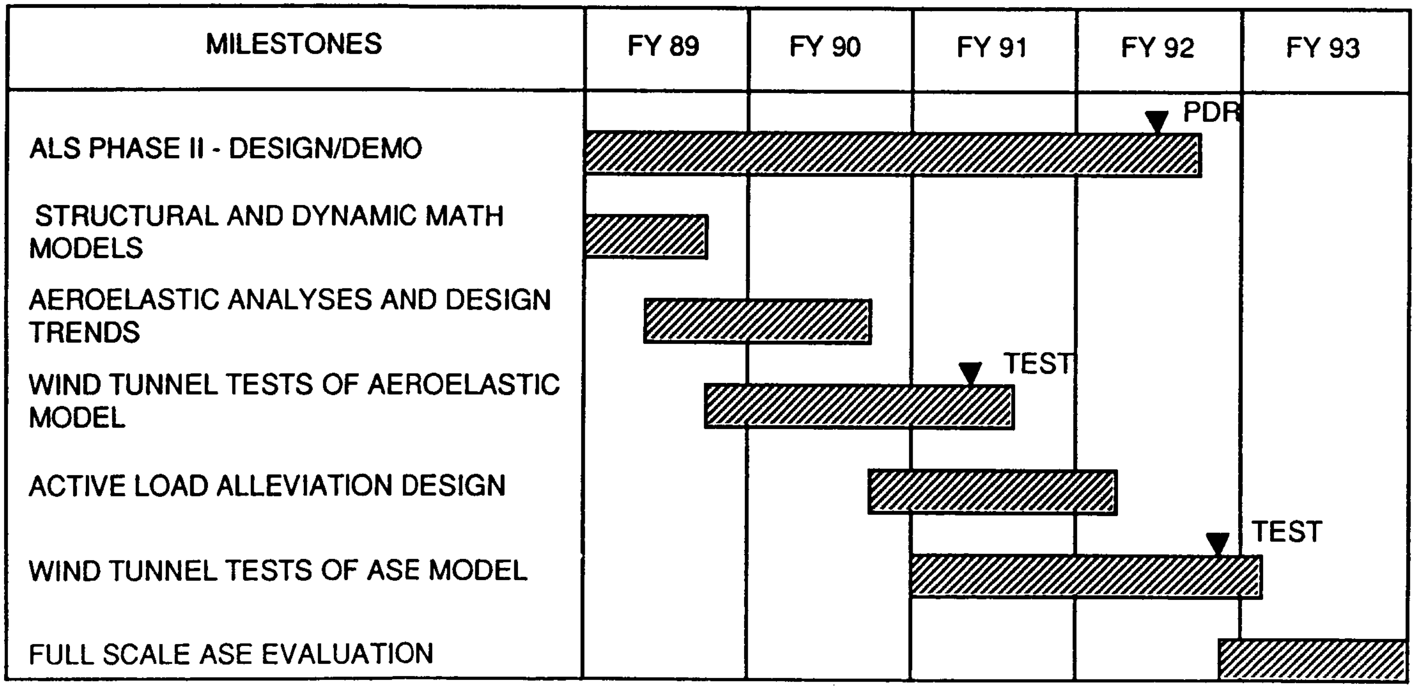


Figure 83 (b).

LANDING AND IMPACT DYNAMICS

FY 89 PLANS

LANDING DYNAMICS

- CONTINUE DEVELOPMENT OF TIRE MODELING STRATEGIES
- DEVELOP DATA BASE ON RADIAL AND H-TYPE AIRCRAFT TIRES
- DEMONSTRATE ACTIVE CONTROL LANDING GEAR
- INITIATE RUNWAY TRACTION PROGRAM
- SUPPORT HEAVY RAIN SIMULATION TESTS

IMPACT DYNAMICS

- CONDUCT STATIC AND DYNAMIC TESTS ON VARIOUS CONCEPTS OF COMPOSITE FRAMES, SUBFLOORS, & ENERGY ABSORBING COMPONENTS
- COMPLETE TESTS AND ANALYSIS OF SCALE MODEL COMPOSITE BEAMS UNDER IMPACT LOADS
- CONTINUE UPDATE OF ELEMENT LIBRARY IN DYCAST FOR COMPOSITE STRUCTURES UNDER CRASH LOADS
- EVALUATE (ON VAX COMPUTER) DYNA3D, PAMCRASH, AND OTHER CODES FOR POTENTIAL APPLICATIONS TO COMPOSITE STRUCTURES
- INITIATE MULTIPLE USAGE TESTS OF COMPOSITE (LEAR) FUSELAGE COMPONENTS

Figure 84.

SPACECRAFT DYNAMICS

FY 89 PLANS

- CONTROLLED MULTIBODY DYNAMICS
  * COMPLETE MULTIBODY MANEUVERING CONTROL EXPERIMENT
  * DEVELOP THEORETICAL BASIS FOR LEARNING CONTROL
  * RELEASE VERSION 1.0 OF 3-D LATDYN AND APPLY TO CSI ARTICULATION STUDIES

- TEST METHODS AND ANALYTICAL SIMULATION
  * COMPLETE TESTS OF GENERIC SPACE STATION MODEL
  * DELIVER PHASE I HYBRID SCALE MODEL AND SUPENSION
  * INITIATE PI ACTIVITIES FOR SPACE STATION STRUCTURAL CHARACTERIZATION EXPERIMENT (SSSCE)
  * INSTALL AND INITIATE TESTS OF PHASE 0 CSI TEST ARTICLE

Figure 85.

NASA National Aeronautics and Space Administration

Report Documentation Page

1. Report No.	2. Government Accession No.	3. Recipient's Catalog No.
NASA TM-101543		

4. Title and Subtitle	5. Report Date
STRUCTURAL DYNAMICS DIVISION RESEARCH AND TECHNOLOGY ACCOMPLISHMENTS FOR FY 1988 AND PLANS FOR FY 1989	January 1989
	6. Performing Organization Code
7. Author(s)	8. Performing Organization Report No.
James E. Gardner	
	10. Work Unit No. 505-63-21-02
9. Performing Organization Name and Address	11. Contract or Grant No.
Langley Research Center Hampton, Virginia 23665-5225	
12. Sponsoring Agency Name and Address	13. Type of Report and Period Covered Technical Memorandum
National Aeronautics and Space Administration Washington, DC 20546	14. Sponsoring Agency Code

15. Supplementary Notes

16. Abstract

The purpose of this paper is to present the Structural Dynamics Division's research accomplishments for FY 1988 and research plans for FY 1989. The work under each Branch (technical area)is described in terms of highlights of accomplishments during the past year and highlights of plans for the current year as they relate to five year plans for each technical area. This information will be useful in program coordination with other government organizations and industry in areas of mutual interest.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
Accomplishments, research plans, aeroelasticity, dynamics	Unclassified - Unlimited Subject Category - 02

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of pages	22. Price
Unclassified	Unclassified	196	A09

NASA FORM 1626 OCT 86