

**NASA Technical Memorandum 100534**

(NASA-TM-100534) LOADS AND AEROELASTICITY  
DIVISION RESEARCH AND TECHNOLOGY  
ACCOMPLISHMENTS FOR FY 1987 AND PLANS FOR FY  
1988 (NASA) 143 p CSDL 01A

N88-16680

Unclas

G3/02 0124551

**LOADS AND AEROELASTICITY DIVISION RESEARCH  
AND TECHNOLOGY ACCOMPLISHMENTS FOR FY 1987  
AND PLANS FOR FY 1988**

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January 1988

**NASA**

National Aeronautics and  
Space Administration

**Langley Research Center**  
Hampton, Virginia 23665

**LOADS AND AEROELASTICITY DIVISION  
RESEARCH AND TECHNOLOGY ACCOMPLISHMENTS FOR FY 1987  
AND PLANS FOR FY 1988**

**SUMMARY**

The purpose of this paper is to present the Loads and Aeroelasticity Division's research accomplishments for FY 87 and research plans for FY 88. The work under each branch (technical area) will be 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 by directorates as shown on figure 1. Directorates are organized by divisions and offices. The Loads and Aeroelasticity 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 69 NASA civil servants and nine members of the Army Aerostructures Directorate, USAARTA, Army Aviation Systems Command located at Langley Research Center. Recent changes in key people include the selection of Dr. Thomas E. Noll as Assistant Head of the Aeroservoelasticity Branch, the reassignment of Robert V. Doggett Jr. from Head, Configuration Aeroelasticity Branch, to LAD Chief Scientist, and the appointment of Dr. John G. Davis as Manager, Technology for Hypersonic Structures. Each branch represents a technical area and disciplines under the technical areas are shown on the figure. All of the Army personnel work on the Rotorcraft Aeroelasticity and Rotorcraft Structural Dynamics disciplines.

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 Prediction and Control of Aeroelastic Stability and Response of Aircraft and Rotorcraft. The Aerothermal Loads Branch (ALB) and the Thermal Structures Branch (TSB) work the areas of Lightweight, Hot Structures for High Speed Vehicles (Aeronautics) and Aerothermal Structures and Materials Technology for Space Transportation Systems (Space).

**RESEARCH PHILOSOPHY**

The basic philosophy and motivation of the Loads and Aeroelasticity Division research program can be captured in selected quotes from some leaders in

Aerospace Research and Development. In his 13th Von Karman lecture on Aeroelasticity (ref. 1), I. E. Garrick related the following: "Von Karman's sense of humor, which was remarkably appropriate to a given occasion, has become legendary. Recognizing that the poor structures engineer was usually held accountable for structural integrity, he quipped, 'The aerodynamicist assumes everything but the responsibility.'"

"It has been gratifying to me to observe that in major aerospace industry the aeroelastician is no longer the stepchild he once was. From an almost parochial isolated specialist, he is now the generalist who tends to pull together the separate efforts in structures, aerodynamics, stability and control, and propulsion, even in early design stages. Yet, there are still human problems such as one-way communications and barriers between departments as well as physical problems that are often so recondite and difficult that aeroelastic problems may slip through the cracks."

In his Wright Brothers Lectureship in Aeronautics on Optimization (ref. 2), Holt Ashley observed: "Further mention will be made in what follows of the keen disappointment felt by many specialists because their theories have received so little practical application. This phenomenon is frequently attributed to a reluctance by developmental engineers to adopt unfamiliar and untried methods of analysis."

In an appraisal of a study of hypersonic airframe structures, (ref. 3), Rene' Miller stated: "The cost effectiveness of .... (Thermal) Structural Concepts is greatly dependent on solutions to the detailed design problems. In fact, it is likely that these detailed design problems as demonstrated in the X-15 program will prove to be the pacing item in the development of Hypersonic Aircraft."

The Loads and Aeroelasticity Division program is aimed at producing the data and analysis methods required by those who are accountable for the structural integrity of aerospace vehicles; to provide the detailed design data and methods for the pacing item of development of hypersonic vehicles - cost effective thermal structures; to continue to pull together those separate efforts that ought to (or must) be considered as a single task to preclude aeroelastic problems from slipping through the cracks; and to alleviate the reluctance by developmental engineers to adopt unfamiliar and untried methods by making them both familiar and proven.

## **FACILITIES**

The Loads and Aeroelasticity Division has two major facilities available to support its research as shown in figure 4.

The Transonic Dynamics Tunnel (TDT) is a Mach 0.2 to 1.2 continuous flow, variable-pressure wind tunnel with a 16-foot-square test section which uses a Freon-12 test medium primarily for dynamic aeroelastic testing. This unique facility is used primarily by the Configuration Aeroelasticity Branch. Semi-span, side-wall mounted

models and full-span cable-mounted models are used for aeroelastic studies of fixed wing aircraft. The Aeroelastic Rotor Experimental System (ARES) test stand is used in the tunnel to study the aeroelastic effects on rotors. A General Rotor Aeroelastic Laboratory, located nearby, is used to setup the ARES test stand in preparation for entry into the TDT and for rotorcraft studies in hover. A modernization of the TDT Data Acquisition System is underway and should be operational by mid FY 88. The maximum Reynolds number is about  $10 \times 10^6/\text{ft}$ . The replacement cost for this facility is \$76M.

The Aerothermal Loads Complex consists of four facilities which are operated by the Aerothermal Loads Branch to carry out their research. The 8-Foot High Temperature Tunnel (8' HTT) is a unique hypersonic Mach 7 blowdown wind tunnel with an 8' diameter test section (uniform temperature test core of 4') that uses products of combustion (methane and air under pressure) as the test medium. The tunnel operates at dynamic pressures of 250 to 1800 psf, temperatures of 2400 to 3600°R and Reynolds numbers of 0.3 to  $2.2 \times 10^6/\text{ft}$ . The tunnel is used to test 2-D and 3-D type models to determine aerothermal loads and to evaluate new high temperature structural concepts. A major Coff item is underway to provide alternate Mach number capability and oxygen enrichment for the test medium. This is being done primarily to allow the tunnel to test models that have hypersonic air-breathing propulsion applications. The replacement cost for the tunnel is \$50.4M.

The 7-Inch High Temperature Tunnel (7" HTT) is a nearly 1/12 scale of the 8' HTT with basically the same capabilities as the larger tunnel. It is used primarily as an aid in the design of larger models for the 8' HTT and for aerothermal loads tests on subscale models. The 7" HTT is currently being used to evaluate various new systems for the 8' HTT. The replacement cost for the tunnel is \$0.8M.

The two Aerothermal Arc Tunnels (20 MW and 5 MW) are used to test models in an environment that simulates the flight reentry envelope for high-speed vehicles such as the Space Shuttle. The amount of usable energy to the test medium in these facilities is 9 MW and 2 MW. The 5 MW is a three phase AC arc heater while the 20 MW is a DC arc heater. Test conditions such as temperature, flow rate, and enthalpy vary greatly since a variety of nozzles and throats are available and since model sizes are different (3" diameter to 1' x 2' panels). The replacement cost for these arc tunnels is \$24M. These facilities are currently on a standby status.

## **FY 87 ACCOMPLISHMENTS**

### **Configuration Aeroelasticity Branch**

The Configuration Aeroelasticity Branch conducts research (fig. 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 87 accomplishments listed below are highlighted by figures 7 through 14.

**Aircraft Aeroelasticity:**

- Improved Adaptive Suppression System Evaluated in TDT Tests
- Some Effects of Speed Brakes on Wing Flutter
- Effects of Mach Number on Buffet Response of a Twin Vertical Tail Airplane
- Active Flexible Wing Model Successfully Tested in TDT

**Rotorcraft Aeroelasticity:**

- Rotor Tracking Sensitivity Investigated Experimentally

**Rotorcraft Structural Dynamics:**

- New Structural Design Methodology Produces Lighter Composite Main Rotor Blade Designs
- Extension-Twist-Coupled Rotor Blade Concept Shown to be Structurally Feasible
- Initial Evaluation of Industry Codes for Calculation of Coupled Rotor-Airframe Vibrations Completed

Each highlight is accompanied by descriptive material.

Unsteady Aerodynamics Branch

The Unsteady Aerodynamics Branch conducts research (fig. 15) 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 16 which shows the five year plan of the branch for theory development, experiments, and design methods.

A major accomplishment of the year was the Symposium on Transonic Unsteady Aerodynamics and Aeroelasticity held at Langley on May 20-22. Ninety attendees represented all major airframe manufacturers, other government research laboratories, and universities. The number of presentations in areas of active research were:

Transonic small disturbance equation - 5 (3 CAP-TSD, 2 XTRAN3S)  
Higher equation levels - 7 (2 full potential, 4 Euler, 1 Navier-Stokes)  
Viscous interaction methods - 3  
Aeroelastic applications - 3  
Rotor unsteady aerodynamics and aeroelasticity - 2  
Semi-empirical methods - 2

There were two presentations on complete aircraft modeling with the CAP-TSD code (F-15 and F-16 aircraft).

The Unsteady Aerodynamics FY 87 accomplishments listed below are highlighted by figures 17 through 26.

**Methods Development:**

- Development of Transonic Aeroelasticity Code for Realistic Aircraft Configurations
- Algorithm Modifications Improve Stability and Accuracy of CAP-TSD Code
- Entropy and Vorticity Effects Improve Accuracy of Unsteady Transonic Small Disturbance (TSD) Theory
- Time-Accurate Euler Code Developed for Vortex-Dominated Flows Around Maneuvering and Deforming Wings
- Improved Boundary Layer Method for Unsteady Potential Flow Agrees with Navier-Stokes Solution

**Applications:**

- CAP-TSD Results Exhibit Good Agreement with Experiment for Complete F-15
- CAP-TSD Validated for Transonic Flutter Prediction
- Flutter Analysis Accomplished for Slender Delta Wings
- CAP-TSD Pressure Distributions Compare Well with CFL3D (Euler) Solutions
- Navier-Stokes Calculations Correlated with Transonic Cryogenic Wind-Tunnel Data

Each highlight is accompanied by descriptive material.

Aeroservoelasticity Branch

The Aeroservoelasticity Branch conducts research (fig. 27) to develop methodologies for the analysis and synthesis of multifunctional active control systems and conceives, recommends, and provides technical support for experiments to validate the methodologies. Mathematical models needed to support NASA projects are used to verify the theoretical developments and their computer implementations. This work is more clearly identified in figure 28 which shows the five year plan of the three disciplines and their expected results.

The Aeroservoelasticity FY 87 accomplishments listed below are highlighted by figures 29 through 34.

### **Analysis and Design Methods:**

- Integrated Structure/Control Law Design
- Digital Robust Active Control Law Synthesis Using Constrained Optimization
- Methodology for Matching Experimental and Computational Aerodynamic Data
- Comparison of SDG and PSD Methods

### **Applications and Validations:**

- Flexible Wing Response Correlates with Changing Flow Conditions in Transonic Range
- Aeroservoelastic Analysis Validated by Wind-Tunnel Tests

Each highlight is accompanied by descriptive material.

### Aerothermal Loads Branch

The Aerothermal Loads Branch conducts research (fig. 35) to develop and validate solution algorithms, modeling techniques, and integrated finite element solutions for fluid-thermal-structural analysis; to identify and understand flow phenomena and flow/surface/structure interaction parameters required to define detailed aerothermal loads for structural design via analysis and test; and to define methods for testing in high enthalpy flow environments including the capability for testing air breathing engines at hypersonic speeds. This work is more clearly identified in figure 36 which shows the five year plan of the three disciplines and their expected results.

The Aerothermal Loads Branch FY 87 accomplishments listed below are highlighted by figures 37 through 42.

### **Experiments:**

- Aerothermal Study of Simulated Shuttle Chine Tile- Gaps in the 8-Foot High Temperature Tunnel
- Leading Edge Sweep Reduces Peak Pressure and Heat Transfer Rate
- Advanced Actively Cooled Radome Performance Demonstrated in the LaRC 8' HTT

### **Analysis:**

- Adaptive Unstructured Remeshing Improves Computational Efficiency
- Finite Element Inviscid Results In Good Agreement With Experimental Results
- Langley Integrated Fluid Thermal Structural Analysis of Blunt Leading Edge

Each highlight is accompanied by descriptive material.

### Thermal Structures Branch

The Thermal Structures Branch conducts research (fig. 43) to develop and validate concepts for aerospace structures whose design is significantly controlled by the

thermal excursions of the operating environments of aerospace vehicles. Systems studies in concert with the Space Systems Division or Hypersonics Technology Office help to identify structures and materials technology needs. Structural concepts are then developed, analyzed, fabricated, and tested to verify the required technology advances. This work is more clearly identified in figure 44 which shows the five year plan of the three major disciplines and their expected results. Thermal structures experimental needs are currently in the definition stage. Static testing of small components is currently being done with the support of contractors, ADFRF, and the Aerothermal Loads Branch.

The Thermal Structures FY 87 accomplishments listed below are highlighted by figures 45 through 51.

**Cooled Structures:**

- Methodology for Designing Cooled Engine Wall Structure

**Hot/Cryo Structures:**

- Carbon-Carbon/Refractory-Metal Heat-Pipe Structure Extends the Use of Carbon-Carbon in Withstanding High Heat Loads
- Blade-Stiffened Carbon-Carbon Compression Panels
- Blatt's Equation Used to Characterize Effects of Thermal Stratification for NASP Vehicle Design

**Analysis and Synthesis Methods:**

- Transient Finite-Element Heat-Pipe Analysis Predicts Startup from a Frozen State
- Improved Intracell Buckling Coefficient Leads to Weight Reduction for Honeycomb Panels
- New Direct Solution Developed for Thermal Stresses in a Spherical Nose Cap Under an Arbitrary Temperature Distribution

Each highlight is accompanied by descriptive material.

**PUBLICATIONS**

The FY 87 accomplishments of the Loads and Aeroelasticity Division resulted in a number of publications. The publications are listed below by organization and are identified by the categories of journal publications, formal NASA reports, conference presentations, contractor reports, tech briefs, and patents.

Division Office

**Formal Reports**

1. Gardner, J. E.; and Dixon, S. C.: Loads and Aeroelasticity Division Research and Technology Accomplishments for FY 1986 and Plans for FY 1987. NASA TM-89084, January 1987.



## Conference Presentations

2. Noor, A. K.; and Dixon, S. C.: Flight-Vehicle Structures Education in the United States - Assessment and Recommendations. Presented at the AIAA/ASME/ASCE/AHS, 28th Structures, Structural Dynamics and Materials Conference, April 6-8, 1987, Monterey, California. AIAA Paper No. 87-0978-CP. Also available in Computers and Structures, Vol.27, No. 6, 1987, pp. 695-719.

## Configuration Aeroelasticity Branch

## Journal Publications

3. Cole, S. R.: Exploratory Flutter Test in a Cryogenic Wind Tunnel. Journal of Aircraft, Vol. 23, No. 12, December 1986, pp. 904-911.
4. Lakin, W. D.: A Combined Integrating and Differentiating Matrix Formulation for Boundary Value Problems on Rectangular Domains. Journal of Engineering Mathematics, Vol. 20, pp. 203-215, 1986.
5. Hodges, D. H.; Hopkins, A. S.; Kunz, D. L.; and Hinnant, H. E.: Introduction to GRASP - General Rotorcraft Aeromechanical Stability Program - A Modern Approach to Rotorcraft Modeling. Journal of the American Helicopter Society, Vol. 32, No. 2, pp. 78-90, April 1987.

## Formal NASA Reports

6. Pearson, R. M.; Giesing, J. P.; Nomura, J. K.; and Ruhlin, C. L.: Transonic Flutter-Model Study of a Multijet Airplane Wing with Winglets Including Considerations of Separated Flow and Aeroelastic Deformation Effects. NASA TM-87753, November 1986.
7. Ruhlin, C. L.; Bhatia, J. G.; and Nagaraja, K. S.: Effects of Winglet on Transonic Flutter Characteristics of a Cantilevered Twin-Engine-Transport Wing Model. NASA TP-2627, December 1986.
8. Hodges, R. V.; Nixon, M. W.; and Rehfield, L. W.: Comparison of Composite Rotor Blade Models: A Coupled-Beam Analysis and an MSC/NASTRAN Finite Element Model. NASA TM-89024 and AVSCOM TM 87-B-2, March 1987.
9. Yeager, W. T., Jr.; Hamouda, M-N. H.; and Mantay, W. R.: An Experimental Investigation of the Aeromechanical Stability of a Hingeless Rotor in Hover and Forward Flight. NASA TM-89107, AVSCOM TM 87-B-5, June 1987.
10. Nixon, M. W.: Preliminary Structural Design of Composite Main Rotor Blades for Minimum Weight. NASA TP-2730, AVSCOM TM 87-B-6, July 1987.

11. Yeager, W. T., Jr.; Mantay, W. R.; Wilbur, M. L.; Cramer, R. G., Jr.; and Singleton, J. D.: Wind-Tunnel Evaluation of an Advanced Main-Rotor Blade Design for a Utility-Class Helicopter. NASA TM-89129, AVSCOM TM 87-B-8, September 1987.

### **Conference Presentations**

12. Kvaternik, R. G.; Bartlett, F. D.; and Cline, J. H.: A Summary of Recent NASA/Army Contributions to Rotorcraft Vibrations and Structural Dynamics Technology. Presented at the 1987 NASA/Army Rotorcraft Technology Conference, March 17-19, 1987, Moffett Field, California. NASA CP pending.
13. Cole, S. R.; Rivera, J. A., Jr.; and Nagaraja, K. S.: Flutter Study of an Advanced Composite Wing with External Stores. Presented at the AIAA/ASME/ASCE/AHS, 28th Structures, Structural Dynamics and Materials Conference, April 6-8, 1987, Monterey, California. AIAA Paper No. 87-0880-CP.
14. Hinnant, H. E.; and Hodges, D. H.: Application of GRASP to Nonlinear Analysis of a Cantilever Beam. Presented at the AIAA Dynamics Specialists Conference, April 9-10, 1987, Monterey, California. AIAA Paper No. 87-0953-CP.
15. Nixon, M. W.: Extension-Twist Coupling of Composite Circular Tubes with Application to Tilt Rotor Blade Design. Presented at the AIAA/ASME/ASCE/AHS, 28th Structures Structural Dynamics and Material Conference, April 6-8, 1987, Monterey, California. AIAA Paper No. 87-0772-CP.
16. Piette, D. S.; and Cazier, F. W., Jr.: Experimental Transonic Steady State and Unsteady Pressure Measurements on a Supercritical Wing During Flutter and Forced Discrete Frequency Oscillations. Presented at the NASA Symposium on Transonic Unsteady Aerodynamics and Aeroelasticity, May 20-22, 1987, Hampton, Virginia. NASA CP pending.

### **Contractor Reports**

17. Dompka, R. V. and Cronkhite, J. D.: Summary of AH-1G Flight Vibration Data for Validation of Coupled Rotor-Fuselage Analyses. NASA CR-178160, Nov. 1986.
18. Straub, F. K., and Byrns, E. V. Jr.: Application of Higher Harmonic Blade Feathering on the OH-6A Helicopter for Vibration Reduction, NASA CR-4031, December 1986.
19. Cronkhite, J. D.; Berry, V. L.; and Dompka, R. V.: Summary of the Modeling and Test Correlations of a NASTRAN Finite Element Vibrations Model for the AH-1G Helicopter. NASA CR-178201, January 1987.
20. Noor, A. K.: Flight-Vehicle Structures Education in the United States - Assessment and Recommendations. NASA CR-4048, February 1987.

21. Halwes, D. R.: Ground and Flight Test Results of a Total Main Rotor Isolation Systems. NASA CR-4082, July 1987.

### Unsteady Aerodynamics Branch

#### **Journal Publications**

22. Batina, J. T.: Unsteady Transonic Flow Calculations for Wing-Fuselage Configurations. Journal of Aircraft, Vol. 23, No. 12, December 1986, pp. 897-903.
23. Seidel, D. A.; Sandford, M. C.; and Eckstrom, C. V.: Measured Unsteady Transonic Aerodynamic Characteristics of an Elastic Supercritical Wing. Journal of Aircraft, Vol. 24, No. 4, April 1987, pp. 225-230.
24. Bennett, R. M.; Wynne, E. C.; and Mabey, D. G.: Calculation of Transonic Steady and Oscillatory Pressures on a Low Aspect Ratio Model. Journal of Aircraft, Vol. 24, No. 6, June 1987, pp. 392-398.
25. Smith, G. E.; Whitlow, W., Jr.; and Hassan, H. A.: Unsteady Transonic Flows Past Airfoils Using the Euler Equations. Journal of Aircraft, Vol. 24, No. 9 September 1987, pp. 670-672.

#### **Formal NASA Reports**

26. Hess, R. W.; Cazier, F. W., Jr.; and Wynne, E. C.: Steady and Unsteady Transonic Pressure Measurements on a Clipped Delta Wing for Pitching and Control-Surface Oscillations. NASA TP-2594, October 1986.
27. Cunningham, H. J.: Steady and Unsteady Aerodynamic Forces From the SOUSSA Surface-Panel Method for a Fighter Wing with Tip Missile and Comparison with Experiment and PANAIR. NASA TP-2736, August 1987.
28. Howlett, J. T.; and Bland, S. R.: Calculation of Viscous Effects on Transonic Flow for Oscillating Airfoils and Comparisons with Experiment. NASA TP-2731, September 1987.

#### **Conference Presentations**

29. Yates, E. C., Jr.; and Whitlow, W., Jr.: Development of Computational Methods for Unsteady Aerodynamics at the NASA Langley Research Center. Presented at the 63rd Meeting of the AGARD Structures and Materials Panel Workshop on Future Research on Transonic Unsteady Aerodynamics and Its Aeroelastic Applications, September 28 - October 3, 1986, Athens, Greece. Also available as NASA TM-89133, March 1987.

30. Yates, E. C., Jr.; and Chu, L. C.: Static Aeroelastic Effects on the Flutter of Supercritical Wing. Presented at the 63rd Meeting of the AGARD Structures and Materials Panel Specialists' Meeting on Static Aeroelastic Effects on High-Performance Aircraft, September 28 - October 3, 1986, Athens, Greece. Also available as NASA TM-89132, March 1987.
31. Batina, J. T.: An Efficient Algorithm for Solution of the Unsteady Transonic Small-Disturbance Equation. Presented at the AIAA 25th Aerospace Sciences Meeting, January 12-15, 1987, Reno, Nevada. AIAA Paper No. 87-0109. Also available as NASA TM-89014, December 1986.
32. Edwards, J. W.; and Thomas, J. L.: Computational Methods for Unsteady Transonic Flows. Presented at the AIAA 25th Aerospace Sciences Meeting, January 12-15, 1987, Reno, Nevada. AIAA Paper No. 87-0107-CP. Also available as NASA TM-89106, March 1987.
33. Hess, R. W.; Seidel, D. A.; Igoe, W. B.; and Lawing, P. L.: Highlights of Unsteady Pressure Tests on a 14 Percent Supercritical Airfoil at High Reynolds Number, Transonic Condition. Presented at the AIAA 25th Aerospace Sciences Meeting, January 12-15, 1987, Reno, Nevada. AIAA Paper No. 87-0035-CP. Also available as NASA TM-89080, January 1987.
34. Batina, J. T.; Seidel, D. A.; Bland, S. R.; and Bennett, R. M.: Unsteady Transonic Flow Calculations for Realistic Aircraft Configurations. Presented at the AIAA/ASME/ASCE/AHS, 28th Structures, Structural Dynamics and Materials Conference, April 6-8, 1987, Monterey, California. AIAA Paper No. 87-0850-CP. Also available as NASA TM-89120, March 1987.
35. Bennett, R. M.; Bland, S. R.; Batina, J. T.; Gibbons, M. D.; and Mabey, D. G.: Calculation of Steady and Unsteady Pressures on Wings at Supersonic Speeds with a Transonic Small Disturbance Code. Presented at the AIAA/ASME/ASCE/AHS, 28th Structures, Structural Dynamics and Materials Conference, April 6-8, 1987, Monterey, California. AIAA Paper No. 87-0851-CP.
36. Seidel, D. A.; Eckstrom, C. V.; and Sandford, M. C.: Investigation of Transonic Region of High Dynamic Response Encountered on an Elastic Supercritical Wing. Presented at the AIAA/ASME/ASCE/AHS 28th Structures, Structural Dynamics, and Materials Conference, April 6-8, 1987, Monterey, California. AIAA Paper No. 87-0735-CP. Also available as NASA TM-89121, March 1987.
37. Batina, J. T.: CAP-TSD: A Program for Unsteady Transonic Analysis of Realistic Aircraft Configurations. Presented at the NASA Symposium on Transonic Unsteady Aerodynamics and Aeroelasticity, May 20-22, 1987, Hampton, Virginia. NASA CP pending.

38. Bennett, R. M.; Bland, S. R.; Batina, J. T.; Gibbons, M. D.; and Mabey, D. G.: Calculation of Steady and Unsteady Pressures on Wings at Supersonic Speeds with CAP-TSD. Presented at the NASA Symposium on Transonic Unsteady Aerodynamics and Aeroelasticity, May 20-22, 1987, Hampton, Virginia. NASA CP pending.
39. Cunningham, H. J.; Bennett, R. M.; and Batina, J. T.: Initial Application of CAP-TSD to Wing Flutter. Presented at the NASA Symposium on Transonic Unsteady Aerodynamics and Aeroelasticity, May 20-22, 1987, Hampton, Virginia. NASA CP pending.
40. Hess, R. W.: Unsteady Pressure Measurements on a Airfoil at High Reynolds Number. Presented at NASA Symposium on Transonic Unsteady Aerodynamics and Aeroelasticity, May 20-22, 1987, Hampton, Virginia. NASA CP pending.
41. Howlett, J. T.: Viscous Flow Calculations for the AGARD Standard Configuration Airfoils with Experimental Comparisons. Presented at the NASA Symposium on Transonic Unsteady Aerodynamics and Aeroelasticity, May 20-22, 1987, Hampton, Virginia. NASA CP pending.
42. Seidel, D. A.; Adams, W. H., Jr.; Eckstrom, C. V.; and Sandford, M. C.: Investigation and Suppression of High Dynamic Response Encountered on an Elastic Supercritical Wing. Presented at the NASA Symposium on Transonic Unsteady Aerodynamics and Aeroelasticity, May 20-22, 1987, Hampton, Virginia. NASA CP pending.
42. Whitlow, W., Jr.: Application of a Full Potential Method to AGARD Standard Airfoils. Presented at NASA Symposium on Transonic Unsteady Aerodynamics and Aeroelasticity, May 20-22, 1987, Hampton, Virginia. NASA CP pending.

### **Tech Briefs**

43. 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.
44. Yates, E. C., Jr.: Model Data for Wing 445.6. NASA Tech Brief LAR-13903.

### Aeroservoelasticity Branch

### **Journal Publications**

45. Mukhopadhyay, V.: Stability Robustness Improvement Using Constrained Optimization Techniques. Journal of Guidance, Control, and Dynamics, Vol. 10, No. 2, March-April 1987, pp. 172-177.

## Conference Presentations

46. Eckstrom, C. V.: Prediction of Wing Aeroelastic Effects on Aircraft Lift and Pitching Moment Characteristics. Presented at the 63rd Meeting of the AGARD Structures and Materials Panel Specialists' Meeting on Static Aeroelastic Effects on High-Performance Aircraft, September 28 - October 3, 1986, Athens, Greece. Conference Proceedings No. 403. Also available as NASA TM-89060, Oct. 1986.
47. Adams, Jr., W. M.; Tiffany, S. H.; and Bardusch, R. E.: Active Suppression of an "Apparent Shock Induced Instability." Presented at the AIAA/ASME/ASCE/AHS 28th Structures, Structural Dynamics, and Materials Conference, April 6-8, 1987, Monterey, California. AIAA Paper No. 87-0881-CP.
48. Arbuckle, P. D.; Buttrill, C. S.; and Zeiler, T. A.: A New Simulation Model Building Process for Use in Dynamic Systems Integration Research. Presented at the AIAA Flight Simulation Technologies Conference, August 17-19, 1987, Monterey, California. AIAA Paper No. 87-2498-CP.
49. Tiffany, S. H.; and Adams, Jr., W. M.: Nonlinear Programming Extensions to Rational Function Approximations of Unsteady Aerodynamics. Presented at the AIAA/ASME/ASCE/AHS 28th Structures, Structural Dynamics, and Materials Conference, April 6-8, 1987, Monterey, California. AIAA Paper No. 87-0854-CP.
50. Mukhopadhyay, V.: Digital Robust Active Control Law Synthesis for Large Order Flexible Structure Using Parameter Optimization. Presented at the Sixth VPI&SU/AIAA Symposium on Dynamics and Control of Large Structures, June 29-July 1, 1987, Blacksburg, Virginia.
51. Buttrill, C.S.; Zeiler, T. A.; and Arbuckle, P. D.: Nonlinear Simulation of a Flexible Aircraft in Maneuvering Flight. Presented at the AIAA Flight Simulation Technologies Conference, August 17-19, 1987, Monterey, California. AIAA Paper No. 87-2501-CP.
52. Gilbert, M. G.; and Silva, W. A.: The Effects of Aeroelastic Deformation on the Unaugmented Stopped-Rotor Dynamics of an X-Wing Aircraft. Presented at the AIAA Atmospheric Flight Mechanics Conference, August 17-19, 1987, Monterey, California. AIAA Paper No. 87-2563-CP. Also available as NASA TM-100480, June 1987.
53. Mukhopadhyay, V.: Digital Robust Active Control Law Synthesis for Large Order Flexible Structure Using Constrained Optimization. Presented at the AIAA Guidance, Navigation, and Control Conference, August 17-19, 1987, Monterey, California. AIAA Paper No. 87-2588-CP.

## **Formal NASA Reports**

54. Camp, D. W.; and Frost, W. (Editors): Atmospheric Turbulence Relative to Aviation, Missile, and Space Programs. NASA CP-2468, April 1987.

## **Contractor Reports**

55. Frost, W.; Chang, H-P.; and Ringnes, E. A.: Analyses and Assessments of Spanwise Gust Gradient Data from NASA B-57B Aircraft. NASA CR-178288, August 1987.

## Aerothermal Loads Branch

### **Formal NASA Reports**

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## **FY 88 PLANS**

The FY 88 plans for the Loads and Aeroelasticity Division are broken out by each of the branches (technical areas) and selected highlights of proposed FY 88 milestones are presented.

### Configuration Aeroelasticity Branch

For FY 88 the Configuration Aeroelasticity Branch (CAB) will continue its broadly based research program on dynamic and aeroelastic phenomena of aircraft and rotorcraft as summarized in figure 52.

A large portion of this work is associated with tests in the Langley TDT with companion theoretical studies. Research studies are planned for both rotorcraft and airplanes. The rotorcraft studies will use the aeroelastic rotor experimental system (ARES). Rotorcraft work will focus on applications of advanced aerodynamic and structural methodology to new rotor concepts. Airplane focused studies will include a parametric study of the flutter characteristics of the Arrow Wing - a supersonic transport-type wing. In addition to research studies, an aeroelastic verification test is planned for the F-16 with a new pylon and with a modified wing.

Work will continue in the area of prediction of helicopter vibration characteristics by using finite element modeling procedures. Studies involving the major airframe manufacturers will be continued. In addition, the development of analysis tools for the design of composite rotor blades, and the testing of tension-torsion coupling of rotating tubes will continue.

Significant progress in the development of a new data acquisition, display, and control system for the TDT is expected. The system is expected to be ready for on-line operation early in 1988.

Selected highlights of proposed FY 88 milestones are listed below and are shown by figures 53 through 56.

**Aircraft Aeroelasticity:**

- Aircraft Aeroelasticity
- Upgrading the Data Acquisition System for the Langley Transonic Dynamics Tunnel

**Rotorcraft Aeroelasticity:**

- Rotorcraft Dynamics and Aeroelasticity

**Rotorcraft Structural Dynamics:**

- A National Capability to Analyze Vibration as Part of Helicopter Structural Design

Each highlight is accompanied by descriptive material.

Unsteady Aerodynamics Branch

For FY 88 there will be continuing activity in developing finite-difference algorithms to solve nonlinear, unsteady fluid flow equations for application to aeroelastic analysis (fig. 57). A major activity will be the release of the CAP-TSD (Computational Aeroelasticity Program-Transonic Small Disturbance) code to industry. Cooperative efforts on applications of the code to complete aircraft configurations will continue to be pursued, in order to assess the accuracy and robustness of the code. Also, work will be continued to assess the utility of the full potential equation for aeroelastic analysis. Work will be initiated to apply Navier-Stokes codes to aeroelastic problems. Interactive viscous boundary layer methods will be further developed; a two-dimensional strip method is available and a three-dimensional unsteady finite difference code will be delivered during this period. Unsteady Euler methods, formulated in body fixed coordinates, will be applied to flexible wing problems to study dynamic vortex-structure interaction.

The experimental program will include two wind tunnel tests. The wind tunnel tests are; a clipped delta wing and a canard interference test and a high angle oscillating delta wing test (the canard model).

Aeroservoelasticity Branch

There are several efforts planned for FY 88 in each of the three major areas of analysis methods, design methods, and applications and validations (fig. 58).

In the analysis methods area, real time simulation using a completely flexible math model with unsteady aerodynamics and pilot inputs is the goal of the FIT

(Functional Integration Technology) activities. An effort to develop unsteady aerodynamic correction factor methodology will be completed. These activities will evaluate the potential for using factors based on experimental steady pressures, total aircraft steady forces, or local aerodynamic section properties for obtaining accurate unsteady force predictions. Studies to evaluate the accuracy and advantages of the British SDG method over the American PSD method of determining gust loads will continue with the addition of flexible mode degrees of freedom in the aircraft representation. As a result of the expanding usage of the inhouse ISAC computer code by the aerospace industry and government labs, both nationally and internationally, all modules of the code will be fully documented with theoretical and user's manuals. Some progress towards the development of aerothermoelastic analysis capability for application to the NASP vehicle will be accomplished in the next year. In the design methods area, the development of an integrated structure and control law design approach based on hierarchal multilevel problem decomposition and optimization techniques using analytical sensitivities of an optimized structural design and of optimum control law solutions will be completed. In addition, control law synthesis methodology will be expanded to include optimization procedures based on constrained time history responses.

In the applications and validations area, a cooperative effort between LaRC and Rockwell International will be initiated to evaluate multipoint (maneuver load control) and multifunction (flutter suppression during rolling maneuvers) control laws using the active flexible wing wind-tunnel model. This effort will involve significant inhouse activity in the areas of design, analysis, real-time simulation, and ground and wind-tunnel testing. Activities to correlate the unsteady flow conditions that occurred on the ARW-2 wind-tunnel model during transition from attached flow to one experiencing a strong recompression shock and boundary layer separation aft of the shock with an unusual high dynamic wing response will be completed. Aeroservoelastic analyses will continue for the RSRA stopped rotor X-Wing aircraft. The X-Wing math model will be modified to include an updated set of vibration modes for the complete vehicle and the flight control laws.

Selected highlights of proposed FY 88 milestones are listed below and are shown by figures 59 through 61.

**Analysis and Design Methods:**

- Develop Methodology to Design Control Laws Based on Constrained Time History Responses

**Applications and Validations:**

- Initiate Cooperative Active Controls Test Program with Rockwell on AFW Model
- Investigate Procedures to Integrate Thermal Effects into ASE Analysis

Each highlight is accompanied by descriptive material.

## Aerothermal Loads Branch

For FY 88, there will be a continuing level of activity in all three disciplines as summarized in figures 62 and 63.

**Experiments** - The major thrusts of the thermal loads research effort for FY 88 consists of five specific tasks: 1) complete tests to establish a design and code validation data base of aerothermal loads for a blunt leading edge model with an impinging shock and for other generic hypersonic vehicle local configurations, 2) document gap heating results for a curved surface subject to a pressure gradient across the surface, 3) characterize the 2-D turbulent boundary layer in the 8' HTT, 4) obtain pressure and heating rate distributions and aerodynamic coefficients for NASP government baseline vehicle, and 5) supersonic jet mass-addition cooling efficiency in presence of impinging shock wave.

**Analysis** - The major thrust for the ALB analytical effort in FY 88 consists of two specific tasks: 1) continued development and validation of finite element methodology for the prediction of aerothermal loads to complement and supplement the experimental effort including implementation and evaluation of 3-D viscous analysis and adaptive/unstructured grid refinement strategies, 2) continue development and validation of the Langley Integrated Fluid-Thermal-Structural (LIFTS) analysis capability as a tool to design and evaluate structural concepts for super/hypersonic vehicles including implementation of non-linear structural behavior. Major efforts will be devoted to completing finite element analyses of a blunt leading edge with an impinging shock and a compression corner utilizing adaptive/unstructured mesh strategies. Specific plans include the development of an implicit time marching algorithm, extension of the 2D adaptive unstructured triangular remeshing scheme to 3D and to thermal structural analysis, evaluation of higher order elements for CFD, and implementation of nonlinear elasto-plastic constitutive relationship for thermal-structural analysis.

**Facilities and Test Techniques** - The facilities effort involves the safe and efficient operation and the expansion of the test capabilities of the high energy facilities of the Aerothermal Loads Branch.

A major effort for facilities development is support of the modification (FY 87 CofF) to the 8' HTT which will make it a unique national research facility for testing air-breathing propulsion systems for very high-speed aircraft and missiles (fig. 64). In support of this effort, the 7" HTT will be used to investigate alternate modes of oxygen injection and to develop propulsion testing support structures.

FY 88 testing in the 8' HTT will include a 2nd generation U.S. Army/DNA model, a Survey of the 2-D Turbulent Boundary Layer on the Panel Holder, a NASP government baseline vehicle, and supersonic-jet mass-addition cooling efficiency. Also, the Reynolds number and Mach number capability of the Calspan 48-Inch

Hypersonic Tunnel will be upgraded and used to complete shock-on-lip tests at high Mach numbers, sweep, and multiple impinging shocks.

Selected highlights of the proposed FY 88 milestones are listed below and are shown in figures 65 through 66.

**Experiments:**

- Incident Shock/ Wall Jet Interaction
- Corner Flow Experiment

Thermal Structures Branch

There are several major research activities for FY 88 which collectively represent a concerted thrust to advance the state of the art in thermal structures (figure 67).

Work on cooled structures will be highlighted by testing of a flightweight SCRAMJET strut in both a combustion facility and a static load laboratory. Initial tests to determine the heat transfer coefficients and pressure drops associated with pin-fin heat exchangers sized for SCRAMJET engines should be completed at NBS, Boulder, CO. Work on the carbon/carbon heat pipe concept will be continued. Testing of water glycol cooled panels in the actively cooled test stand (ACTS) will be completed.

In the area of hot/cryogenic structures, development of carbon/carbon structure for the control surfaces of hypersonic vehicles will be continued. Emphasis will be on control surfaces for the National Aero-Space Plane. Tunnel time permitting, the redesigned curved, thermal protection system (TPS) panel array will be tested in the 8'HTT to complete our studies of panel-to-panel gap heating. The development of requirements for the certification/recertification of large, reusable cryogenic tanks for hypersonic vehicles will be initiated in FY 88.

Research in the area of analysis and synthesis will include studies to investigate the level of discretization required to accurately predict the behavior of thermal structures. These studies will include both general structure (skin, ribs, etc.) and discontinuities such as joints. A highlight in this area will be initial studies to determine thermal effects on global vehicle stiffness. This work is directed toward assessing the impact of transient thermal histories on the aeroelastic behavior of hypersonic vehicles. Additionally, a model-size reduction technique which will result in shorter solution times will be applied to quasi-symmetric thermal problems, transform-method-based finite elements will be extended to nonlinear thermal/structural problems, and the development of optimal radiation surface elements will be completed.

The development of the Thermal Structures Laboratory will continue in FY 88.

## **CONCLUDING REMARKS**

This publication documents the FY 1987 accomplishments, research and technology highlights, and FY 1988 plans for the Loads and Aeroelasticity Division.

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# LANGLEY RESEARCH CENTER

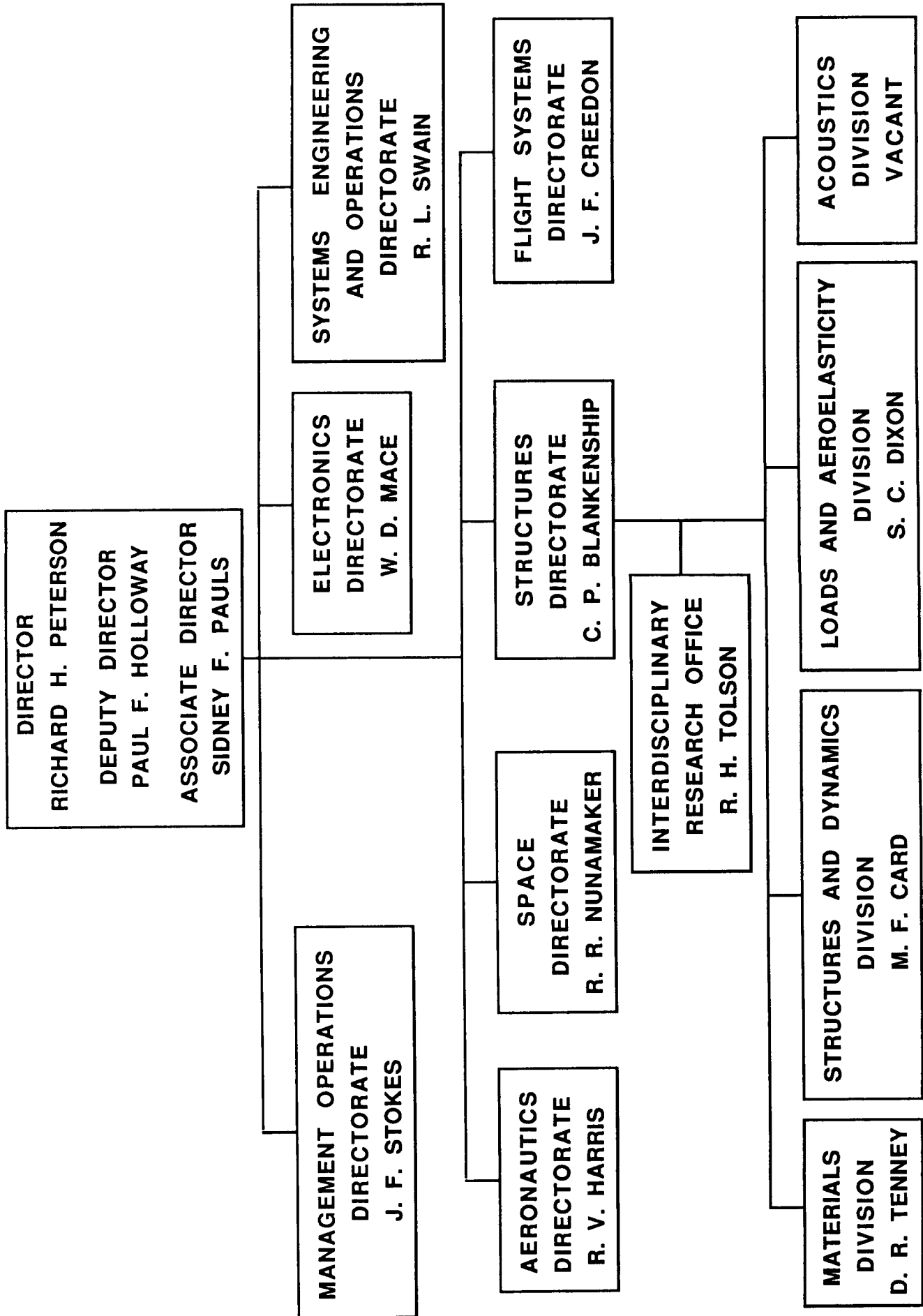


Figure 1.

LOADS AND AEROELASTICITY DIVISION

8

CHIEF: SIDNEY C. DIXON  
ASSISTANT CHIEF: IRVING ABEL  
CHIEF SCIENTIST: ROBERT V. DOGGETT  
TECHNICAL ASSISTANT: JAMES E. GARDNER  
MANAGER, TECHNOLOGY FOR JOHN G. DAVIS  
HYPERSONIC STRUCTURES:

CONFIGURATION AEROELASTICITY      UNSTEADY AERODYNAMICS      AEROSERVOELASTICITY

12 + 9

11

10

ACTING HEAD: IRVING ABEL  
ASSIST: BILL CAZIER

HEAD: JOHN EDWARDS

ACTING HEAD: IRVING ABEL  
ASSIST: THOMAS NOLL

- AIRCRAFT AEROELASTICITY
- ROTORCRAFT AEROELASTICITY
- ROTORCRAFT STRUCTURAL DYNAMICS

- THEORY DEVELOPMENT
- DESIGN METHODS
- EXPERIMENTS

- ANALYSIS METHODS
- DESIGN METHODS
- APPLICATIONS AND VALIDATIONS

AEROTHERMAL LOADS

15

HEAD: ALLAN WIETING

- EXPERIMENTS
- ANALYSIS
- FACILITIES AND TEST TECHNIQUES

THERMAL STRUCTURES

13

HEAD: DONALD RUMMLER  
ASSIST: RODNEY RICKETTS

- PROPULSION STRUCTURES
- AIRFRAME STRUCTURES
- ANALYSIS AND SYNTHESIS METHODS

TOTAL NASA - 69

TOTAL ARMY - 9

# LOADS AND AEROELASTICITY DIVISION

## LONG RANGE THRUSTS

### AERONAUTICS

- AEROELASTIC STABILITY AND RESPONSE OF AIRCRAFT AND ROTORCRAFT
- LIGHTWEIGHT, HOT STRUCTURES FOR HIGH SPEED AIRCRAFT

### SPACE

- AEROTHERMAL STRUCTURES & MATERIALS TECHNOLOGY FOR STS

Figure 3.

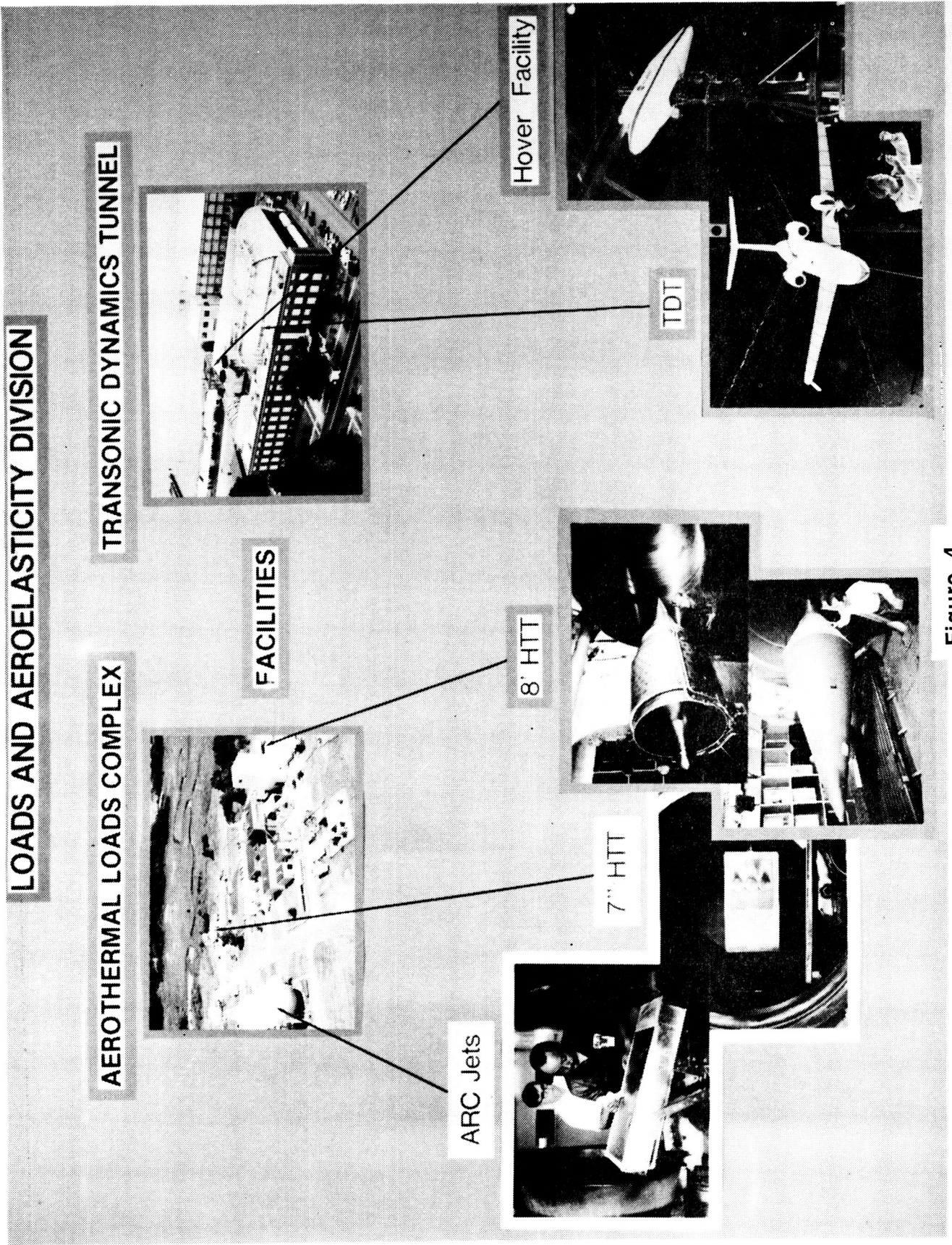


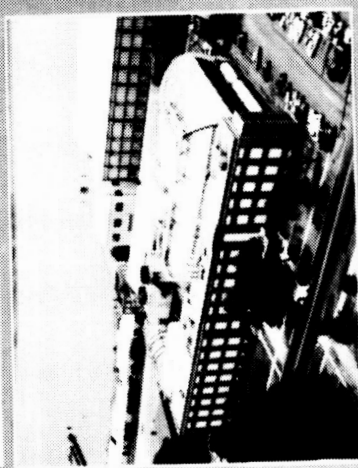
Figure 4.

# CONFIGURATION AEROELASTICITY

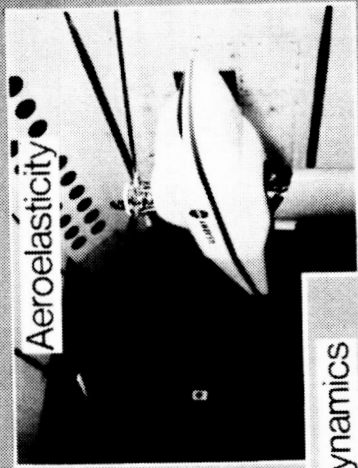
Aircraft



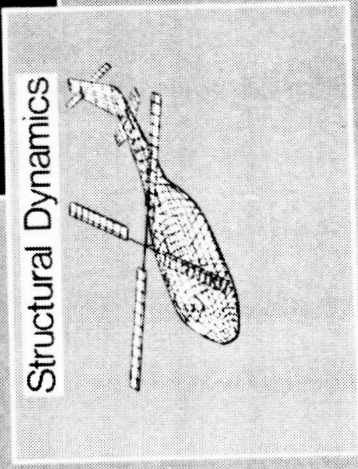
Transonic Dynamics Tunnel



Rotorcraft



Structural Dynamics



Basic Studies

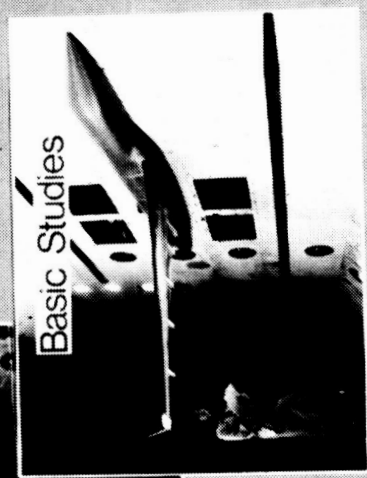


Figure 5.

# CONFIGURATION AEROELASTICITY FIVE YEAR PLAN

DISCIPLINARY THRUSTS	FY 87	FY 88	FY 89	FY 90	FY 91	EXPECTED RESULTS
AIRCRAFT AEROELASTICITY	ACTIVE CONTROL					ACTIVE/PASSIVE CONTROL OF AERO-ELASTIC RESPONSE  DATA BASE, NEW CONCEPTS/CONFIG.  FLUTTER FREE DESIGNS
	SIO/BUFFET/ARROW/BENCHMARK/NASP/ . . . LAUNCH VEHICLES					
	MILITARY/CIVIL FLUTTER CLEARANCE					
	TEST TECHNIQUES					
ROTORCRAFT AEROELASTICITY	PARAMETRIC STUDIES					REDUCED VIBRATION THROUGH PASSIVE CONTROL  ROTOR DESIGN FOR MINIMUM VIBRATION  NEW ROTOR CHARACTERISTICS EXPLORED
	MODAL TAILORING					
	AEROELASTICALLY OPTIMIZED ROTOR					
	NODALIZED					
	HINGELESS					
	MULTI-SPEED					
ROTORCRAFT STRUCTURAL DYNAMICS	NEW ROTOR CONCEPTS EVALUATIONS					SUPERIOR FEM CAPABILITY  INTEGRATED ROTOR/AIRFRAME ANAL. METHOD  ROTOR MODELING GUIDES
	BEARINGLESS					
	BASIC MODELING APPLICATIONS					
	DIFFICULT COMPONENT STUDIES, TEST/ANALYSIS					
	ADVANCED FEM TECHNIQUES					
AIRFRAME STRUCTURAL OPTIMIZATION						
COUPLED ROTOR AIRFRAME VIBRATIONS						
COMPOSITE BLADE ANALYSES						

Figure 6.

## IMPROVED ADAPTIVE SUPPRESSION SYSTEM EVALUATED IN TDT TESTS

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Configuration Aeroelasticity Branch  
Extension 2661

RTOP 505-63-21

**Objective** - Modern fighter aircraft carry a large variety of external wing-mounted stores. In some instances it is necessary to placard the operational envelope because of flutter. One approach to avoiding this restriction is to use an active control system to suppress flutter. Such a system operates by sensing wing motion and feeding back these signals through appropriate control laws to drive control surfaces to produce forces and moments to suppress flutter. Adaptive control systems are particularly attractive for this application because no knowledge of the store configuration being flown is required. Adaptive control systems continually measure system response due to control inputs and continually update control laws based on these measurements.

**Approach** - A 1/4-scale, cable-mounted, full-span F-16 aeroelastic model equipped with an Adaptive Flutter Suppression System (AFSS) was tested in the Langley Transonic Dynamics Tunnel (TDT) in a joint USAF/NASA/ General Dynamics test. A photograph of the model mounted in the wind tunnel is shown in figure 7(b). Accelerometers were used to measure wing motion. The AFSS digital computer was located in the control room. Computer generated signals were sent to the flaperons to provide a continuous low amplitude random excitation to the model. The resultant wing motion was measured and a mathematical representation of the model was determined by using an analysis implemented on the digital computer. Using this derived mathematical model a control law was then developed to suppress flutter.

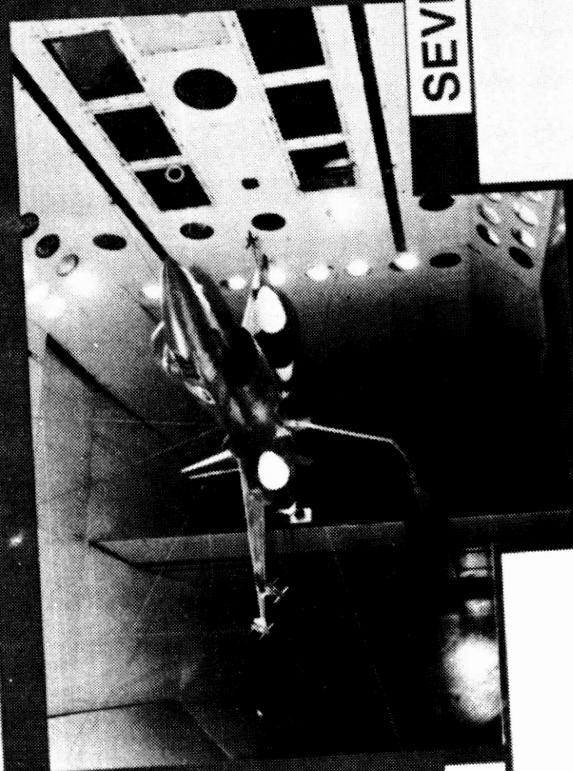
**Accomplishment Description** - Flutter data, AFSS on, and AFSS off, were obtained for three flutter critical store configurations. Basic systems-on testing began below the flutter boundary and proceeded above the system-off flutter boundary. The improved adaptive control law updated itself significantly faster than on a previous test, up to 2500 times during a test pass, without failing to suppress the flutter. For two of the store configurations (one with mild flutter onset, the other with moderate flutter onset) a 30-percent increase in flutter speed was demonstrated; for the third configuration (violent flutter onset) about a 20-percent increase in flutter speed was obtained. The results of the system for two store configurations is given on the figure. In addition to the basic testing, the system was evaluated using computer simulated store drops and actual store drops where wing tip missiles were ejected during testing causing the model to rapidly change from a stable flutter free configuration to an unstable flutter configuration. In these instances the AFSS quickly recognized that the dynamics of the model had changed and updated itself to suppress the flutter.

**Significance** - Adaptive flutter suppression with no predetermined control laws or knowledge of store configuration has been successfully demonstrated on a full-span, "free-flying" model. Data obtained during this study provides an invaluable foundation upon which to build future studies and flight applications.

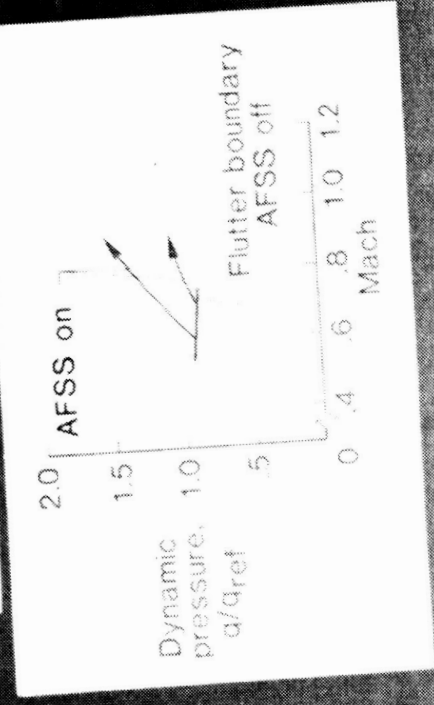
**Future Plans** - No additional wind-tunnel tests are planned.

Figure 7(a).

**IMPROVED ADAPTIVE FLUTTER SUPPRESSION SYSTEM EVALUATED IN TDT TESTS**



**MILD FLUTTER**



**SEVERE FLUTTER**

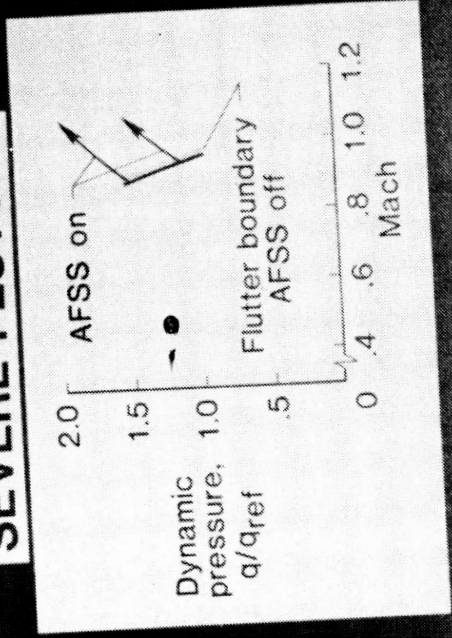


Figure 7(b).



## SOME EFFECTS OF SPEED BRAKES ON WING FLUTTER

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Extension 2661

RTOP 505-63-21

**Research Objective** - Speed brakes, spoilers, and other devices that are extended from the surface of a lifting surface have been used effectively in many stability and control applications in aeronautics. One use that has not been studied, however, is as a flutter suppressor. The purpose of this study was to obtain some parametric results of the effects of speed brake size and deployment angle on wing flutter.

**Approach** - A relatively simple, paddle type flutter model was equipped with a speed brake that could be deployed over a range of angles by adjusting a mechanical mechanism. The model (fig. 8(b)) was tested at Mach number (M) 0.80 in the Langley Transonic Dynamics Tunnel (TDT) with provisions for using speed brakes of varying sizes. That portion of the paddle model exposed to the flow was "rigid." Model stiffness was determined by the dimensions of the "paddle handle" that was behind the splitter plate which was used to get the wing root outside the wind-tunnel wall boundary layer. The bending and twisting of the handle provided the wing with flapping and pitching degrees of freedom. The paddle handle was shielded from the flow by a fairing. The model was ballasted so that the mass and inertia did not change as speed brake parameters were varied, thus, the natural frequencies remained the same such that flutter characteristics could be directly attributed to aerodynamic effects of the speed brakes.

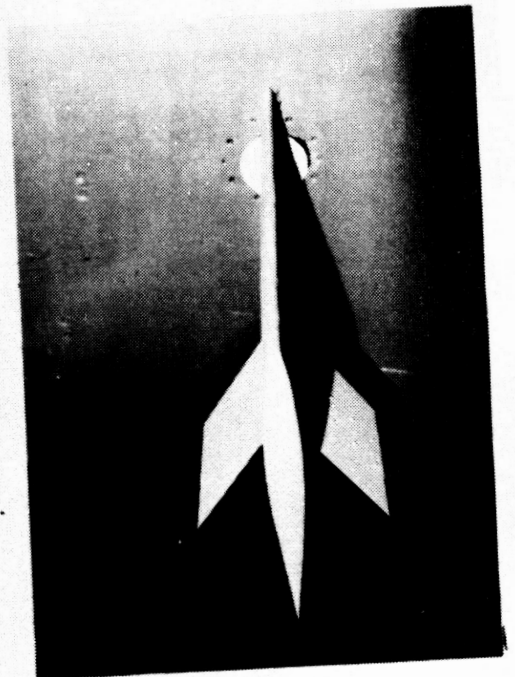
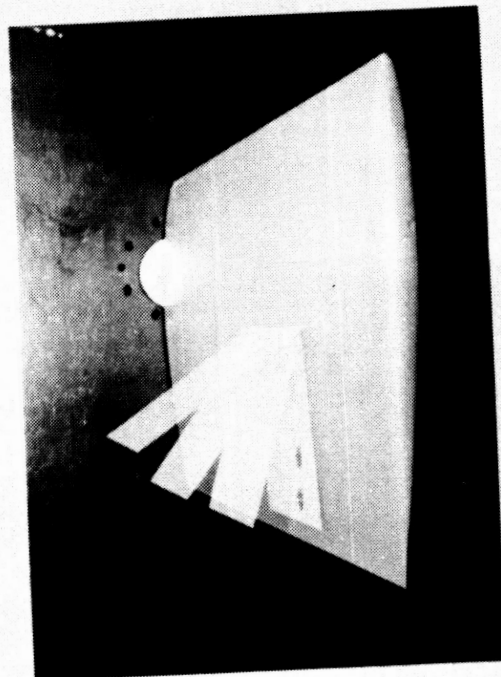
**Accomplishment Description** - Experimental flutter results were obtained at  $M = 0.80$  for variations in speed brake deployment angle for a constant speed brake size and for variations in speed brake size for a given deployment angle. These results are shown in the figure as the relative variation of flutter dynamic pressure with the respective parameter. These results show that the flutter dynamic pressure is increased by increasing either deployment angle or size and indicate that a significant deployment angle is required before the flutter speed is affected. Further, the data show that speed brake size has a stronger effect on flutter than does deployment angle over the range of variables studied.

**Significance** - The data give a basis to evaluate the effect speed brakes have on suppressing lifting surface flutter.

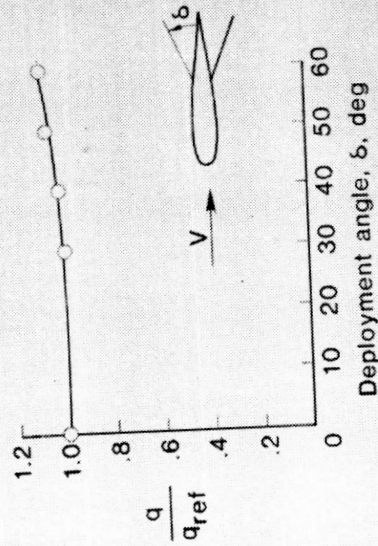
**Future Plans** - The results from this study will be documented in a formal NASA publication.

Figure 8(a).

# SOME EFFECTS OF SPEED BRAKES ON WING FLUTTER ( $M = 0.80$ )



DEPLOYMENT ANGLE EFFECTS  
( $AB/AW = 0.047$ )



BRAKE SIZE EFFECTS  
( $\delta = 40^\circ$ )

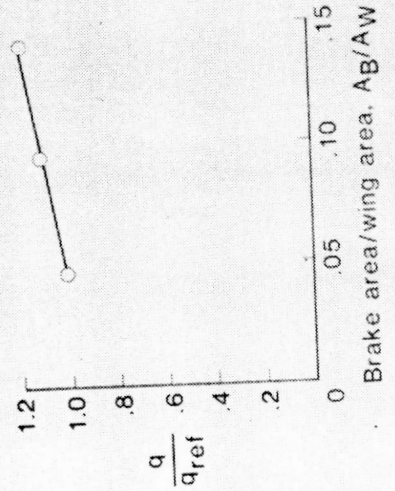


Figure 8(b).

# EFFECTS OF MACH NUMBER ON BUFFET RESPONSE OF A TWIN VERTICAL TAIL AIRPLANE

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Extension 2661

RTOP 505-63-21

**Research Objective** - Recent experiences from the operational use of high performance, twin vertical tail airplane configurations have shown that relatively large dynamic response of the tail structure occurs at certain often-encountered flight conditions. These buffet like responses are larger than those anticipated in the structural design and can have a significant adverse effect on service life. The objective of this study was to obtain data that can be used to better understand the characteristics of these undesirable responses.

**Approach** - A full-span, "rigid," sting mounted model of a high performance twin vertical tail airplane was equipped with elastic vertical tail and buffet tested over a range of angles of attack and Mach numbers in the TDT (fig. 9(b)). Although the elastic tails did not scale the dynamic characteristics of a specific full scale design, their stiffness and mass were chosen so that the dynamics characteristics were representative.

**Accomplishment Description** - Experimental buffet response data are shown in the figure as the variation of a normalized root-mean square bending moment response parameter with angle of attack for several different Mach numbers (M). The commonly-used response parameter is derived from generalized harmonic analysis considerations. It is assumed that the aerodynamic damping is very small compared to the structural damping, a responsible assumption because the tails were at near zero lift during the test. The response was primarily in one structural mode as shown by the spectra. The data for all Mach numbers is similar in that the bending moment is small and relatively constant up to an angle of attack of about 15°, where a relatively sharp increase in bending moment begins to occur. Although the details of the data are different for different Mach numbers, it does appear that a peak response occurs in the neighborhood of about 30-35 degrees angle of attack. The magnitude of the maximum value appears to be a function of Mach number.

**Significance** - These data provide a basis for assessing the Mach number effects on the buffet characteristics to twin vertical tail airplane configurations.

**Future Plans** - The results from this study will be documented in a formal NASA publication.

Figure 9(a).

**EFFECTS OF MACH NUMBER  
ON BUFFET RESPONSE  
OF TWIN VERTICAL TAIL  
( $Q = 50$  PSF)**

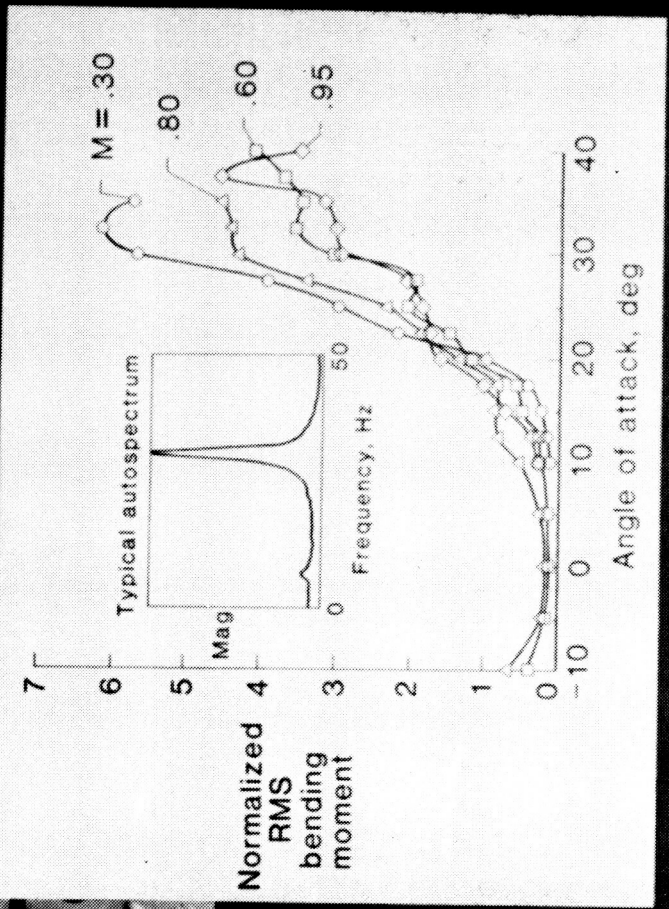
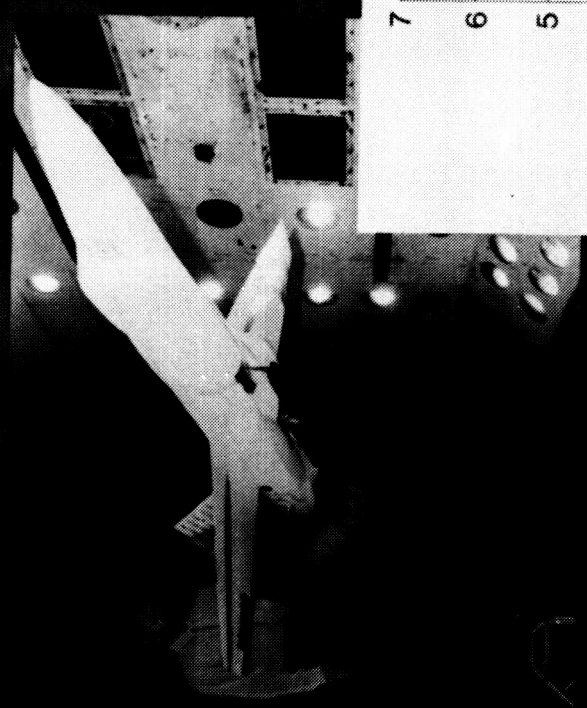


Figure 9(b).

## ACTIVE FLEXIBLE WING MODEL SUCCESSFULLY TESTED IN TDT

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RTOP 505-63-21

**Research Objective** - Utilizing wing flexibility with active controlled aerodynamic surfaces to provide control power is a technological concept that would be a useful capability for an advanced high performance fighter vehicle. This concept would use a novel combination of structural tailoring and multiple active control surfaces to provide variations in wing shape for maneuver load and roll control. The objective was to evaluate experimentally the use of this concept.

**Approach** - A 1/6-scale, full-span, active flexible wing model was designed and built by Rockwell International (RI) and then tested in the TDT as part of a cooperative program with RI, AFWAL, and NASA. This model was mounted on a sting so that the complete model had freedom to roll about an axis near the sting centerline (fig. 10(b)). An onboard system was used for remotely adjusting the angle of attack. Model loads were measured using a six-component balance. For safety, a hydraulic brake located on the sting at the rear of the model was used to prevent excessive roll motion. The wings of the model were aeroelastically tailored to obtain the desired wing structural stiffness properties. The model had eight large control surfaces, two each on the leading and trailing edges of both wing panels.

**Accomplishment Description** - Maneuver load control and roll control data were obtained at several wind-tunnel conditions using control laws implemented on a digital computer. These control laws were designed using control effectiveness and transfer function data obtained during a previously conducted TDT test using this same model. During the present test the data obtained clearly showed that the maneuver load characteristics of the model were significantly improved by using control laws developed by RI. In addition, control laws designed by both RI and NASA considerably enhanced the roll characteristics over those of the unaugmented model.

**Significance** - These tests showed that the active flexible wing concept is a viable one for enhancing the maneuver and roll control characteristics of advanced highly maneuverable airplanes.

**Future Plans** - Two more wind-tunnel entries are planned. NASA, with support from RI, will be designing and testing active flutter suppression systems to be used in conjunction with maneuver load control and active roll control systems.

Figure 10(a).

**ACTIVE FLEXIBLE WING MODEL  
SUCCESSFULLY TESTED IN TDT**

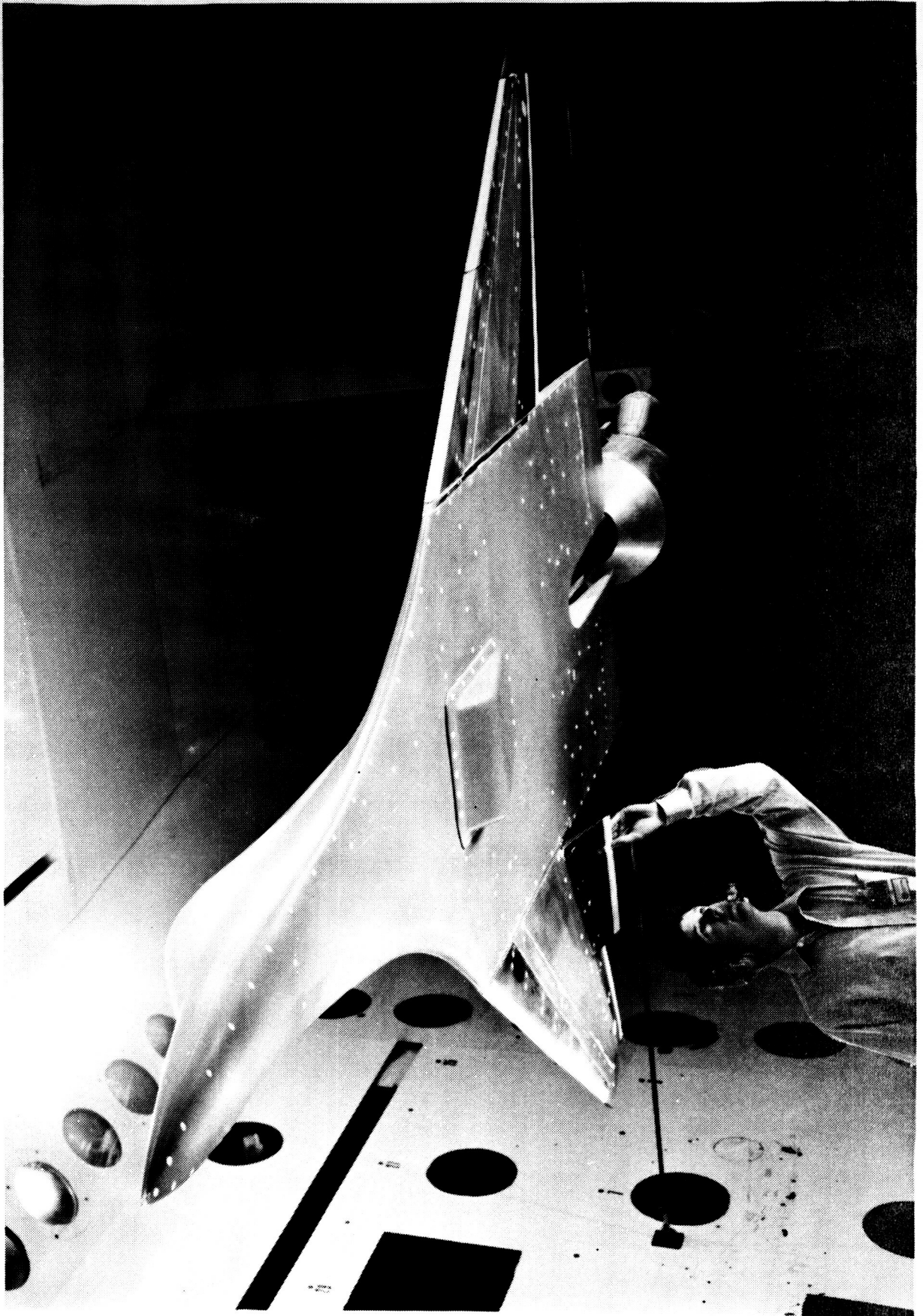


Figure 10(b).

# ROTOR TRACKING SENSITIVITY INVESTIGATED EXPERIMENTALLY

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RTOP 505-63-51

**Research Objective** - To minimize once-per-revolution (1P) vibration levels on conventional helicopters, it is desirable to operate the main rotor with all blades flying in an "in track" condition, i.e. with the individual tip paths of each blade coincident about the rotor azimuth. Until recently, emphasis on tracking research within the rotorcraft industry has been placed primarily on correction of track problems after they have been encountered rather than identifying and preventing the most significant causes of these problems. Therefore, an effort has been made to parametrically evaluate the tracking sensitivity to various blade design variables in a controlled experimental environment. These results will be used to develop analytical methods needed to predict tracking behavior and prevent major tracking problems from occurring in new rotor designs.

**Approach** - First, a series of hover tests were conducted using the Aeroelastic Rotor Experimental System (ARES) in the General Rotor Aeroelastic Laboratory (GRAL). Hover tests were conducted on five separate rotor configurations. ARES model hardware was a conventional four-bladed articulated rotor hub and two blade sets. All blades were of rectangular planform, were untwisted, and had a symmetrical NACA 0012 airfoil. Blade Lock number (a mass ratio-like parameter important to rotor blade scaling), section chordwise centers of gravity, and section torsional inertias could be changed for each case by varying the distributions of tungsten and aluminum counterweights inside the blades. For each test configuration, measurements of blade coning angles, rotor thrust, and torque were taken. With these data, a comparative assessment of the parametric effect of known changes in blade inertial properties on coning could be made. A structurally stiff aluminum spar blade set was used to examine the effect of a gross change in blade torsional stiffness on coning. The sensitivity of blade track to incremental adjustments of blade root pitch was also examined for each configuration. Pretest analyses of coning behavior and performance were conducted using the Comprehensive Analytical Model of Rotorcraft Aerodynamics and Dynamics (CAMRAD) computer code for comparison with the experimental results.

**Accomplishment Description** - Examples of hover data obtained in these tests and analytical results obtained by using CAMRAD are shown in figure 11(b). The example on the left shows the variation in coning angle with thrust coefficient for a high flapping inertia (low Lock number, case A in figure) configuration and for a low flapping inertia (high Lock number, case B) configuration. Although predicted coning magnitudes are somewhat larger than the observed values, the proportional difference in coning angle between the two cases as indicated by CAMRAD is in approximate agreement with the observed difference. In general, CAMRAD was useful in predicting gross coning trends. However, detailed effects observed experimentally were not predicted with CAMRAD. In the example at the right, data are presented where a single rotor blade was driven out of track by mechanically offsetting the blade root pitch by two known amounts. This was done with adjustments to the blade pitch link. During a test run the difference between the coning angle of the out-of-track blade and a reference blade is recorded. This allows an estimate to be made of the relative sensitivity of blade track to pitch link adjustment. Similar results were obtained for other configurations. Actual out of track cases such as these cannot be directly modeled with CAMRAD; therefore, no attempt to correlate these data with analysis has yet been made.

**Significance** - This study represents an important first step in developing means for improving rotor blade track and balance procedures.

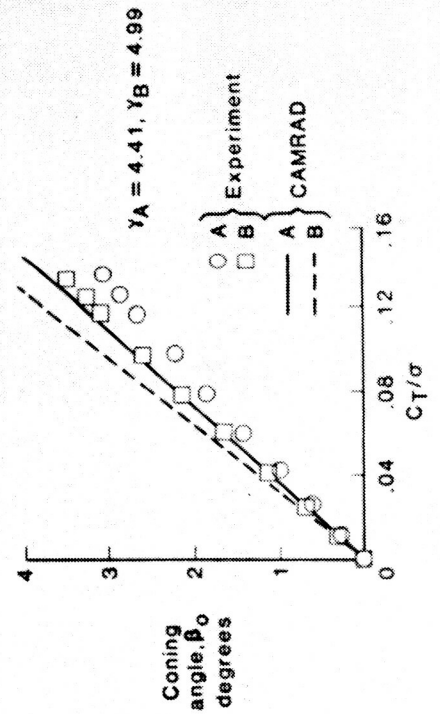
**Future Plans** - A test in the TDT to examine higher order tracking effects in forward flight is being planned for early 1988. Additional hover testing in the GRAL to examine some observed coning trends in greater detail and to investigate trailing edge tab effects on tracking is also planned. Results of this research will be documented in a NASA formal publication.

Figure 11(a).

# ROTOR TRACKING SENSITIVITY INVESTIGATED EXPERIMENTALLY



CONING ( $\beta_0$ ) VS. GROSS CHANGE  
IN BLADE LOCK NUMBER ( $\gamma$ )



CONING SENSITIVITY TO BLADE  
PITCH LINK ADJUSTMENTS ( $\Delta\theta$ )

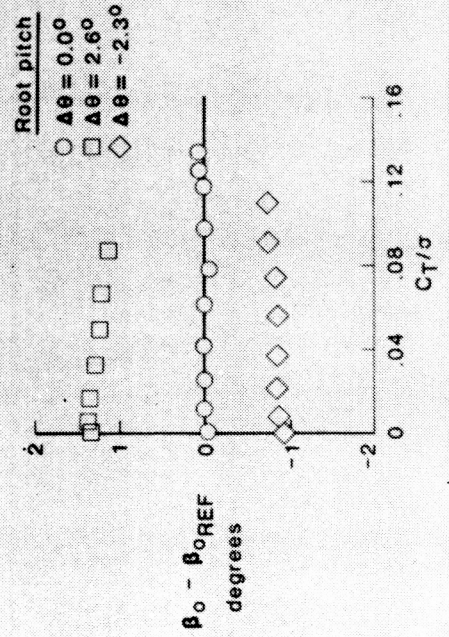


Figure 11(b).



# NEW STRUCTURAL DESIGN METHODOLOGY PRODUCES LIGHTER COMPOSITE MAIN ROTOR BLADE DESIGNS

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Extension 2661

RTOP 505-63-51

**Research Objective** - Composite rotor blades have demonstrated improvements over metal blades in fatigue strength, damage tolerance, corrosion resistance, and life-cycle costs. However, these improvements were gained with little regard to the tailorability aspects of composite materials. Design methodologies which do not take advantage of composite tailoring may overlook potential advances in blade technology. The objective of this research was to develop a methodology to perform minimum weight structural design of composite main rotor blades subject to aerodynamic performance, material strength, autorotation, and frequency constraints.

**Approach** - The methodology was developed to take advantage of composite tailoring, and therefore uses composite-material related design variables such as ply orientation and ply thickness. The constraints shown in figure 12(b) were defined using U. S. Army military specifications as defined in MIL-S-8698 with strength being the principle driver. Design exercises were performed to validate the methodology.

**Accomplishment Description** - Rotor blade designs based on the aerodynamics of the UH-60A rotor blade were developed to demonstrate the design methodology. The first design represented a single titanium-spar UH-60A type cross section. In this metallic-blade design exercise, the composite-material related design variables were not incorporated so results could be compared to past design procedures. The figure shows this metallic design blade using the new design methodology and compares its weight with the actual UH-60A blade. Because the two weights are very similar, the results demonstrated that the design methodology can lead to blade designs similar to those produced with conventional design procedures and also that the UH-60A is nearly an optimum design. The second design, labeled composite design in the figure, used a single graphite/epoxy spar with design variables of spar thickness and ply orientation. A significant weight savings, about 21 percent, was achieved over the metallic design.

**Significance** - Results of this work suggest that a design methodology which explicitly uses composite material related design variables can yield rotor blade designs with significant weight savings over those produced by conventional methodologies.

**Future Plans** - This work was performed as a precursor to development of the structural portion of a rotor blade design optimization package. The methodology and associated analysis will eventually be tied to other analyses which define optimum aerodynamics and dynamics for rotor blade systems. Structural-coupling concepts will also be included in future work.

Figure 12(a).

# NEW STRUCTURAL DESIGN METHODOLOGY PRODUCES LIGHTER COMPOSITE MAIN ROTOR BLADE DESIGNS

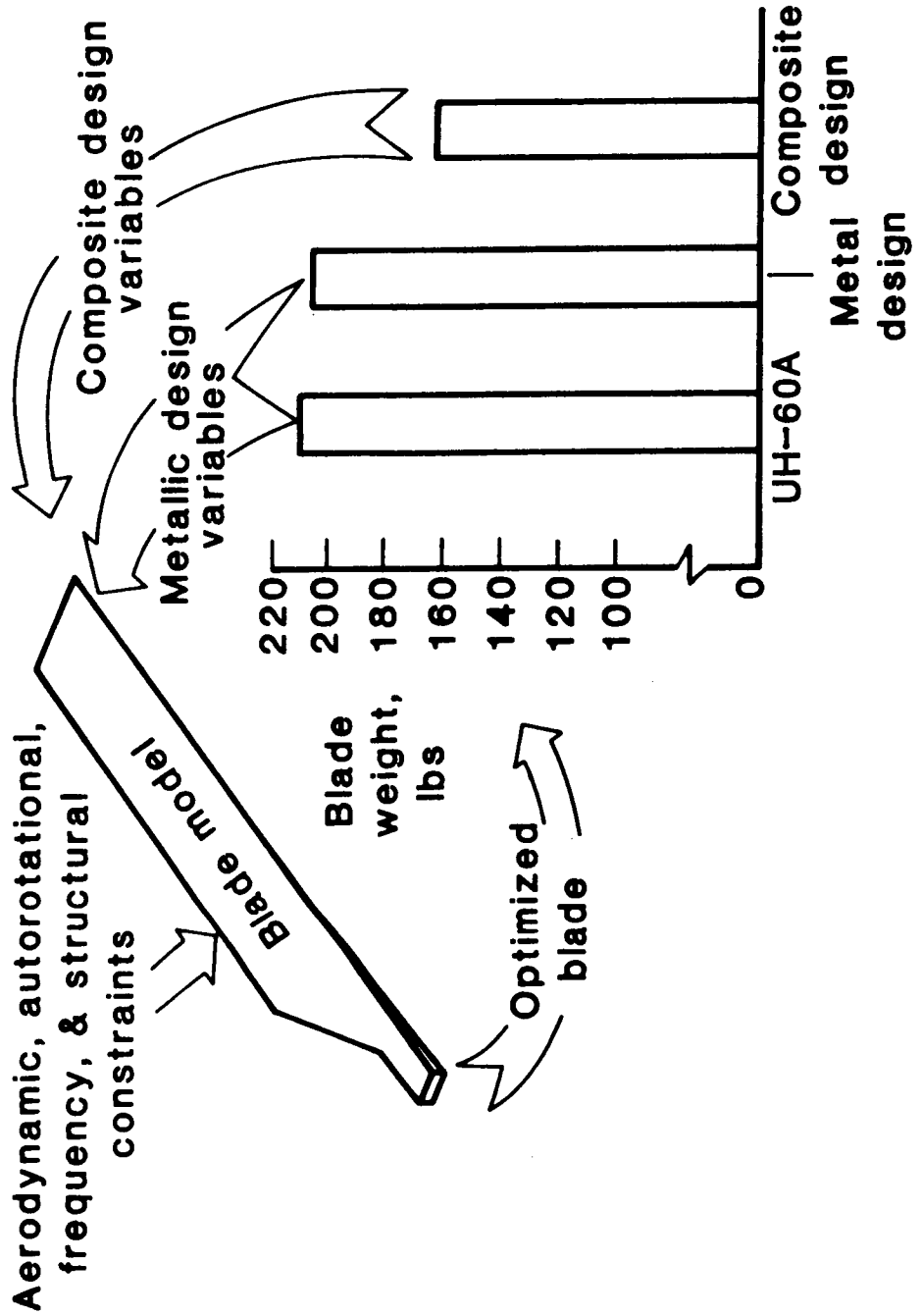


Figure 12(b).

# EXTENSION-TWIST-COUPLED ROTOR BLADE CONCEPT SHOWN TO BE STRUCTURALLY FEASIBLE

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RTOP 505-63-51

**Research Objective** - Currently a design compromise exists in the rotor system of a typical tilt-rotor aircraft because of the difference between forward flight (airplane) and hover (helicopter). The blade design for a tilt rotor has a twist distribution which produces acceptable hover performance at the cost of a lower flight propulsive efficiency. One solution to this problem is to vary the blade twist between airplane and helicopter modes. This could be done by using an extension-twist-coupled rotor blade. The objective was to show that the twist variations needed to get a significant aerodynamic performance payoff in an extension-twist-coupled rotor blade could be gotten within material design limits.

**Approach** - The structural feasibility was determined as follows: (1) Determine the maximum twist deformation to give the needed twist changes between helicopter and forward flight modes for a full-scale design; (2) Fabricate specimens using material, wall thickness, and laminate to simulate a full-scale rotor blade cross section and achieve desired twist changes; (3) Perform static tension tests to determine twist deformations as a function of axial load; (4) Determine the twist deformation at the design load and compare to the required twist deformations for a full-scale blade.

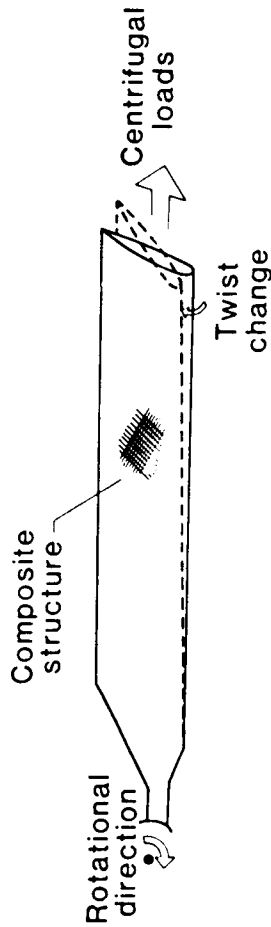
**Accomplishment Description** - An extension-twist-coupled tilt-rotor blade design was developed based on a theoretical optimum-twist model. The design showed that the maximum twist deformation required in the rotor blade was on the order of 0.4 deg/inch. Static tube test specimens were fabricated using T300/5208 Gr/E and IM6/R6376 Gr/E materials. The specimens had a 1.58 inch diameter with a .048 inch wall thickness, and consisted of [(+40/+50)2]S and [(+20/-70)2]S layouts. One set of results for the [(+20/-70)2]S specimens is shown in figure 13(b). The graph is a plot of twist rate as a function of axial load, and shows that at the design limit loads the corresponding twist rates are at and above 0.4 deg/inch desired for the theoretical tilt-rotor blade.

**Significance** - Results of this work have shown that the aerodynamically beneficial extension-twist-coupled rotor blade concept is structurally feasible statically.

**Future Plans** - This work will be continued using static specimens of other cross-section-designs and work will focus on producing an extension-twist-coupled model rotor blade design which is both structurally and dynamically feasible.

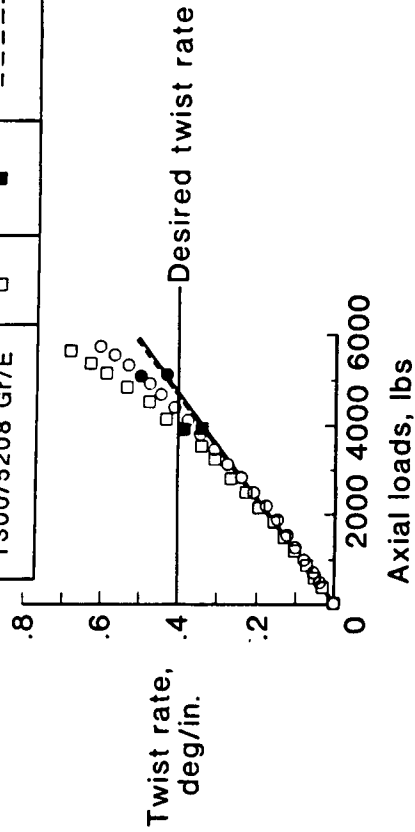
Figure 13(a).

# EXTENSION-TWIST-COUPLED ROTOR BLADE CONCEPT SHOWN TO BE STRUCTURALLY FEASIBLE



## TWIST COMPARISON OF [(+20/-70)2]S SPECIMENS

MATERIAL	TEST	DESIGN LIMIT	ANALYSIS
IM6/R6376 Gr/E	○	●	—
T300/5208 Gr/E	□	■	- - - -



## TILT ROTOR PERFORMANCE TRENDS

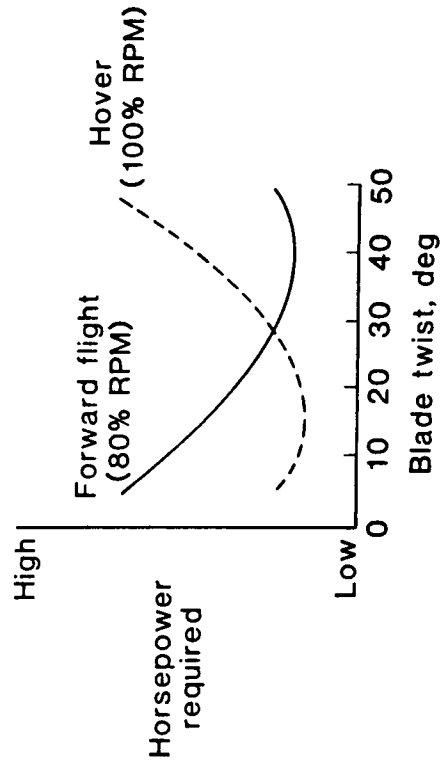


Figure 13(b).

# INITIAL EVALUATION OF INDUSTRY CODES FOR CALCULATION OF COUPLED ROTOR-AIRFRAME VIBRATIONS COMPLETED

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RTOP 505-63-51

**Research Objective** - One of the critical considerations in the design of any new helicopter is that of vibrations. However, U. S. helicopter companies have rarely relied on analysis during design in their efforts to limit vibration. With only a few exceptions, helicopters have been designed to performance requirements with past experience with vibrations taken into account, and new problems with vibration have been solved during flight tests and operation. Typical practice in those (few) instances when vibrations analysis was used has been to separately compute rotor vibratory loads & then apply them to an analytical (finite element) model of the airframe for determining airframe responses. Because of increasing demands for further reductions in vibrations to achieve the goal of a "jet smooth" ride, it is now recognized that analysis methods which accurately account for the coupling between the rotor and the airframe in producing vibrations must be employed in 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 activities being conducted under the DAMVIBS program is an activity which is aimed at evaluating existing analysis methods for calculating coupled rotor-airframe vibrations for the purpose of supporting helicopter airframe design work.

**Approach** - Industry teams from each of the four major helicopter airframe manufacturers will each separately and independently apply different analysis methods, one method per company, to calculate flight vibration levels of an existing helicopter and will each compare the analysis results with existing flight vibrations data. In all cases the (different) analysis methods will be applied to the same helicopter, and the comparisons will be made for the same (level, steady-state) flight conditions. An existing validated NASTRAN finite element model of the subject airframe, adjusted to correspond to the flight conditions for which comparisons are required, would be furnished by the manufacturer to the other participants as part of the common data to be utilized for the subject activity.

**Accomplishment Description** - In the initial effort in this area Bell, Boeing Vertol, McDonnell-Douglas, and Sikorsky have applied existing company-developed methods to calculate the coupled vibrations of the Bell AH-1G helicopter. Comparisons were also made with existing data from an Army Operational Loads Survey. The exercise on the AH-1G has been completed and the results have been presented at the NASA/industry meetings held under the DAMVIBS program. Draft final reports have been submitted and are under NASA review. An illustrative example of the type of results obtained from this study is given in figure 14(b) which shows a comparison of the calculated and measured 2/rev (two times the rotor rotational speed) vertical and lateral vibrations as a function of airspeed for two locations on the airframe. The analytical results obtained by the four companies for the 2/rev vibrations are in fair to poor agreement with measured data. (It should be noted that 2/rev is the primary main rotor excitation in the airframe.) In general, best agreement was obtained for vertical vibrations; the worst for lateral vibrations.

**Significance** - The subject studies on the AH-1G represent the first comparative evaluation of industry codes applicable to computation of coupled rotor-airframe vibrations. The results obtained indicate that current methods of analysis do not yield the level of accuracy needed to rely on analytical predictions of vibration during design. It is clear that further work is needed in this area. In particular, there is a need to identify and understand the reasons for the unacceptable predictions, to reconcile them and, ultimately, to correct the deficiencies causing them.

**Future Plans** - The work on evaluating existing methods of analysis for computing coupled rotor-airframe vibrations is continuing through a combination of in-house and contractual studies.

Figure 14(a).

# AH-1G VIBRATORY ACCELERATIONS

(FLIGHT VS. COUPLED ROTOR/AIRFRAME ANALYSIS)

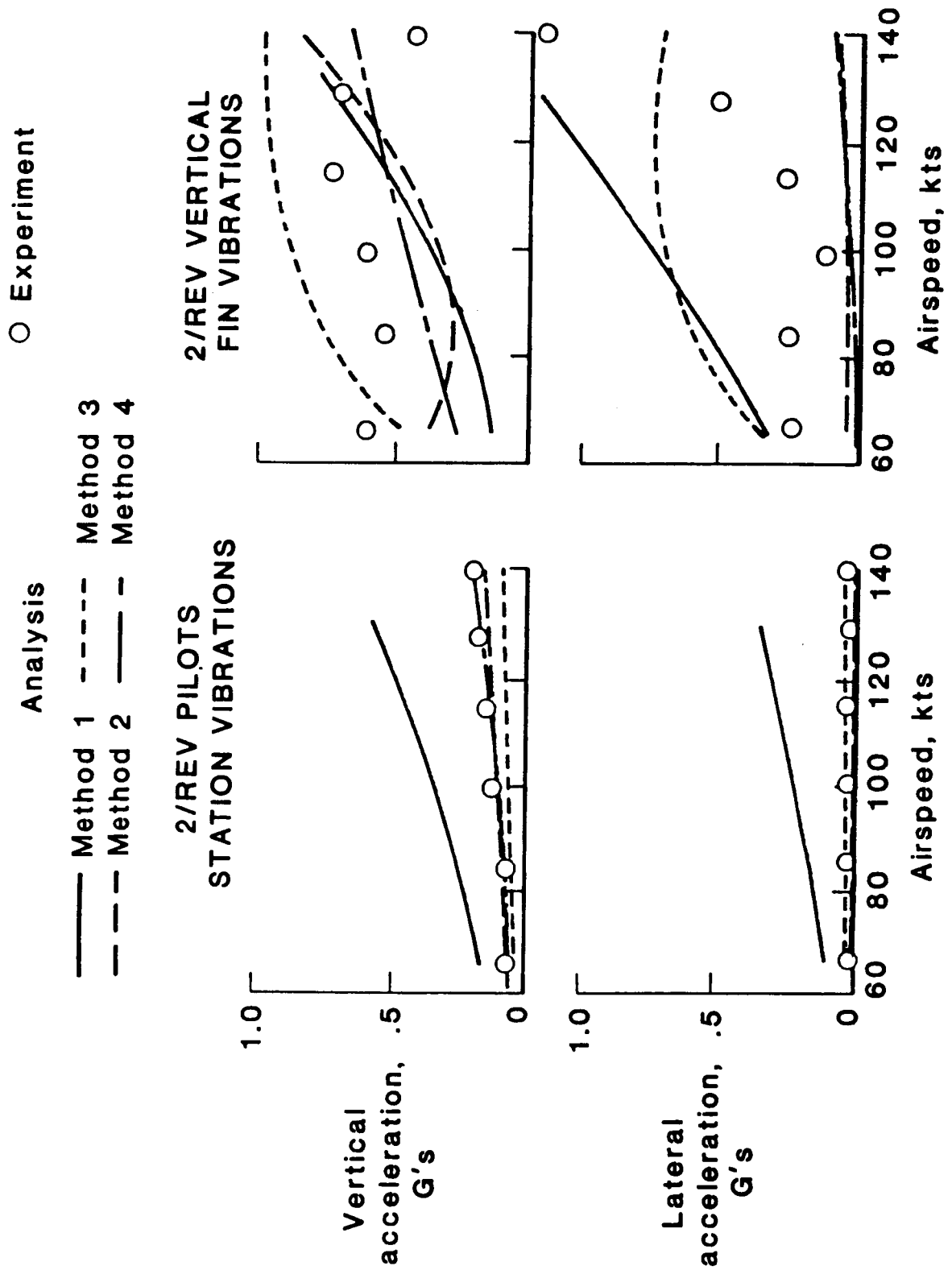
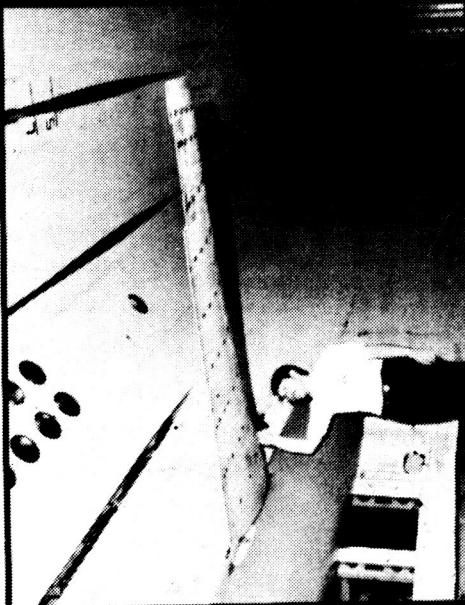


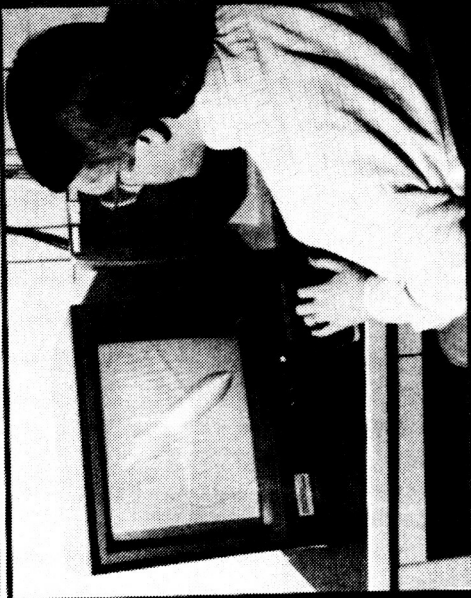
Figure 14(b).

# UNSTEADY AERODYNAMICS BRANCH

Experimental Aerodynamics



Computational Aeroelasticity



Unsteady Flowfield Analysis



Complete Vehicle Analysis

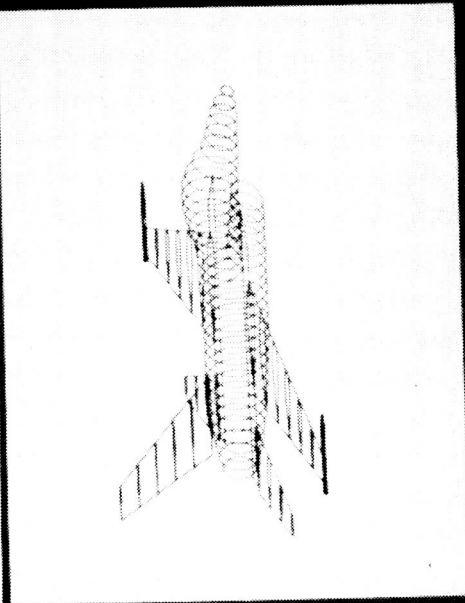


Figure 15.

# UNSTEADY AERODYNAMICS FIVE YEAR PLAN

DISCIPLINARY THRUSTS	FY 87	FY 88	FY 89	FY 90	FY 91	EXPECTED RESULTS
THEORY DEVELOPMENT	<b>COMPLETE AIRCRAFT ANALYSIS</b> attached flows, CAP-TSD shock-induced separation, DAST ARW-2 wing-store LCO EE/NS, CFL3D hypersonic marching code					Validated Predictions of Aeroelastic Response for Conventional and Novel A/C
	<b>HIGH ALPHA VORTEX FLOWS</b> vortex-structure instability, B-1 vortex instability, CFL3D dynamic loads buffet flows					
DESIGN METHODS	<b>EFFICIENT AEROELASTIC ANALYSIS</b> hinge moments/control effectiveness zonal methods computational sensitivity					Accurate and Affordable Aeroelastic Analysis Tools
	<b>UNSTEADY PRESSURES</b> canard/wing interference high Re cryogenic tests					
EXPERIMENT	<b>BASIC AEROELASTICITY BURT</b> installation & checkout wing-vortex interaction vortex induced buffet global imaging/code validation active control validation					Data for Code Validation
	<b>BENCHMARK MODELS</b> transonic flutter wing-store flutter					

ORIGINAL PAGE IS  
OF POOR QUALITY

Figure 16.



## DEVELOPMENT OF TRANSONIC AEROELASTICITY CODE FOR REALISTIC AIRCRAFT CONFIGURATIONS

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Extension 4236

RTOP 505-63-21

**Research Objective** - The objective of the research was to develop a computer code for transonic aeroelastic analysis of realistic complete aircraft configurations.

**Approach:** - A time-accurate approximate factorization (AF) algorithm has been implemented for solution of the unsteady small-disturbance potential equation for transonic flow. The new algorithm is very efficient for transonic unsteady aerodynamic and aeroelastic analyses when compared with the previously used alternating-direction implicit algorithm. A new computer code has subsequently been developed to fully exploit the computational efficiency and superior stability characteristics of the AF algorithm. The new code is called CAP-TSD which is an acronym for Computational Aeroelasticity Program - Transonic Small Disturbance. The CAP-TSD code is capable of treating complete aircraft geometries with multiple lifting surfaces and bodies.

**Accomplishment Description** - Calculations were performed for the General Dynamics one-ninth scale F-16C aircraft model. The F-16C is modeled using four lifting surfaces and two bodies as shown in the figure 17(b). The lifting surface include: (1) the wing with leading and trailing edge control surfaces, (2) the launcher, (3) a highly-swept strake, aft strake, and shelf surface, and (4) the horizontal tail. The bodies include: (1) the tip missile, and (2) the fuselage. Steady pressure distributions near the wing mid-semispan are shown in the left part of figure 17(c) for the F-16C aircraft at a freestream Mach number of  $M = 0.9$  and an angle of attack of  $2.38^\circ$ . For this case, there is a moderately strong shock wave on the upper surface of the wing and the CAP-TSD pressures agree well with the experimental data. Unsteady pressure distributions are shown in the right part of figure 17(c) for the entire F-16C aircraft undergoing a rigid pitching motion with an amplitude of  $0.5^\circ$  and a reduced frequency of  $k = 0.1$ . No unsteady data are available for code validation. For this case, there is a relatively large shock pulse in the real part of the wing upper surface pressures, which is of larger magnitude and located further downstream in the complete airplane model. The stronger shock on the wing is attributed to the accelerated flow about the fuselage and launcher/tip missile. The differences between complete airplane and wing alone results emphasize the importance of including all components in the calculation.

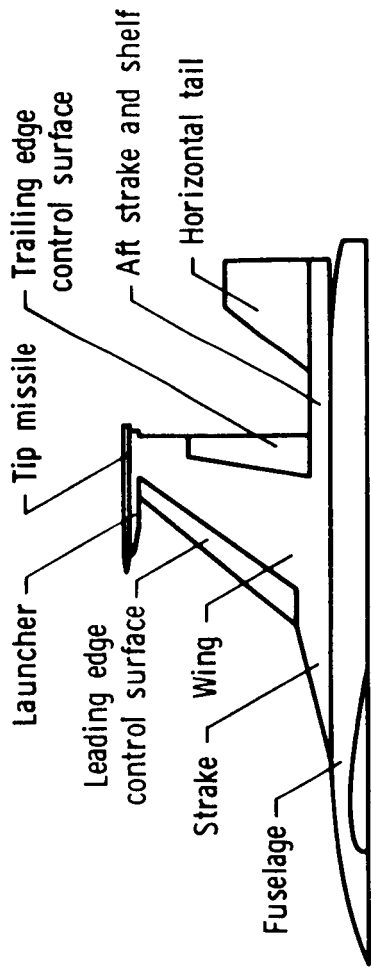
**Significance** - The capability permits the modeling of realistic complete aircraft configurations for transonic aeroelastic analyses and allows the prediction of component aerodynamic interference effects on aircraft unsteady airloads and flutter characteristics.

**Future Plans** - Further pressure correlations and aeroelastic calculations will be performed for complex configurations to further assess and validate the CAP-TSD code.

Figure 17(a).

DEVELOPMENT OF TRANSONIC AEROELASTICITY CODE FOR REALISTIC  
AIRCRAFT CONFIGURATIONS

- CAP-TSD MODELING OF GENERAL DYNAMICS ONE-NINTH SCALE F-16C AIRCRAFT MODEL



- F-16C MODELED USING FOUR LIFTING SURFACES AND TWO BODIES

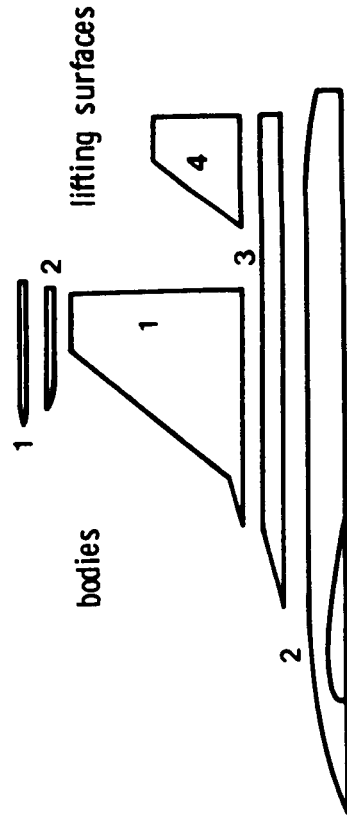


Figure 17(b).

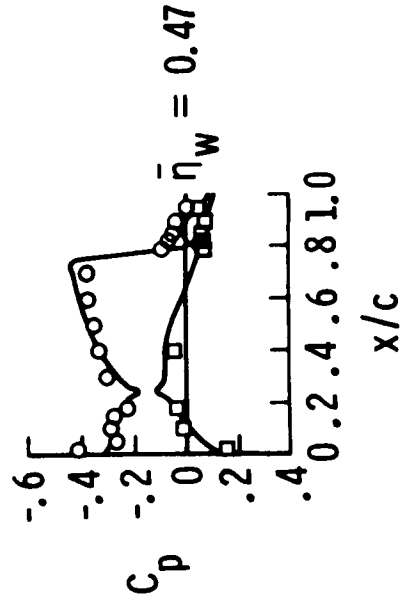
CAP-TSD CALCULATIONS FOR F-16C AIRCRAFT MODEL

- STEADY PRESSURE COMPARISON

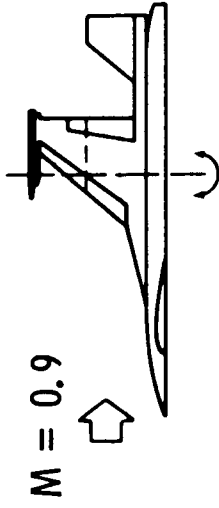


— CAP-TSD

- Upper surface } Experiment
- ◻ Lower surface }



- UPPER SURFACE UNSTEADY PRESSURE COMPARISON



— CAP-TSD

- Complete airplane
- - - - Wing alone

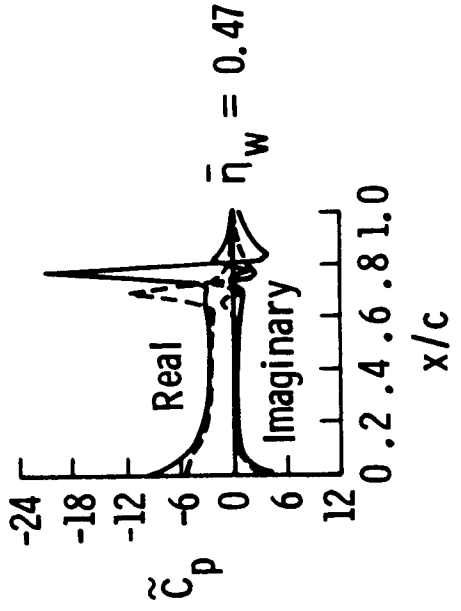


Figure 17(c).

# ALGORITHM MODIFICATIONS IMPROVE STABILITY AND ACCURACY OF CAP-TSD CODE

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RTOP 505-63-21

**Research Objective** - The objective was to improve the numerical stability and accuracy of a time-accurate approximate factorization (AF) algorithm for steady and unsteady transonic analysis of realistic aircraft configurations.

**Approach** - Algorithm modifications have been made to the CAP-TSD (Computational Aeroelasticity Program - Transonic Small Disturbance) code which was recently developed for aeroelastic analysis of complete aircraft configurations in the flutter-critical transonic speed range. The AF algorithm of the CAP-TSD code solves the unsteady transonic small-disturbance equation. The algorithm modifications include Engquist-Osher (E-O) type-dependent differencing to more efficiently treat regions of supersonic flow and extension of the E-O switch for second-order spatial accuracy to more accurately treat these regions.

**Accomplishment Description** - Results were obtained for the ONERA M6 wing to assess the algorithm modifications. The M6 wing has an aspect ratio of 3.8, a leading edge sweep angle of  $30^\circ$ , and a taper ratio of 0.562. The freestream Mach number was selected as  $M = 0.84$  and the wing was at  $3.06^\circ$  angle of attack. This rather difficult case could not be computed using the original Murman-Cole differencing. Comparisons of steady pressures computed using first-order and second-order accurate spatial differencing are presented in the left-half of figure 18(b), for span station  $\eta = 0.44$ . These comparisons show that the supersonic-to-supersonic shock near 15% chord is much more sharply captured by the second-order method and consequently the calculated pressures are in very good agreement with the experimental data. To demonstrate the robustness of the modified algorithm, an unsteady calculation was performed for the M6 wing at  $M = 0.84$ . The wing was forced to oscillate in pitch about the root midchord with a  $2^\circ$  peak-to-peak oscillation amplitude. The reduced frequency was selected as  $k = 0.1$  and only 300 steps per cycle of motion were used. Instantaneous pressure distributions at two points during the cycle are shown in the right-half of the figure for  $\eta = 0.44$ . The results illustrate the large shock motions that the modified algorithm is capable of computing. Both the supersonic-to-supersonic shock and the supersonic-to-subsonic shock oscillate over approximately 10% of the chord during a cycle of motion. The algorithm including second-order accurate differencing captures the shocks sharply and is sufficiently robust to compute this complex unsteady flow using only 300 steps per cycle.

**Significance** - The modified CAP-TSD code is now capable of accurately and efficiently computing unsteady flows involving complex shock structures and large shock motions.

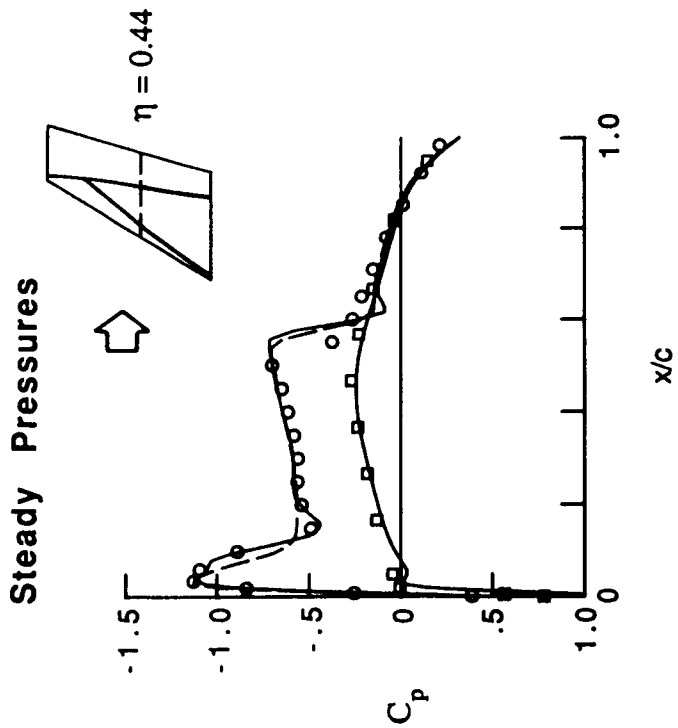
**Future Plans** - Further algorithm modifications are being implemented to more accurately treat cases with relatively strong shock waves.

ALGORITHM MODIFICATIONS IMPROVE STABILITY AND ACCURACY OF CAP-TSD CODE

- ENGQUIST-OSHER SWITCH ENABLES COMPUTATION FOR DIFFICULT TRANSONIC CASES
- SECOND-ORDER UPWIND DIFFERENCING IMPROVES ACCURACY OF SOLUTION

ONERA M6 Wing,  $M = 0.84$ ,  $\alpha = 3.06$  deg.

Experiment  
 ○ Upper  
 □ Lower



CAP-TSD  
 — Second-Order-Accurate  
 - - - First-Order-Accurate

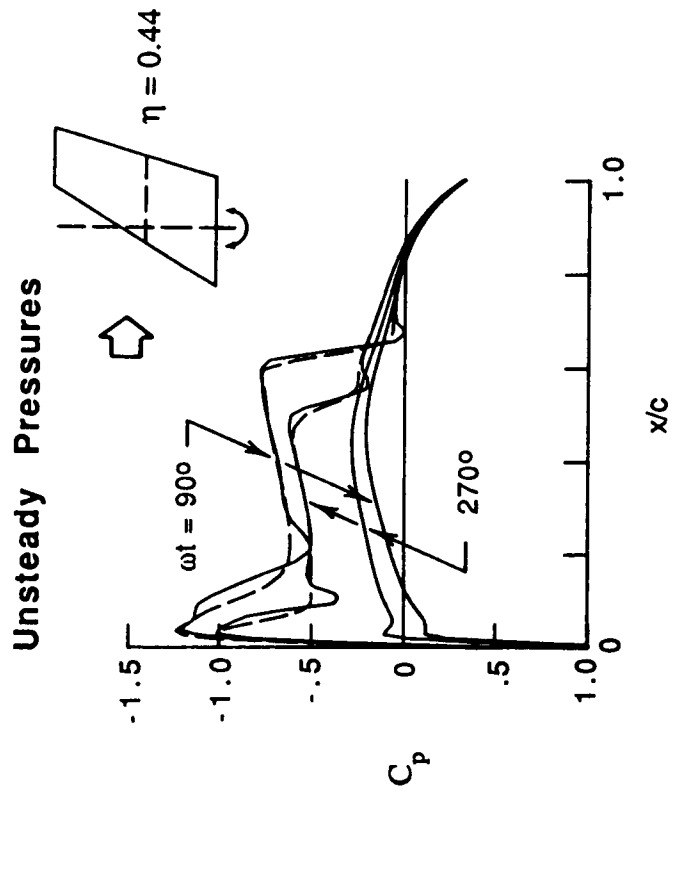


Figure 18(b).

# ENTROPY AND VORTICITY EFFECTS IMPROVE ACCURACY OF UNSTEADY TRANSONIC SMALL-DISTURBANCE (TSD) THEORY

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RTOP 505-63-21

**Research Objective** - The objective of the research was to modify unsteady transonic small-disturbance (TSD) theory to more accurately treat cases with relatively strong shock waves.

**Approach** - Modifications have been made to the CAP-TSD (Computational Aeroelasticity Program - Transonic Small Disturbance) code to more accurately treat attached flows with strong shock waves involving entropy and vorticity generation. The CAP-TSD code solves the transonic small-disturbance potential equation and was developed for aeroelastic analysis of complete aircraft configurations in the flutter-critical transonic speed range. The classical TSD theory is only applicable to cases involving mild shock waves. Thus to treat strong shocks, the code was modified to include the effects of shock-generated entropy and vorticity which are important for such cases. The modified code includes these effects while retaining the relative simplicity and cost efficiency of the TSD formulation.

**Accomplishment Description** - To test the modified TSD theory, calculations were performed for the ONERA M6 wing. The M6 wing has an aspect ratio of 3.8, a leading edge sweep angle of  $30^\circ$ , and a taper ratio of 0.562. Results were obtained at the freestream Mach number of  $M = 0.92$  with the wing at  $0^\circ$  angle of attack. Steady pressure distributions along three span stations of the wing are presented in figure 19(b). For this case, the flow is symmetric about the wing with shocks on the upper and lower surfaces. As shown in the left-half of the figure, the results from the unmodified theory indicate that the shock is located too far aft and is too strong outboard near the tip in comparison with an Euler calculation. When the entropy and vorticity corrections are included in the calculation, the shock is in excellent agreement with the Euler result in both strength and location, as shown in the right-half of the figure. Consequently, the steady pressure distributions from the modified TSD theory compare very well with the Euler pressures. The entropy correction alone produces a shock that is still downstream of the Euler location, approximately half-way between the two sets of CAP-TSD results shown in the figure. Therefore, both entropy and vorticity corrections are required to give Euler-like accuracy.

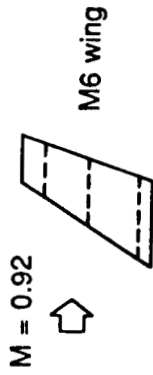
**Significance** - The capability now provides the aeroelastician an affordable method to analyze relatively difficult transonic cases consisting of attached predominantly streamwise flows with strong shocks, without resorting to solving the computationally more expensive Euler equations.

**Future Plans** - Further steady and unsteady results are being obtained to assess the modified theory. Comparisons with Euler results and experimental data will further determine the accuracy and efficiency of the CAP-TSD code including entropy and vorticity effects.

Figure 19(a).

# ENTROPY AND VORTICITY EFFECTS IMPROVE ACCURACY OF UNSTEADY TRANSONIC SMALL-DISTURBANCE (TSD) THEORY

- UNMODIFIED THEORY GIVES INACCURATE SHOCK PREDICTION
- ENTROPY AND VORTICITY CORRECTIONS YIELD EULER-LIKE RESULTS FOR STRONG-SHOCK CASE



— CAP-TSD  
 o Euler

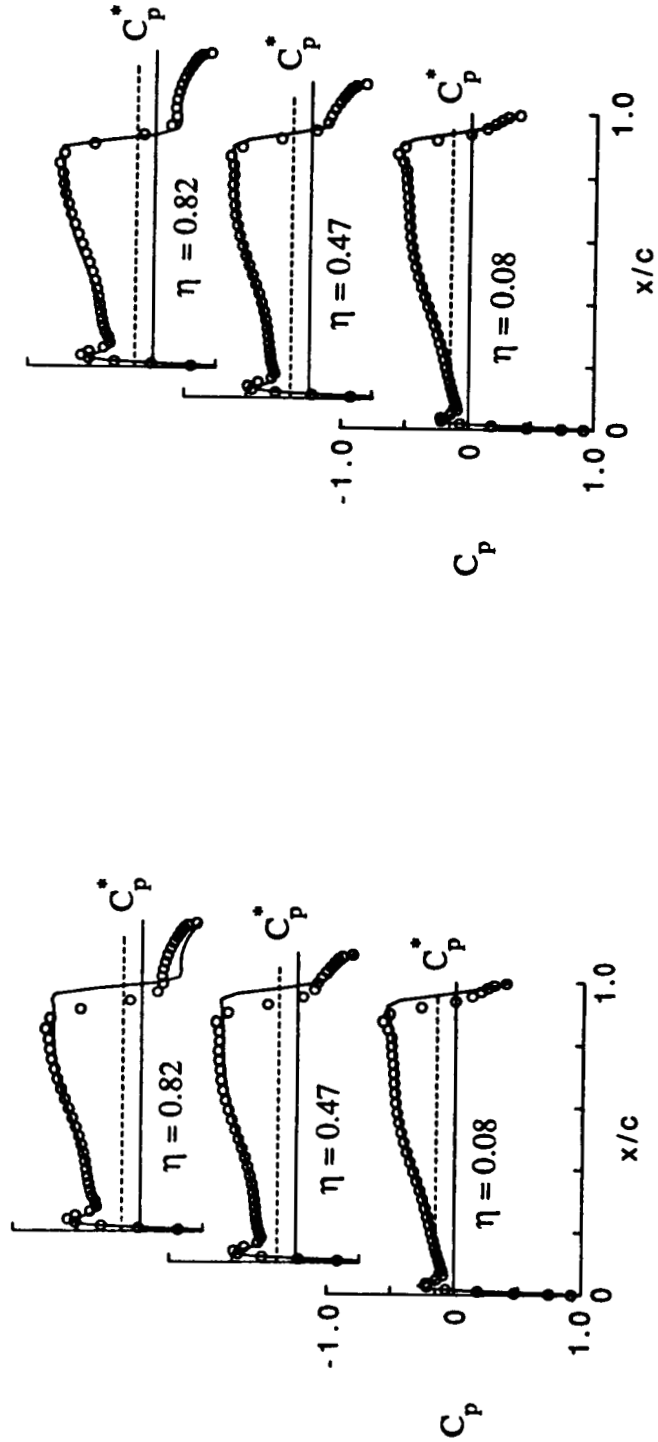


Figure 19(b).

# TIME-ACCURATE EULER CODE DEVELOPED FOR VORTEX-DOMINATED FLOWS AROUND MANEUVERING AND DEFORMING WINGS

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RTOP 505-63-21

**Research Objective** - The objective of the research is to develop a time-accurate computer code for Unsteady Transonic Vortex-Dominated Flows around Maneuvering and Deforming Wings.

**Approach** - The Unsteady Euler equations have been derived with respect to a moving frame of reference which is attached to the wing. The resulting equations preserve the conservation form and reduce to the conservative form of the steady flow equations for a uniformly translating airfoil. The maneuvering motion of the wing is prescribed through the Eulerian angles and their rates of change. The deforming motion of the wing may be prescribed through the surface boundary condition and the deformation may be prescribed a priori or obtained from the solution of the aeroelastic equation. For a rigid wing, the grid is generated one time only without any need for dynamic grid computation. Two finite-volume Euler schemes have been developed, one is an explicit scheme and the other is an implicit scheme. The explicit scheme uses four-stage Runge-Kutta time stepping with second and fourth-order dissipation terms. The implicit scheme uses the spatial approximate factorization of Beam and Warming, and implicit second-order and explicit second-and fourth-order dissipation terms are added to the scheme.

**Accomplishment Description** - The explicit and implicit schemes have been programmed for the CDC VPS-32 computer. The explicit code has been applied to calculations for steady and unsteady flows about delta wings. Specific applications include: (1) subsonic, transonic, and supersonic steady three-dimensional flows around a sharp-edged delta wing, (2) steady supersonic conical flow around a uniformly rolling delta wing and (3) unsteady locally-conical flows for a rolling-oscillation of a delta wing. Figure 20(b) shows the computed surface pressure at a representative spanwise location for the latter case at four instants during a cycle of oscillation. The flow conditions are  $M = 2$ ,  $\alpha = 10$ , for this aspect ratio 1.4 wing and the wing is undergoing  $\pm 15^\circ$  roll oscillations at a reduced frequency of 1.337. The surface pressures show the evolution of the peaks of the unsteady vortex loading to be clockwise on the left wing panel and counterclockwise on the right wing panel. These differences are caused by the time delays involved in the vortex flows.

**Significance** - The present capabilities allow accurate and reliable prediction of steady and unsteady flows for a wide range of Mach numbers, angles of attack, reduced frequencies and aspect ratios with reasonable computational costs.

**Future Plans** - Our future plans call for further developments, extensions and applications which include, (1) three-dimensional transonic flow around pitching and maneuvering delta wings, (2) three-dimensional transonic flow around deforming delta wings, (3) solution of the unsteady vortex-wing structural dynamic interaction problem.

Figure 20(a).



# VORTEX FLOW CALCULATED FOR ROLLING DELTA WING

- Euler equations in moving, body-fixed system
- Sharp-edged delta wing at angle of attack
- Supersonic, quasi-conical flow
- Developing implicit, fully 3-D, transonic capability

70 deg. sweep,  $M = 2$ ,  $\alpha = 10$  deg

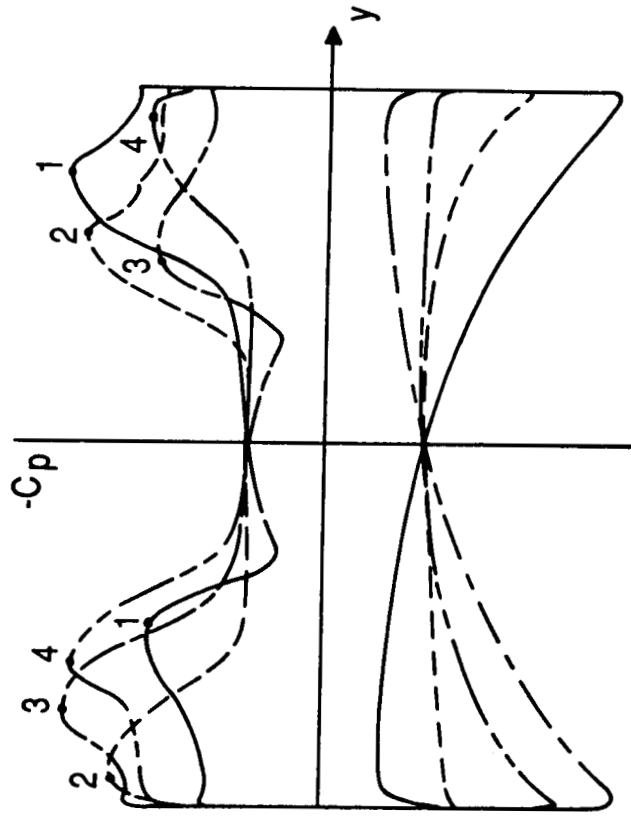
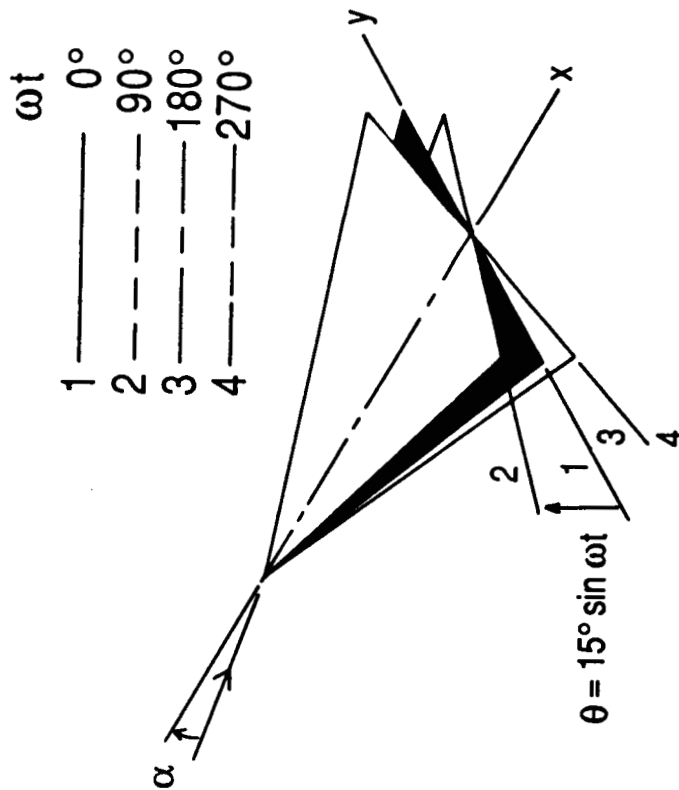


Figure 20(b).

# IMPROVED BOUNDARY LAYER METHOD FOR UNSTEADY POTENTIAL FLOW AGREES WITH NAVIER-STOKES SOLUTION

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RTOP 505-63-21

**Research Objective** - The objective of this research is to develop accurate techniques for predicting unsteady transonic airloads for flows with significant viscous effects. Computationally efficient techniques are sought for 2-D airfoil sections and fully 3-D configurations.

**Approach** - Unsteady transonic flow fields are routinely calculated with computer codes based upon Transonic Small Disturbance (TSD) potential theory. Results are quite accurate for thin bodies so long as viscous effects are small. As moderate strength shock waves develop, the inclusion of viscous effects is essential for accurate predictions of aerodynamic loading. For attached flows, techniques are available for interacting a boundary layer analysis (such as the integral method of Green) with the inviscid solution algorithm. For separated or nearly separated flows, this interactive approach fails due to a singularity in the boundary layer equations. This singularity can be circumvented by solving the boundary layer equations in inverse form as has been demonstrated by Carter and others. This improved interactive boundary layer technique has been implemented in the unsteady transonic small disturbance computer code XTRAN2L so that separated flows can be calculated with this code.

**Accomplishment Description** - A computer code incorporating the improved interactive boundary layer method has been developed and applied to several test cases. This code uses Carter's relaxation method to update the boundary layer displacement thickness. Figure 21(b) shows plots of lift and moment coefficients for a NACA 0012 airfoil oscillating in pitch about a mean angle of  $4.86^\circ$  with an amplitude of  $2.44^\circ$ . The flow is close to separation at the larger angles of attack. Transition is specified to be at 20% chord for the inverse boundary layer calculations. Calculated results with the improved interactive (inverse) boundary layer method agree very well with the CFL2D Navier-Stokes code calculations for both the lift and moment coefficients. Lift coefficients from both theories are in good agreement with the experimental results except for the higher angles of attack. Some discrepancies between calculated and experimental moment coefficient results are evident, although the improved boundary layer method agrees well with Navier-Stokes results. The differences between the calculated and experimental results are believed to be due to wind tunnel wall effects. The results demonstrate that the improved boundary layer method can accurately predict unsteady transonic flow fields for moderately thick airfoils at large angles of attack with incipient flow separation.

**Significance** - The close agreement indicates that transonic small disturbance theory combined with the improved interactive boundary layer method can be successfully applied well beyond the traditional limits associated with small disturbance assumptions. This increases the range of conditions which can be treated with the less expensive TSD methods, a significant advantage for applications to aeroelastic analysis where many conditions must be computed.

**Future Plans** - Efforts are underway to implement the improved boundary layer method in a stripwise manner in the 3-D code CAP-TSD and entropy effects are being included to allow calculations for cases involving strong shocks. Improvements to boundary layer parameters internal to the method are planned to make the boundary layer analysis more appropriate for separated flows. The results will be published in a NASA TP.

Figure 21(a).

# IMPROVED INTERACTIVE BOUNDARY LAYER METHOD AGREES WITH NAVIER-STOKES CALCULATIONS FOR INCIPIENT SEPARATION CASE

NACA 0012,  $M = 0.6$ ,  $k = 0.081$

- Experiment
- XTRAN2L, inviscid
- - - XTRAN2L w/inverse boundary layer
- CFL2D, thin-layer Navier-Stokes

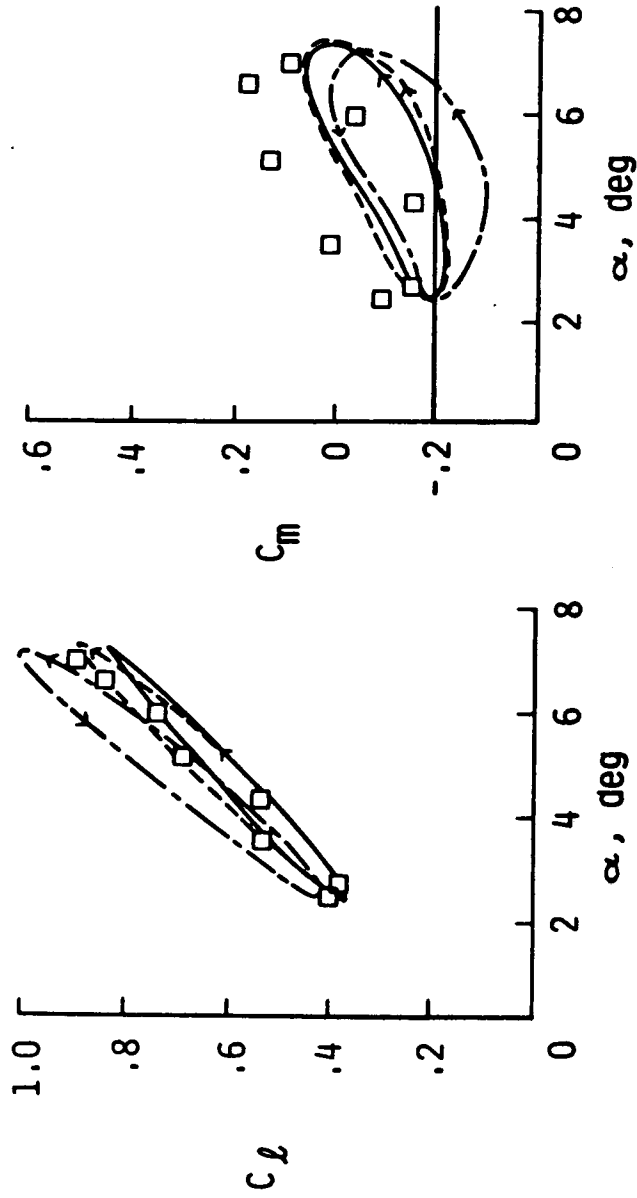


Figure 21(b).

# CAP-TSD RESULTS EXHIBIT GOOD AGREEMENT WITH EXPERIMENT FOR COMPLETE F-15

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RTOP 505-63-21

**Research Objective** - The objective of the research is to evaluate the capability of the CAP-TSD computer code for performing transonic aeroelastic analysis of realistic complete aircraft configurations.

**Approach** - A computer code has recently been developed to solve the unsteady transonic small-disturbance potential equation on complete aircraft geometries with multiple lifting surfaces and bodies. The code is called CAP-TSD which is an acronym for Computational Aeroelasticity Program - Transonic Small Disturbance. As part of a cooperative effort with McDonnell Aircraft Company, the CAP-TSD code is being applied to analyzing the steady and unsteady aerodynamic characteristics of several aircraft configurations. Comparisons between CAP-TSD and experimental results are being evaluated to assess the accuracy of the program for predicting transonic aeroelastic behavior of realistic aircraft configurations.

**Accomplishment Description** - Calculations were performed for the McDonnell F-15 aircraft. Two lifting surfaces are used to model the wing and the horizontal tail and two bodies are used to model the fuselage and the engine inlet as shown in the upper left of figure 22(b). A flow-through boundary condition is applied on the upstream side of the engine inlet to simulate the airflow through the inlet. The vertical tails shown in the figure were not modeled. Steady pressure distributions were calculated for the F-15 for freestream Mach numbers from 0.8 to 1.2 at angles of attack from 0° to 5°. Shown in the upper right is a color contour of the steady pressure distribution on the F-15 at a freestream Mach number of 0.9 and an angle of attack of 2.46°. The dark bands of color (blue - black) represent expansion of the flow (low pressure) while the light bands of color (red - white) represent compression of the flow (high pressure). A comparison with experimental chordwise steady pressures at 59% wing semispan is shown in the lower left.

**Significance** - The CAP-TSD predicted steady pressures compare well with the experimental data for the F-15. This good agreement was observed for the range of Mach numbers and angles of attack at which comparisons were made. This gives confidence that the aerodynamic characteristics of the F-15 aircraft will be correctly predicted when transonic aeroelastic flutter analyses are performed.

**Future Plans** - The research was conducted as part of an effort to evaluate the capability of CAP-TSD for performing transonic aeroelastic analysis of realistic aircraft. Now that the static aerodynamic predictions of CAP-TSD have been validated, the program will be used to predict flutter boundaries for the F-15. Comparisons will be made with available experimental data to assess the program's capability.

Figure 22(a).

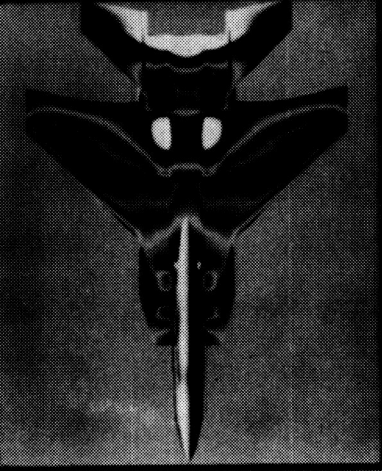
**CAP-TSD RESULTS EXHIBIT GOOD AGREEMENT  
WITH EXPERIMENT FOR COMPLETE F-15**

$M = 0.9, \alpha = 2.46^\circ$

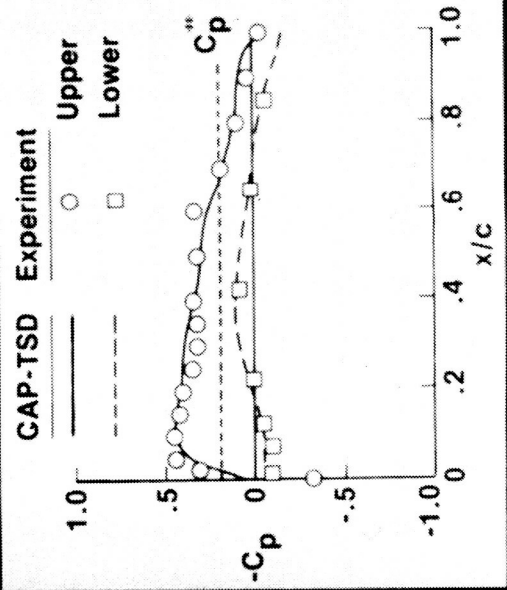
F-15 Geometry modeled



Steady surface pressures



Comparison with experiment at  $\eta = 0.59$



- Cooperative effort with McDonnell Aircraft for transonic flutter studies
- Complete geometry modeled
  - Wing
  - Horizontal tail
  - Fuselage
  - Flow-through inlet

Figure 22(b).

# CAP-TSD VALIDATED FOR TRANSONIC FLUTTER PREDICTION

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RTOP 505-63-21

**Research Objective** - The objective was to evaluate the capability of the CAP-TSD computer code for flutter prediction through the transonic speed range.

**Approach** - A recently developed computer code, CAP-TSD, solves the transonic small-disturbance potential equation using finite difference methods. The experimental results from a series of closely related wing models were selected for comparative analysis<sup>1</sup>. These models were flutter tested in air and Freon gas at Mach numbers ranging from 0.34 to 1.14. The experimental flutter speed index versus Mach number displays the characteristic transonic "bucket" or dip near  $M = 1.0$ . These planar wing models are swept back  $45^\circ$  at the quarter chord, have a taper ratio of 0.66, and have been proposed as an AGARD standard aeroelastic configuration.

**Accomplishment Description** - The flutter boundary results are shown in figure 23(b), the flutter speed index at the left and the flutter frequency ratio on the right, both as functions of the Mach number. Three different types of calculations were made; CAP-TSD results calculated using the linear-theory option, linear subsonic lifting-surface theory results and nonlinear CAP-TSD results for  $M = 0.678$ ,  $0.901$ , and  $0.96$ . For this thin swept wing, linear theory, as expected, predicts the subsonic flutter speed very well and the excellent correspondence of the linear CAP-TSD results indicates the validation of CAP-TSD for subsonic speeds. The nonlinear CAP-TSD results lower the flutter speed, especially at  $M = 0.96$  where the flutter boundary is particularly sensitive to Mach number. The bottom of the transonic flutter speed "bucket" is very well defined as is the steep supersonic portion of the curve.

**Significance** - The excellent agreement in flutter speeds represents the first successful calculation of a complete transonic flutter bucket, including the supersonic portion of the curve, using a CFD code. This provides confidence in the application of the code to complete aircraft configurations, for which it was developed.

**Future Plans** - The code is being made available to industry for further evaluation, application to complex configurations and comparison with flutter test data.

**Reference 1.** Yates, E. C., Jr.; Land, N. S.; and Foughner, J. T.: Measured and Calculated Supersonic and Transonic Flutter Characteristics of a  $45^\circ$  Swept-Back Wing Plan Form in Air and in Freon-12 in the Langley Transonic Dynamics Tunnel. NASA TN-D 1616, March 1963.

Figure 23(a).

# CAP-TSD VALIDATED FOR TRANSONIC FLUTTER PREDICTION

- O FIRST SUCCESSFUL CFD CALCULATION OF COMPLETE TRANSONIC FLUTTER "BUCKET"
- O AGARD STANDARD AEROELASTIC WING; 45 DEG SWEEP, 60% TAPER, 4% THICK

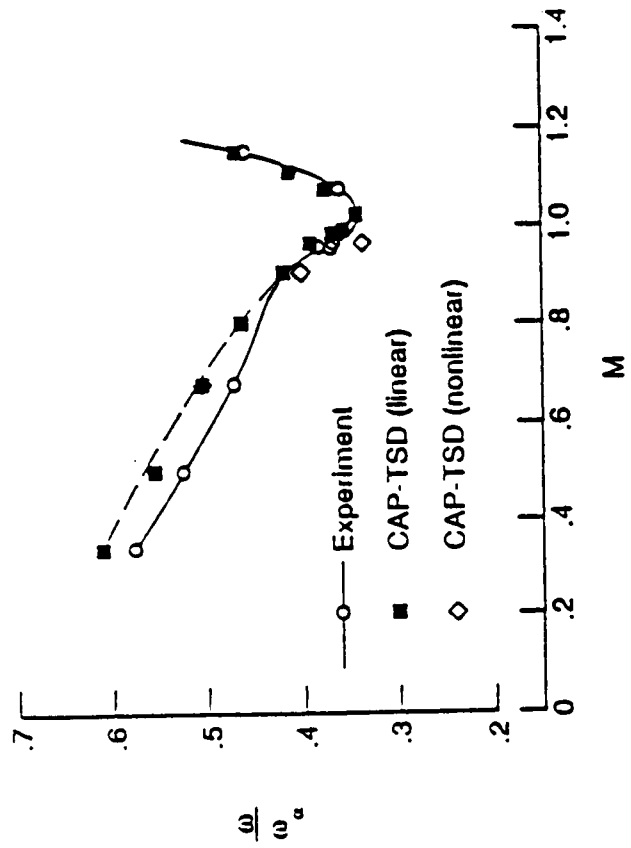
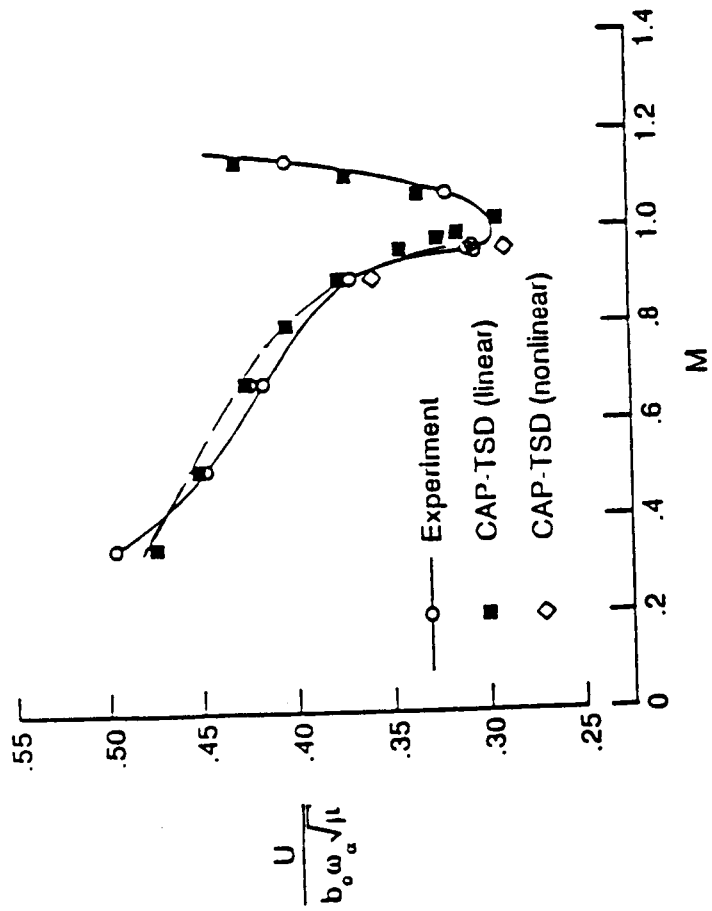
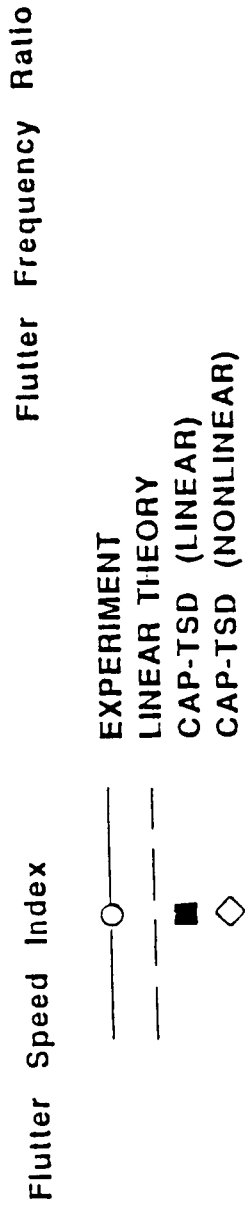


Figure 23(b).

## FLUTTER ANALYSIS ACCOMPLISHED FOR SLENDER DELTA WINGS

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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.

**Approach** - Comparisons of flutter conditions are made between calculations using linear theory unsteady aerodynamics and experimental flutter values. To calculate the aerodynamics, three different methods were used; a subsonic kernel function method, a subsonic doublet lattice method, and second-order piston theory.

**Accomplishment Description** - The flutter calculations were performed on six delta wings, three of which had clipped tips. The leading edge sweep of the delta wings was 70°, 75°, 80° while the sweep of the clipped delta wings was 54°, 62°, and 71°. Flutter calculations were made over a range of Mach numbers  $M = 0.2$  to 3.0. The experimentally measured mode shapes and frequencies were used in computing flutter boundaries. Comparisons of theory and experiment are shown in figure 24(b) for the 70° delta wing. Calculations using FAST (a subsonic kernel function for aerodynamics) and second order piston theory generally show good agreement with experiment, although in some cases a higher mode than given by experiment was calculated to flutter. The piston theory results below  $M = 2.0$  shows surprisingly good agreement even though it is not generally considered applicable in this region.

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

**Future Plans** - Further work on improving the comparisons between theory and experiment will be done. Flutter calculations with a nonlinear aerodynamic code will also be made for several of the clipped delta wings in order to validate the results.

Figure 24(a).



# FLUTTER ANALYSES ACCOMPLISHED FOR SLENDER DELTA WINGS

RIGOROUS CALIBRATION OF LINEAR AND TRANSONIC CODES USING FLAT PLATE FLUTTER DATA TO ESTABLISH BASELINE

O "HYPERSONIC STYLE" PLANFORMS - DATA FROM NASA TN D-2038

- DELTA WINGS OF 70°, 75°, 80° LEADING EDGE SWEEP

- CLIPPED DELTA WINGS OF 54°, 62°, 71° LEADING EDGE SWEEP

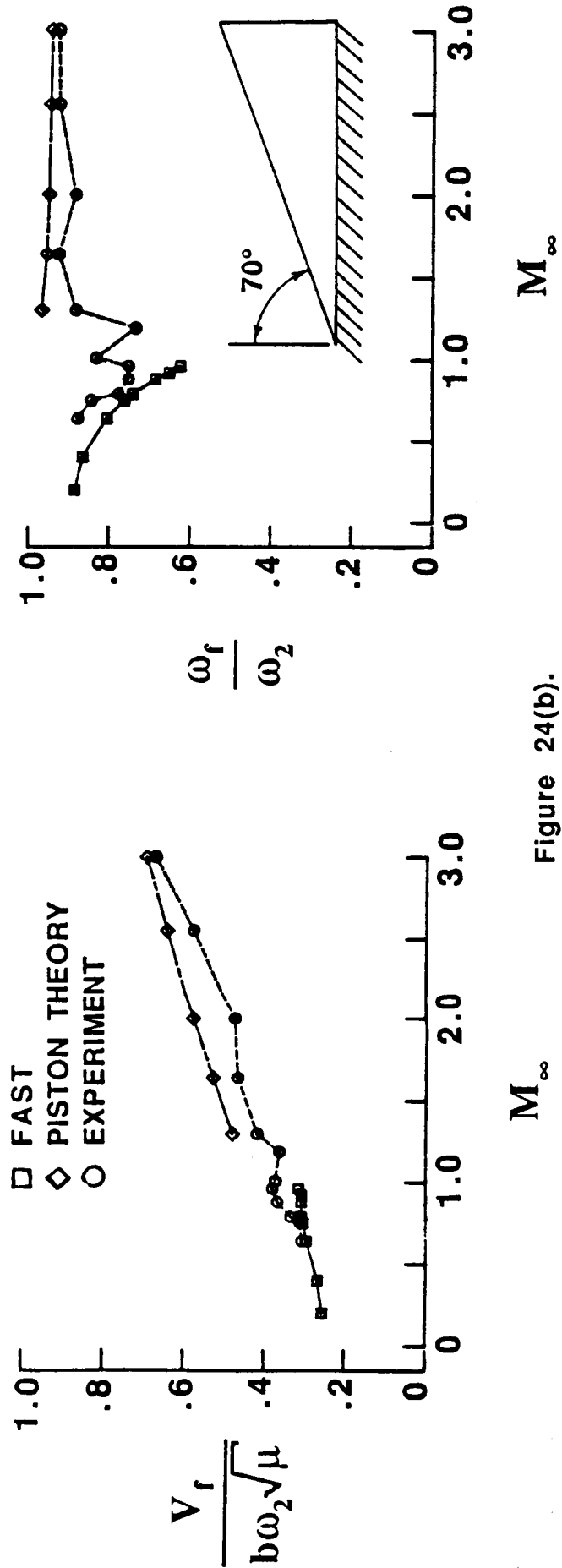


Figure 24(b).

CAP-TSD PRESSURE DISTRIBUTIONS COMPARE WELL WITH  
CFL3D (EULER) SOLUTIONS

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W. Kyle Anderson  
Analytical Methods Branch (LSAD)  
Extension 4646

RTOP 505-63-21

**Research Objective** - The objective of the research was to assess the accuracy of the CAP-TSD (Computational Aeroelasticity Program - Transonic Small Disturbance) code for steady and unsteady aerodynamic applications.

**Approach** - The accuracy of CAP-TSD was assessed by making detailed comparisons with Euler solutions, computed using the CFL3D (Computational Fluids Laboratory - Three Dimensional Euler and Navier Stokes) code. The comparisons were made by performing calculations for the F-5 wing. The F-5 wing has a leading edge sweep angle of  $32^\circ$ , a full-span aspect ratio of 3.16, and a taper ratio of 0.28. Steady-state results were obtained at freestream Mach numbers of  $M = 0.9$  and  $1.3$ . Unsteady calculations were also performed at  $M = 1.3$ , for the rigid wing pitching harmonically about a line perpendicular to the root at the root midchord. The reduced frequency was  $k = 0.198$  and the amplitude of oscillation was  $0.222^\circ$ . The calculations were further assessed by making comparisons with the experimental pressure data measured at the NLR in the Netherlands.

**Accomplishment Description** - Steady pressure distributions at  $\eta = 0.84$  are shown in figure 25(b). The results for  $M = 0.9$ , shown in the left half of the figure, indicate the presence of a relatively mild shock wave in the midchord region of the wing. The results for  $M = 1.3$  are shown in the right half of the figure. The CAP-TSD steady pressure distributions generally compare well with the CFL3D results and both sets of calculations are in agreement with the experimental data. Unsteady results are shown in figure 25(c) for both the upper and lower wing surfaces at  $\eta = 0.84$ . These results are presented in the form of real and imaginary components of the first harmonic of the pressure distributions divided by the oscillation amplitude. Similar to the steady-state comparisons, the CAP-TSD unsteady calculations agree well with both the CFL3D pressures and the experimental data.

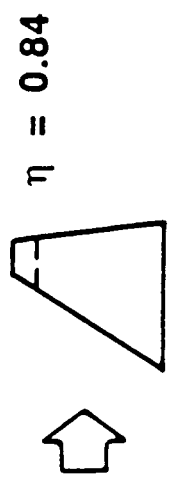
**Significance** - The good general agreement between CAP-TSD, CFL3D (Euler), and experimental pressure distributions indicates that TSD theory is accurate for steady and unsteady aerodynamic applications where shock waves are relatively mild and the amplitude of oscillation is small.

**Future Plans** - Additional CAP-TSD calculations are currently being performed to assess the influence of grid density, grid extent, outer boundary conditions, and time-step size on steady and unsteady solutions for airfoils and wings. Similar CFL3D results are being obtained to provide detailed solutions for comparison.

Figure 25(a).

# COMPARISON OF POTENTIAL AND EULER EQUATION STEADY PRESSURES

● F-5 WING,  $\alpha = 0^\circ$



- CAP-TSD
- - - CFL3D (Euler)
- □ Experiment

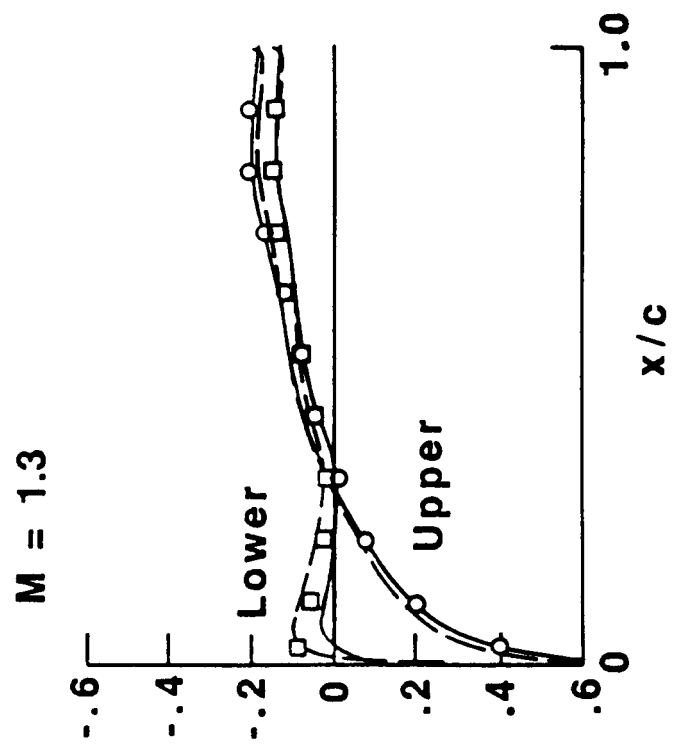
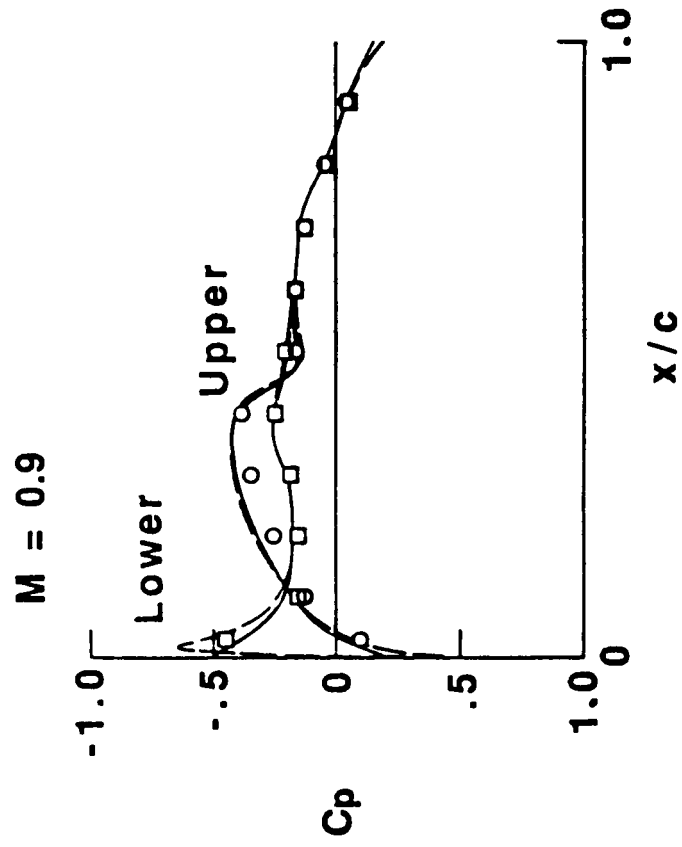
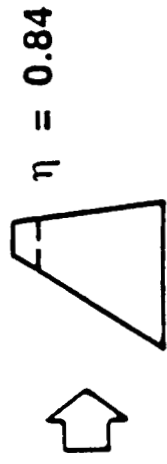


Figure 25(b).

# COMPARISON OF POTENTIAL AND EULER EQUATION UNSTEADY PRESSURES

● F-5 WING,  $M = 1.3$ ,  $\alpha = 0^\circ$ ,  $k = 0.198$



- CAP-TSD
- - - CFL3D (Euler)
- □ Experiment

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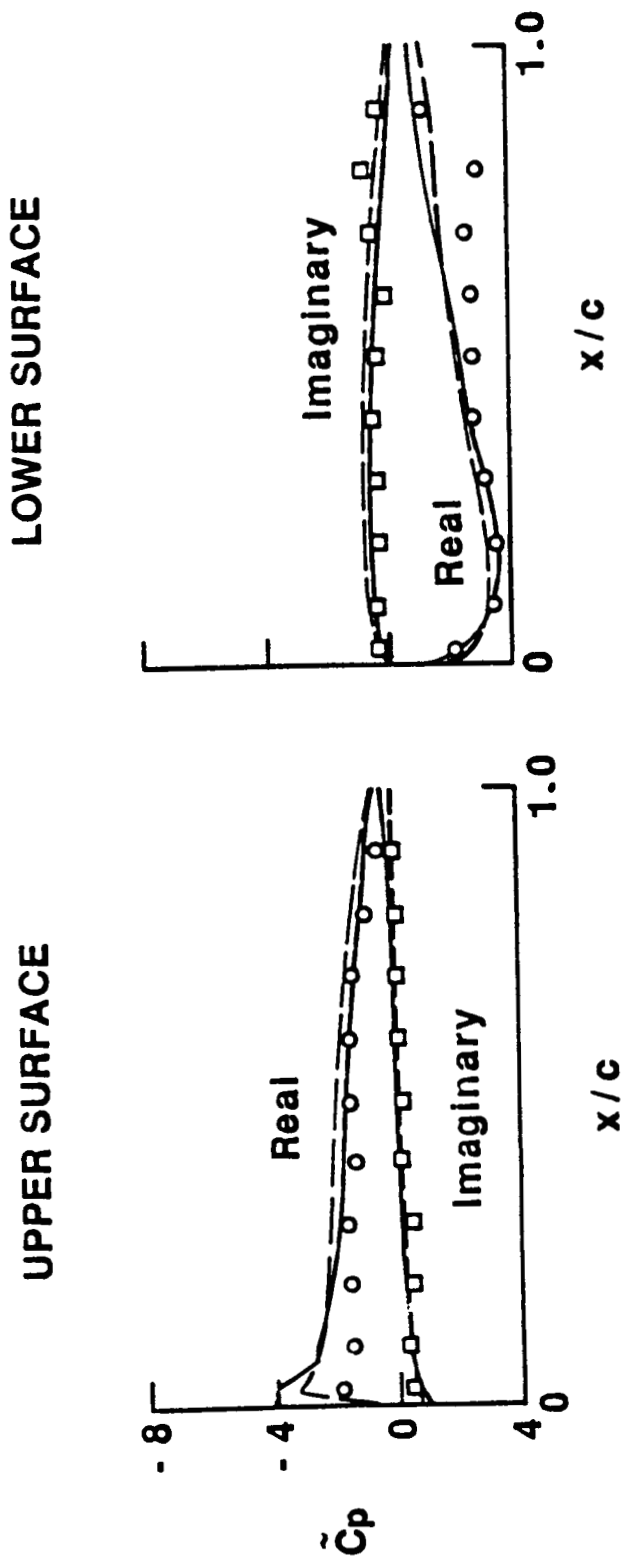


Figure 25(c).

# NAVIER-STOKES CALCULATIONS CORRELATED WITH TRANSONIC CRYOGENIC WIND TUNNEL DATA

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RTOP 505-63-21

**Research Objective** - The objective of this effort is to calibrate an unsteady Navier-Stokes code by comparing a series of calculated and measured pressures on an airfoil.

**Approach** - The approach is to compare calculated transonic pressures with those measured on a 14% thick supercritical airfoil at cryogenic temperatures shown in the upper left of figure 26(b). Reynolds numbers for the test data range from 5 million to 37 million. Surface pressures are calculated using the CFL2D code. It uses upwind differencing with flux-splitting and an approximate factorization algorithm to solve either the thin layer or the complete Reynolds averaged form of the 2-D Navier-Stokes equations. Comparisons of calculated and measured surface pressures and calculated and measured harmonics of surface pressures will be made.

**Accomplishment Description** - Steady calculations were made for a Reynolds number of 6.035 million and for wind tunnel conditions of free stream Mach number = 0.72 and angle of attack =  $2.504^\circ$ . The computational grid is shown at the upper right of the figure and the comparison of experimental data and calculations using the tunnel conditions as input into the code is shown at the lower left. This results in the calculated shock being further aft than the measured shock. Using the corrected Mach number, 0.701, and an angle of attack estimated to match the corrected experimental lift as input conditions, the agreement between the calculations and experiment is very good although there are slight differences in the pressure distributions, shown at the lower right of the figure. The corrected measured lift coefficient is 0.9753, and the calculated lift coefficient is 0.9626.

**Significance** - The results will provide information to help guide researchers in the development of viscous boundary layer methods that are coupled with inviscid flow codes. In addition, insight into the flow conditions for which the various forms of the Navier-Stokes equations are necessary for accurate flow field calculations will be obtained.

**Future Plans** - Future efforts will include correlation of additional steady flow data and unsteady data. Comparisons of steady state pressure distributions and the magnitude and phase of unsteady pressures will be made.

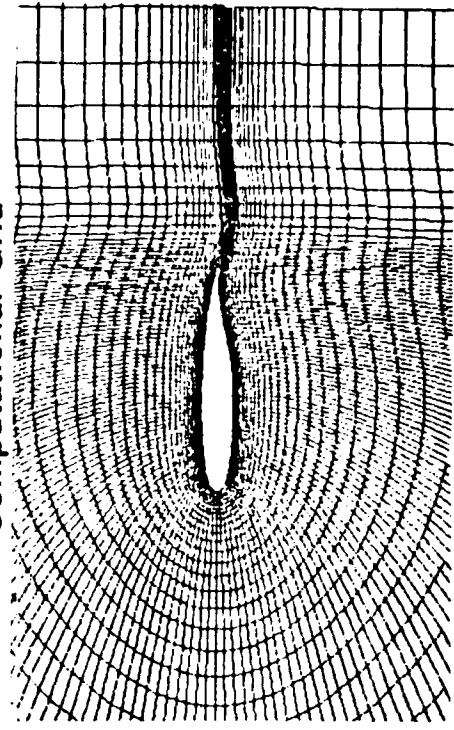
Figure 26(a).

# NAVIER-STOKES CALCULATIONS CORRELATED WITH TRANSONIC CRYOGENIC WIND TUNNEL DATA

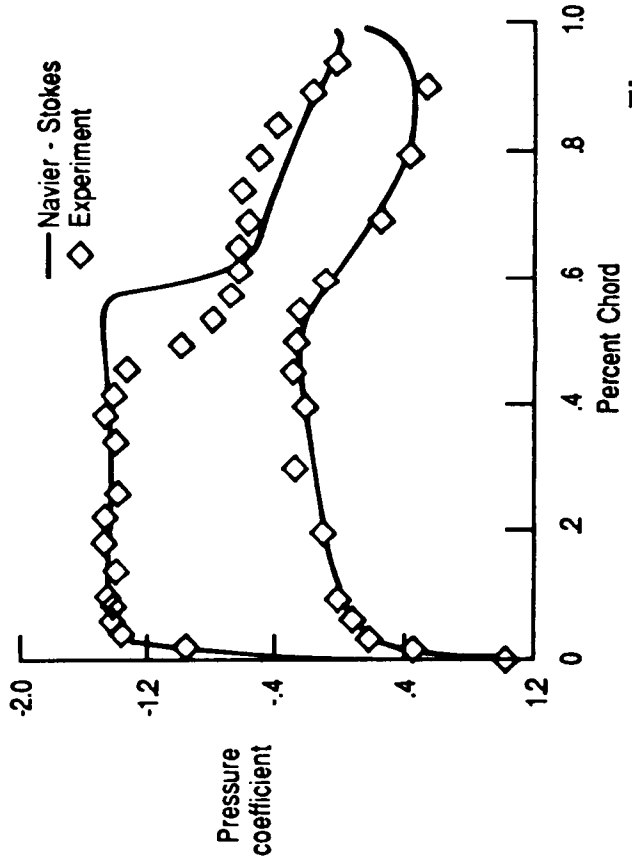
Supercritical Airfoil



Computational Grid



$M = 0.72, \alpha = 2.504$  degrees



$M = 0.701, \alpha = 2.17$  deg. (matched lift)

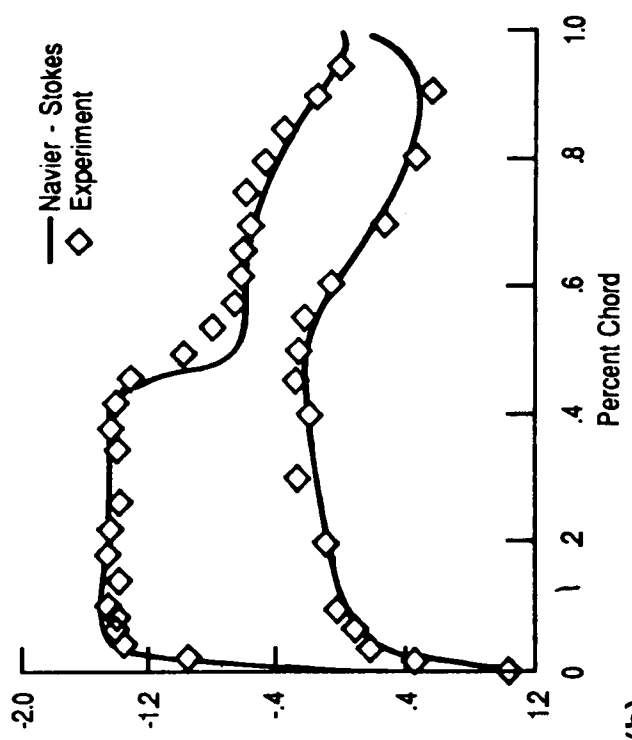


Figure 26(b).

# AEROSERVOELASTICITY

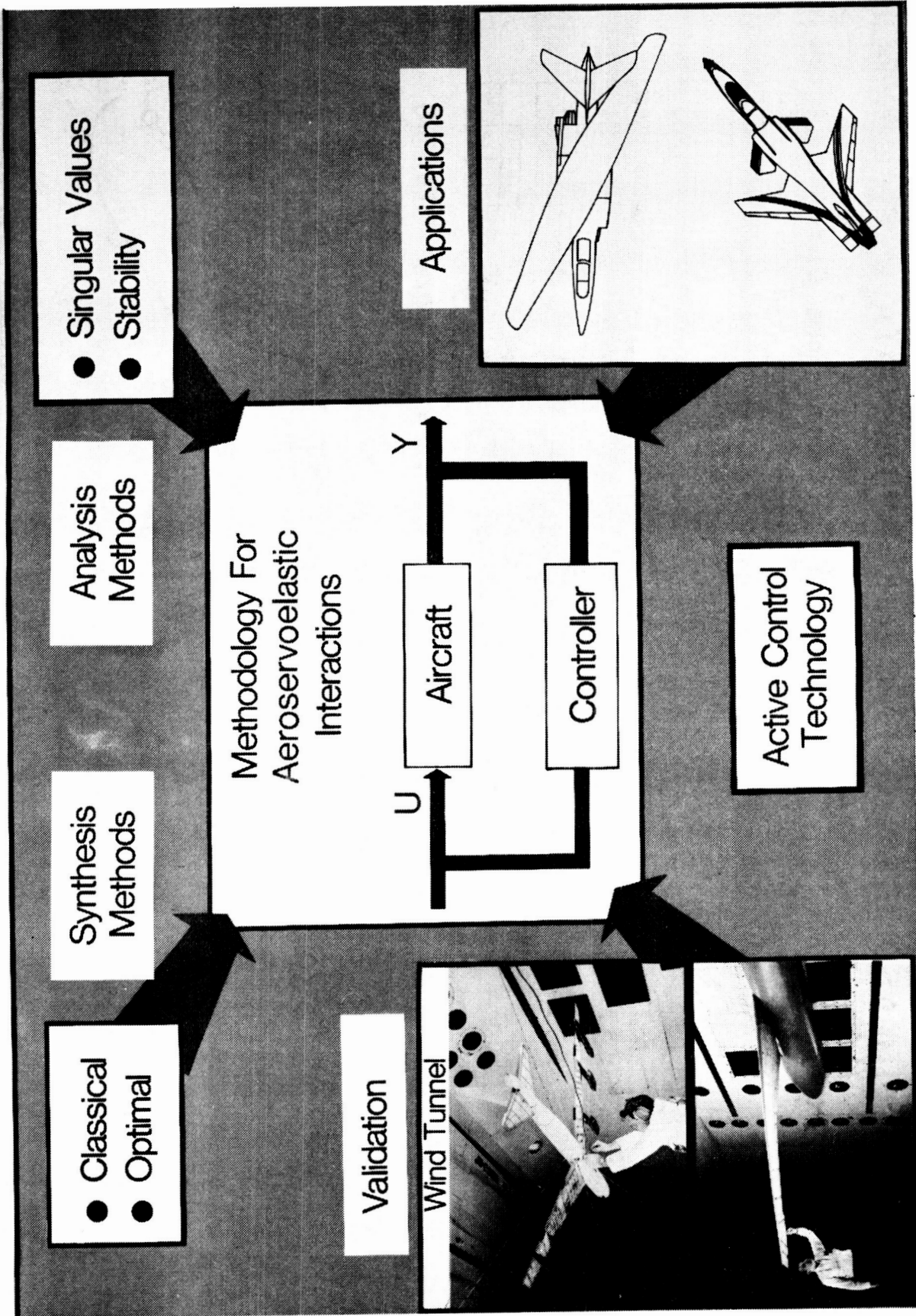


Figure 27.

# AEROSERVOELASTICITY FIVE YEAR PLAN

DISCIPLINARY THRUSTS	FY-87	FY-88	FY-89	FY-90	FY-91	EXPECTED RESULTS
<b>ANALYSIS METHODS</b>	MODELING TECHNIQUES AERO, THERMO, CONTROLS					AEROSERVO-ELASTIC METHODS FOR ANALYSIS AND SYNTHESIS TO ALLOW ACTIVE CONTROLS INTEGRATION INTO AIRCRAFT OPTIMIZATION
	EMPIRICAL CORRECTIONS					
	STATIC AEROSERVOELASTICITY LINEAR, NONLINEAR AERO					
	DYNAMIC AEROSERVOELASTICITY LINEAR, NONLINEAR AERO					
<b>DESIGN METHODS</b>	OPTIMAL SENSITIVITY METHODS					VALIDATED ANALYSIS AND DESIGN METHODS
	INTEGRATED STRUCTURAL/CONTROL DESIGN					
	ADVANCED ANALOG/DIGITAL CONTROL LAW SYNTHESIS					
<b>APPLICATIONS AND VALIDATIONS</b>	NEW AIRCRAFT DESIGNS (I.E. X-WING, NASP)					VALIDATED ANALYSIS AND DESIGN METHODS
	WIND TUNNEL AND FLIGHT EXPERIMENTS (I.E. AFW, ARW-II, F-18, NASP)					

Figure 28.



# INTEGRATED STRUCTURE/CONTROL LAW DESIGN

Michael G. Gilbert  
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Extension 2388

RTOP 505-63-21

**Research Objective** - The objective of the current research is to complement the existing aeroservoelastic analysis techniques by developing integrated structure and control law design methods applicable to aircraft with significant aeroservoelastic interactions.

**Approach** - The integrated structure/control law design methodology development is based on hierarchical multilevel problem decomposition and optimization techniques. The hierarchical decomposition techniques allow for a natural ordering of design objectives into system level objectives and subordinate subsystem objectives. This ordering provides a structure within the design methodology to trade off subsystem performance for improved system performance. A rational means for making subsystem performance trade offs is provided through the use of optimization techniques for subsystem design and the use of sensitivity of optimum solution concepts to obtain subsystem design sensitivity information. The subsystem design sensitivity information is used at the system design level to make decisions which influence the subsystem designs in such a way that overall system performance is improved. This approach is illustrated on the left half of figure 29(b).

**Accomplishment Description** - Structural multilevel optimization methods and sensitivities of optimized structural design solutions have been developed and are reported elsewhere. Analytical sensitivity of optimum control law solutions to problem parameters have been developed for the Linear system, Quadratic cost, Gaussian distributed disturbance (LQG) optimal control law problem. The analytical sensitivity equations were derived by differentiating the necessary conditions of optimality for the LQG problem, thus eliminating the need for perturbed optimal control law solutions and finite difference derivative calculations. The change in the optimum solution due to changes in a problem parameter  $p$  from  $p_0$  to  $p_1$  can be calculated using only the information available from the solution of the problem at  $p_0$ , shown on the right side of the figure. The accuracy of the sensitivity calculations were verified by comparing predicted changes with actual changes for parameter variations of  $\pm 10\%$ .

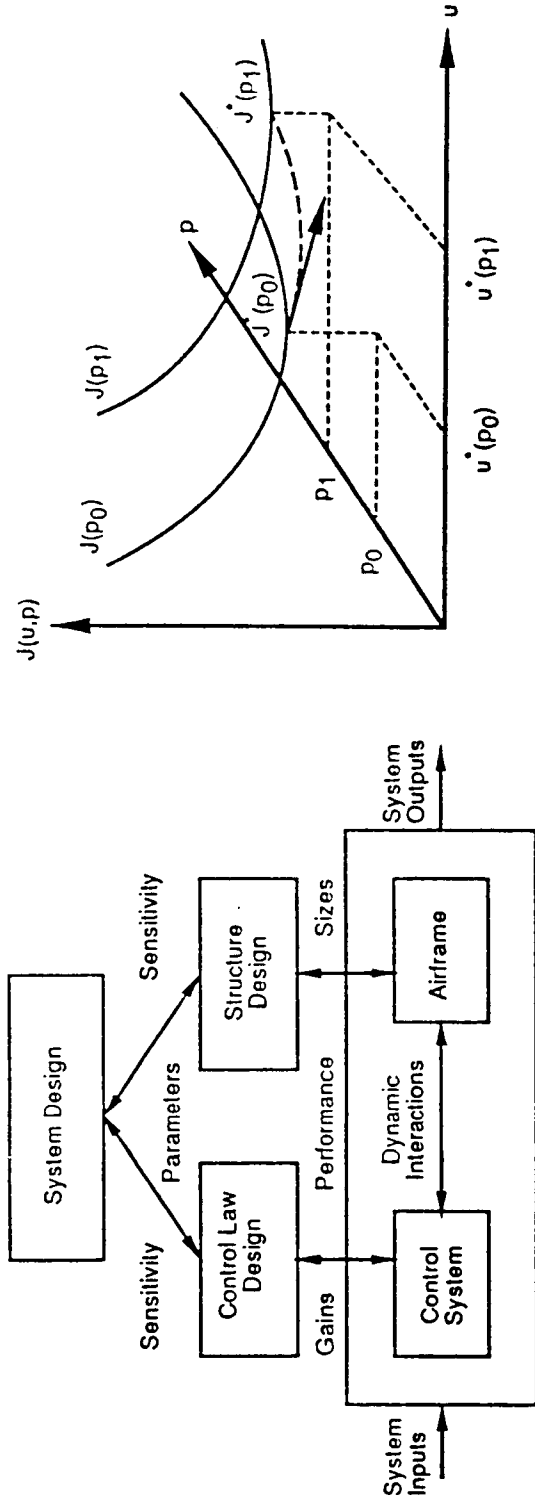
**Significance** - The sensitivity of optimal LQG control laws to LQG problem formulation parameters and structural design parameters (for structural systems) can be calculated using analytical sensitivity equations without multiple perturbed LQG solutions and finite differencing. The use of analytical sensitivity expressions results in a significant decrease in the computation burden required for the integrated structure/control law design method outlined above and yields reasonably accurate sensitivity information. Sensitivity of the LQG solution to structural parameters permits tailoring the structural design to improve the control subsystem (and also overall system ) performance.

**Future Plans** - Continued development of the integrated structure/control law design methodology will proceed along two lines: a) derivation of analytical sensitivity expressions for reduced order, constrained optimal control law problems reflecting a more realistic control law design methodology, and b) exercising the complete design methodology with structural optimization and the LQG control law problem formulation. Results for the complete methodology will be compared with results using other integrated structure/control law design methods under development in the large space structures fields in order to evaluate the effectiveness of the current approach.

Figure 29(a).

# INTEGRATED STRUCTURE/CONTROL LAW DESIGN

Multilevel Integrated Design Approach      Sensitivity of Optimum



- Multilevel Structural Design Available
- Analytical Sensitivity Equations for LQG Optimal Control Law Problem Have Been Developed
- Analytical Sensitivity Equations for Constrained Optimal Control Law Design Under Development

Figure 29(b).

## DIGITAL ROBUST ACTIVE CONTROL LAW SYNTHESIS USING CONSTRAINED OPTIMIZATION

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PRC Kentron Inc.  
Extension 2012  
RTOP 505-63-21

**Research Objective** - The objective of the present work is to develop methodology and computer-aided design software for synthesis of digitally implementable active control laws for aeroservoelastic systems, which would meet multiple design requirements while maintaining reasonable stability margins.

**Approach** - The system is mathematically modeled by a set of discrete state space equations at a specified sampling rate. A Linear Quadratic Gaussian (LQG) type cost function consisting of the weighted sum of steady state Root-Mean-Square (RMS) responses is minimized by updating the free parameters of the discrete control law while satisfying a set of constraints on the design loads and responses. The stability robustness of the system can also be improved by imposing constraints on the minimum singular values at the plant input and output.

**Accomplishment Description** - The expressions for the gradient of the cost function and the constraints were derived analytically to facilitate rapid numerical convergence. You can choose the structure of the control law and design variables, such as an existing control law, or full or reduced order LQG control laws that can be modified to meet specific design objectives. The methodology was used to synthesize 2nd order robust digital control laws for gust load alleviation of a drone aircraft modeled by a 32nd order sampled data system (figure 30(b)). The control law reduced the open loop RMS bending moment and shear at the wing root by over 50% without increasing the outboard bending moment and torsion, and keeping the elevator and aileron deflections and rates within allowable limits (figure 30(c)).

**Significance** - Since the actual implementation of active control laws is usually by digital microprocessors for flexibility and cost effectiveness, the direct synthesis and simulation of digital control laws, which includes the effect of sampling, computational delay, anti-aliasing filters, etc., will reduce the possible problems associated with digital implementation of an analog design.

**Future Plans** - The future plans are the following: a) Develop the capability of imposing constraints on the transient responses, b) include a control law structure similar to a current and predictive type discrete Kalman Filter, c) develop an interactive version of the synthesis software and a user's manual, and d) synthesize a digital flutter suppression control law for the AFW (Active Flexible Wing) wind-tunnel model.

Figure 30(a).

# DIGITAL ROBUST ACTIVE CONTROL LAW SYNTHESIS USING CONSTRAINED OPTIMIZATION

Methodology based on

- analytical gradient expressions

- design parameters are digital control law elements
- design criteria incorporated as inequality constraints

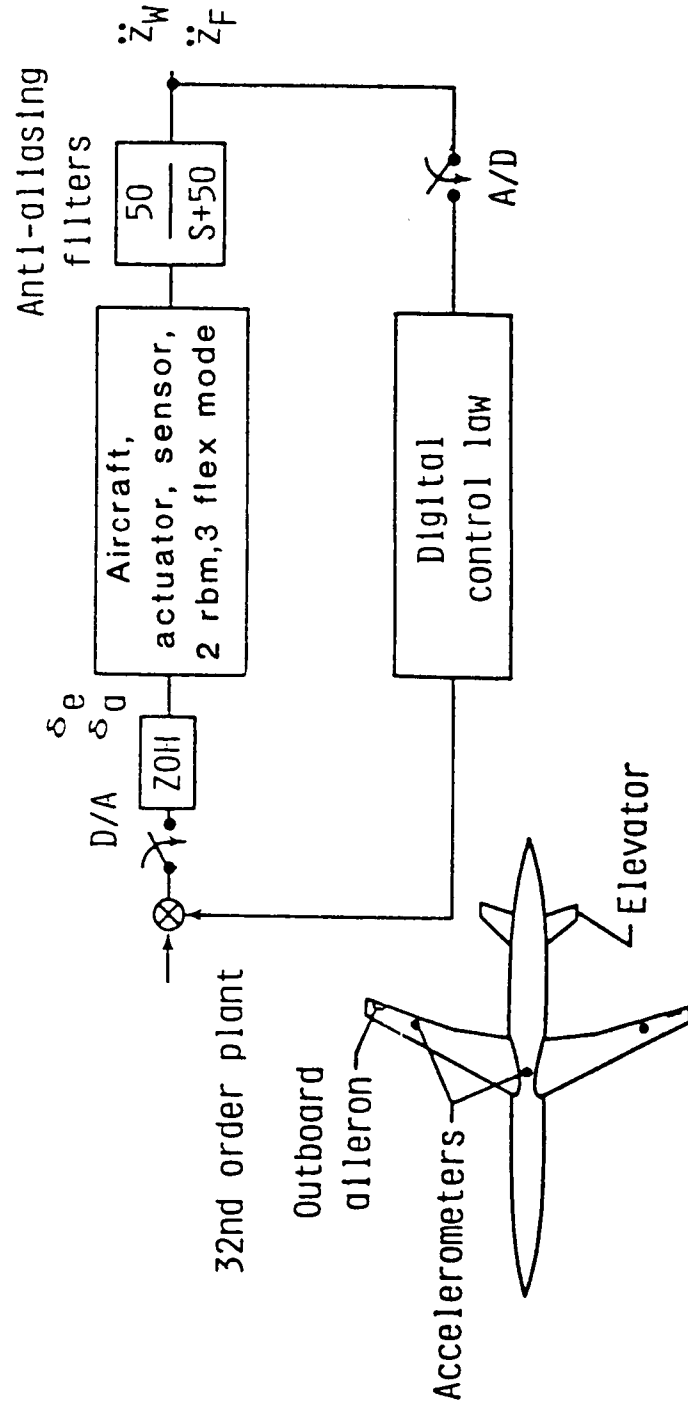
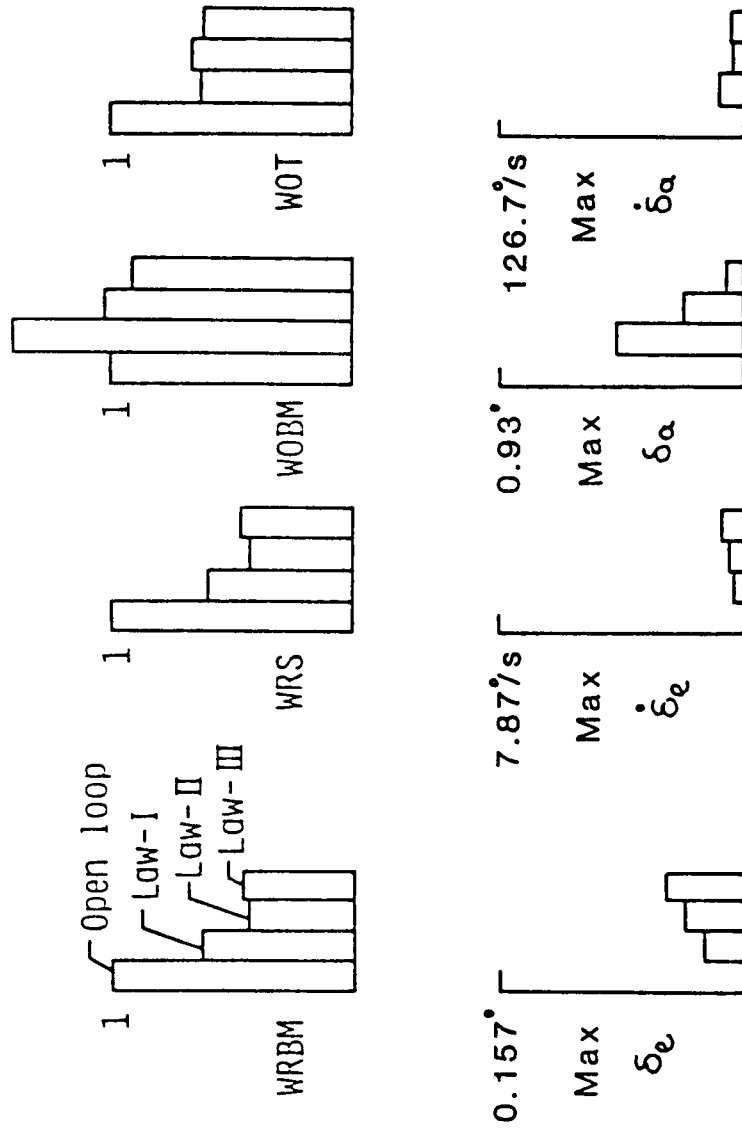


Figure 30(b).

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# COMPARISON OF NORMALIZED RMS RESPONSES

## DUE TO 1 M/S RMS DRYDEN GUST



o law-I Initial 2nd order digital

o law-II After optimization

o law-III After optimization with constraints

Figure 30(c).

# METHODOLOGY FOR MATCHING EXPERIMENTAL AND COMPUTATIONAL AERODYNAMIC DATA

Carol D. Wieseman  
Aeroservoelasticity Branch  
Extension 4232

RTOP 505-63-21

**Research Objective** - The objective of the present work is to develop and implement 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 correction factors involves steady pressures and forces from wind tunnel and flight experiments, or calculated steady data from a higher level source code, such as a CFD code.

**Approach** - Three different approaches to calculate correction factors are being developed and implemented. Correction factors are calculated by matching pressure data at analytical box locations (upper left of figure 31(b)), by matching section properties, or by matching total forces. The approaches for matching section properties and total forces use optimization procedures. Correction factors are then applied to either box downwashes or to box pressures.

**Accomplishment Description** - Spline interpolations for steady and unsteady pressure distributions are operational. Typical spline fits of experimental pressure distributions at several chord sections are shown in the lower left of the figure. One-dimensional splines are used instead of surface splines because of the known inaccuracies of surface-splining pressure distributions. Currently, pressure and downwash correction factors have been calculated for a rectangular supercritical wing which was previously tested in the TDT at a Mach number = 0.4. This involved calculating unsteady pressures at reduced frequencies for which there was wind-tunnel pressure data, and spline fitting the experimental unsteady pressure data to the analytical box locations. In addition, methodology for matching force data has been implemented, and correction factors have been calculated for an F-18 configuration. Some of the options that are available for choosing objective functions and design variables to achieve the goal of matching, as closely as possible, the experimental and analytical total force derivatives were exercised. Results for a typical case are shown in the right of the figure.

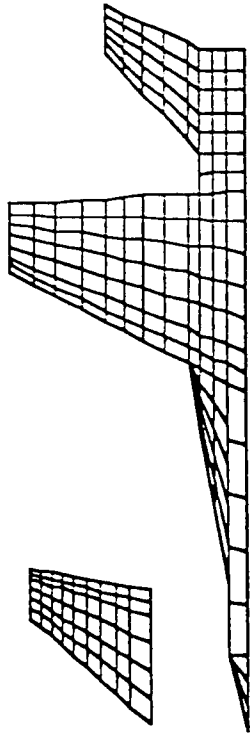
**Significance** - In performing aeroservoelastic analyses, it is important to have accurate predictions of the unsteady aerodynamic forces. This methodology will help improve the accuracy of the Doublet Lattice Method which is currently the state-of-the-art in calculating unsteady oscillatory aerodynamics.

**Future Plans** - The methodology will be validated by comparing experimental unsteady pressure data with corrected Doublet Lattice data. Methodology for matching section properties will be completed and validated. Test cases using wind-tunnel pressure data from the rectangular supercritical wing will be used to validate the methodology for matching pressures and section properties. The F-18 will be used as a test case for matching total forces.

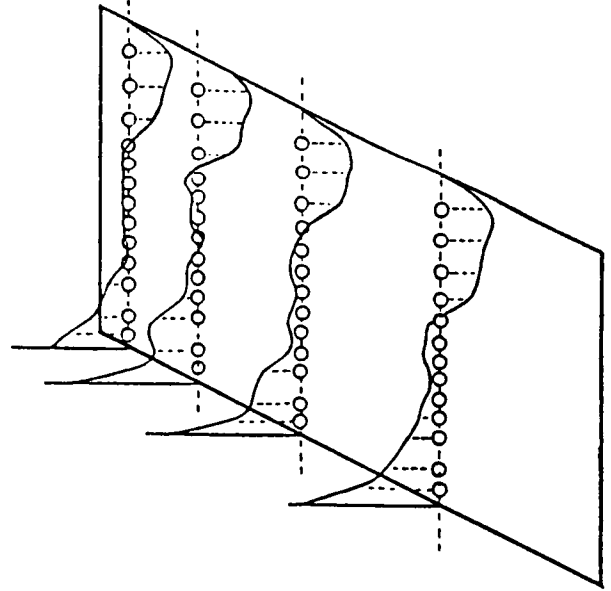
Figure 31(a).

# AERODYNAMIC CORRECTION FACTOR METHODOLOGY

Box Layout for Analytical  
Calculation of  
Aerodynamic Forces

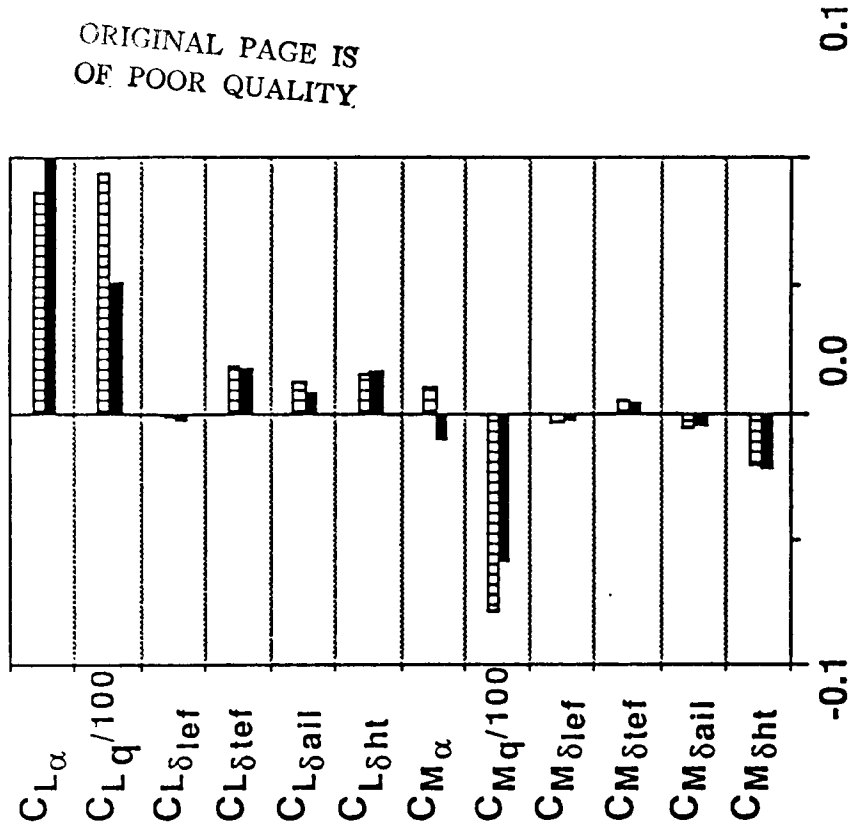


Experimental Pressure Distributions



Comparison of F-18 Force  
Coefficients

■ Experimental  
▤ Analytical



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Figure 31(b).

## COMPARISON OF SDG AND PSD METHODS

Boyd Perry III  
Aeroservoelasticity Branch  
Extension 3323

RTOP 505-63-21

**Research Objective** - The objective of the present work is to compare the results of two gust response analysis methods: the Statistical Discrete Gust (SDG) Method and the Power Spectral Density (PSD) Method.

**Approach** - The first figure shows the essential elements of both the SDG and PSD Methods. The SDG Method is performed in the time domain with response quantity  $\bar{\gamma}$ ; the PSD Method is performed in the frequency domain with response quantity  $\bar{A}$ . Shown for each are representative inputs and outputs. Jones' claim about the SDG and PSD Methods is expressed quantitatively by the equation at the top of figure 32(b). The approach taken is to perform analyses using the SDG and PSD Methods for several airplanes and to compare the corresponding responses from each method to see if the "10.4 factor" is obtained. Rigid-body analyses are performed first followed by fully-flexible analyses.

**Accomplishment Description** - Figure 32(c) contains typical results from a rigid-body analysis of a transport at the indicated flight condition. Five center-of-gravity (c.g.) positions were investigated, resulting in five short-period frequencies, which have been normalized by the value of frequency corresponding to the "knee" ( or maximum ) of the  $\Phi_{wg}$  plot in the first figure. Plotted are ratios of  $\bar{\gamma} / \bar{A}$  for pitch-rate and c.g. vertical-acceleration responses as a function of normalized frequency. The frequency corresponding to the nominal c.g. is indicated on the plot. Both ratios approach 10.4 very closely for the higher values of normalized frequency, but begin to diverge from 10.4 at the lower values.

**Significance** - The 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 establish 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. The developer of the SDG Method, J. G. Jones of the Royal Aircraft Establishment, claims that, under certain conditions, the SDG and PSD Methods produce essentially the same numerical results. 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, in September 1986 the FAA requested NASA's assistance in investigating the claim made by J. G. Jones regarding the SDG and PSD Methods. The present work will provide the FAA with the desired information.

**Future Plans** - The primary concern in this activity is to determine if the structural-load responses for a fully-flexible airplane exhibit the "10.4 factor." This concern is presently being addressed. A final report including the fully-flexible results will be prepared and presented to the FAA in April 1988.

Figure 32(a).



# COMPARISON OF SDG AND PSD METHODS

FAA has requested assistance from NASA to validate J.G.Jones' claim

*"The SDG and PSD analysis methods produce essentially the same numerical results..."*

$$\bar{\gamma} = 10.4 \bar{A}$$

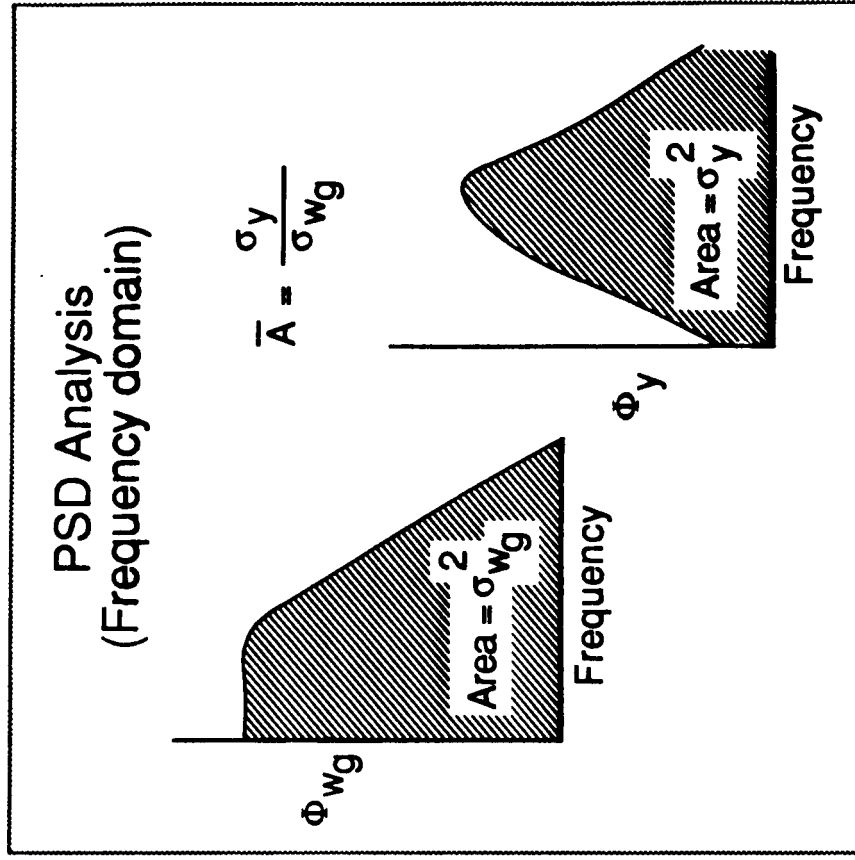
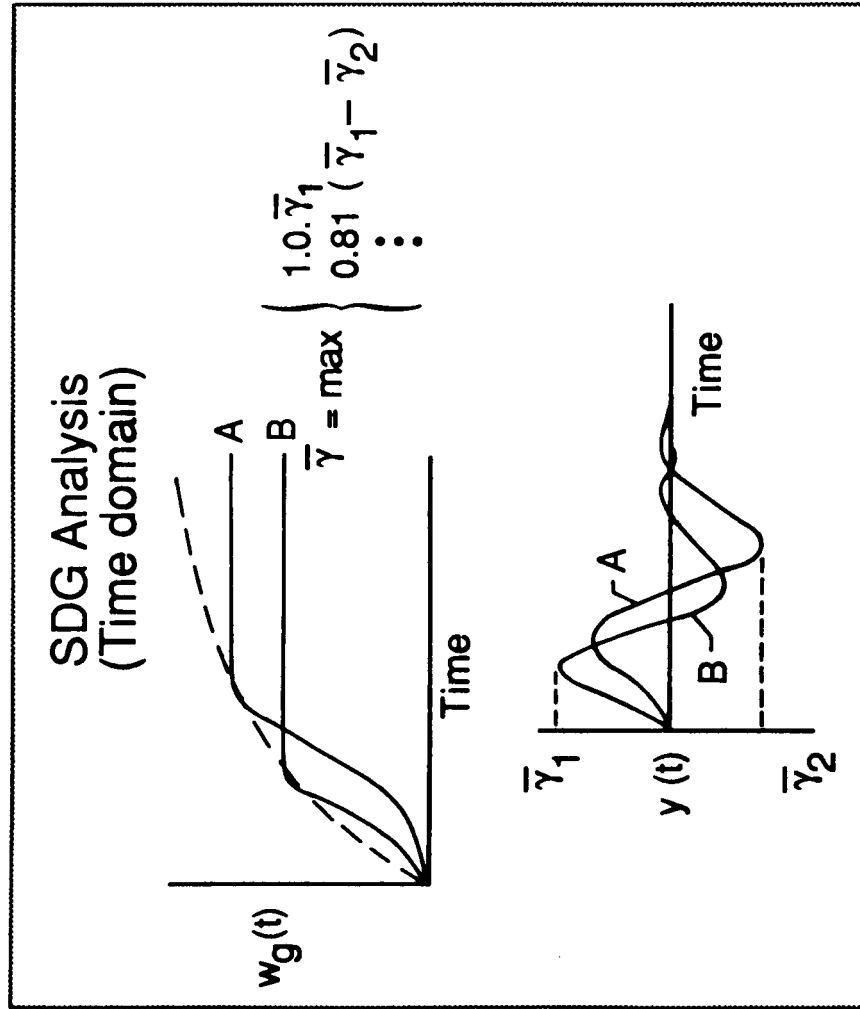


Figure 32(b).

# COMPARISON OF SDG AND PSD METHODS

Example transport  
 Alt = 30,000 ft Mach = 0.74

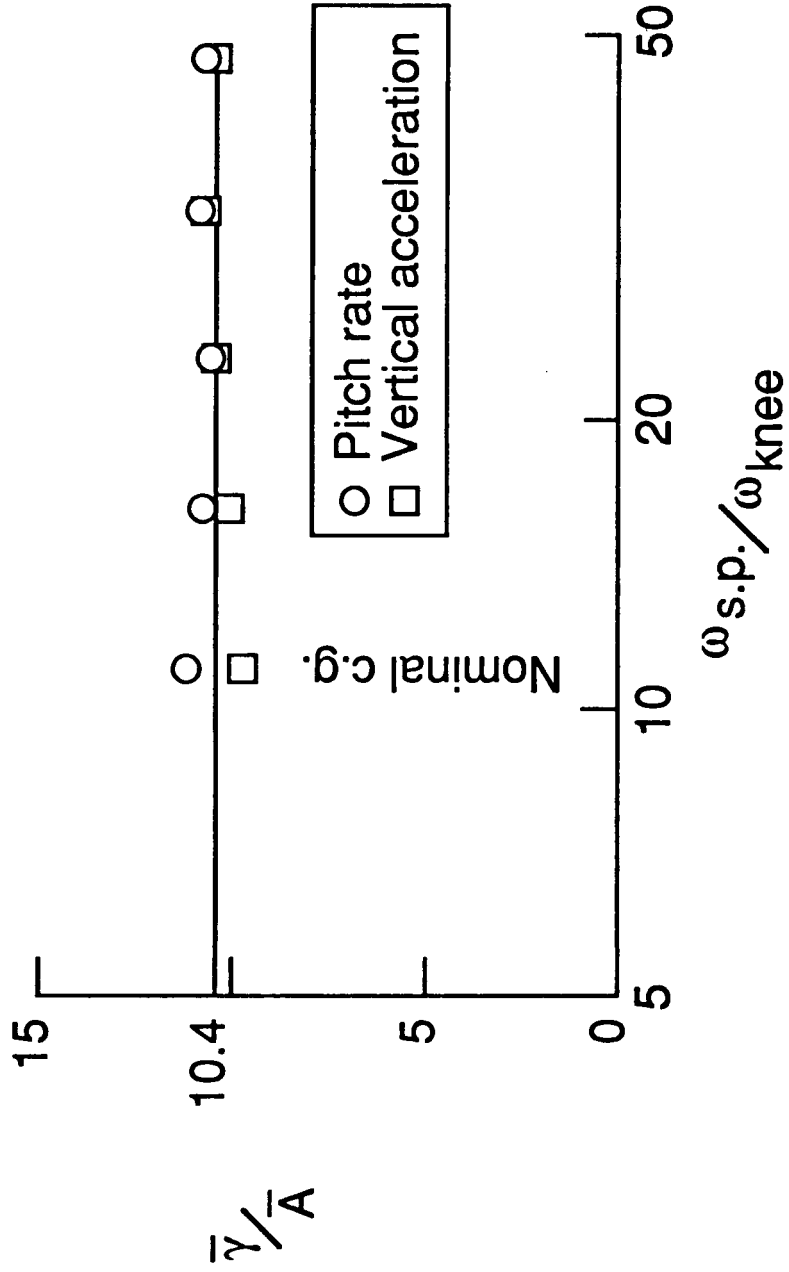


Figure 32(c).

# FLEXIBLE WING RESPONSE CORRELATES WITH CHANGING FLOW CONDITIONS IN TRANSONIC RANGE

Cilinton V. Eckstrom, Maynard C. Sandford, and David A. Seidel  
Aeroservoelasticity, Configuration Aeroelasticity, and Unsteady Aerodynamics Branches  
Extensions 3834, 2661, and 4236

RTOP 505-63-21

**Research Objective** - The objective was to define the unsteady flow conditions associated with an unusual dynamic wing response encountered in the transonic range on the Aeroelastic Research Wing (ARW-2) as tested in the TDT.

**Approach** - The initial test of the ARW-2 in the TDT in 1983 resulted in the prediction of what appeared to be a wing first-bending mode instability at dynamic pressures well below the predicted classical flutter boundary. A second test in 1986 to explore the predicted instability found no "hard flutter" conditions but rather a narrow transonic region of high dynamic wing response. During that test, continuous time history measurements of unsteady wing surface pressures were obtained for evaluation to determine if a physical understanding of the flow conditions could be developed.

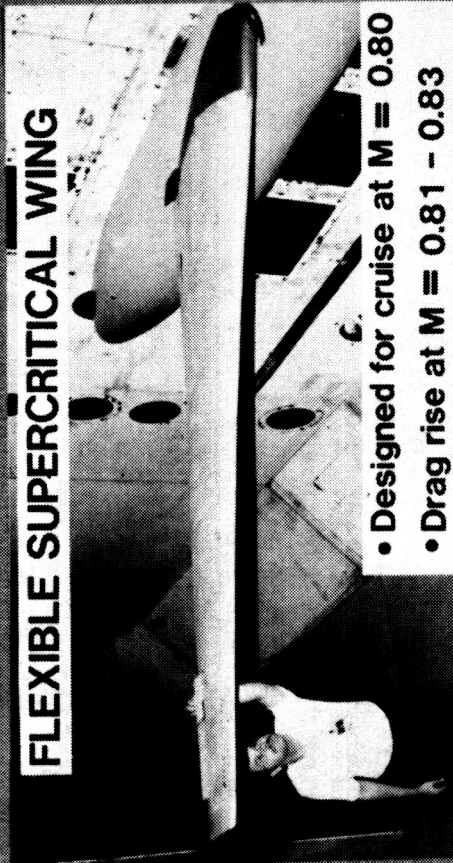
**Accomplishment Description** - The ARW-2 is a flexible supercritical wing designed for a cruise Mach number of 0.80. Previous tests on a full-span scale model determined that for this configuration the drag rise occurs in the Mach number range 0.81 to 0.83. Drag rise is considered a strong indicator of the onset of a breakdown in steady flow conditions followed by the development of a strong recompression shock with associated boundary layer flow separation. Figure 33(b) presents sample surface pressure time histories measured on both the upper and lower surfaces of the ARW-2 at a span location of  $\eta = .871$  for a range of Mach numbers. At  $M = 0.80$  some unsteadiness is apparent at 1. At  $M = 0.88$  the upper surface has developed steady flow at location 1 ahead of a strong oscillating shock at 2 while the lower surface has unsteady flow in the range of 3 to 4. At  $M = 0.92$  there is steady supersonic flow at 1 and 3 with strong oscillating shocks at 2 and 4. At  $M = 0.96$  the strong recompression shocks have moved aft of 2 and 4 and no oscillating pressures were observed. Although not shown, the flow aft of the shock locations at  $M=0.96$  is completely separated and steady.

**Significance** - This investigation has shown that the magnitude of the flexible wing dynamic response in the transonic region correlates closely with the variation in unsteady flow conditions which occur during the transition from subcritical attached flow to supercritical flow with a strong recompression shock and associated separation of boundary layer flow aft of the shock location.

**Future Plans** - The data analysis will be completed to further evaluate the effects of variations in Mach number, dynamic pressure, angle of attack, and control surface deflection position and the results will be published in a conference paper and a formal NASA paper.

Figure 33(a).

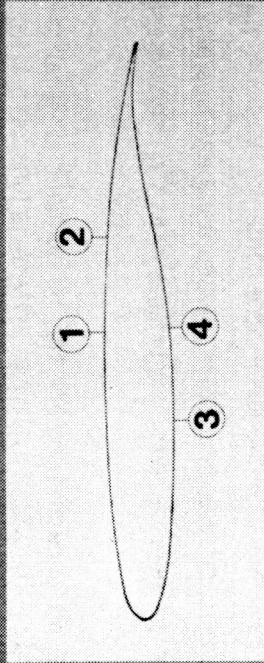
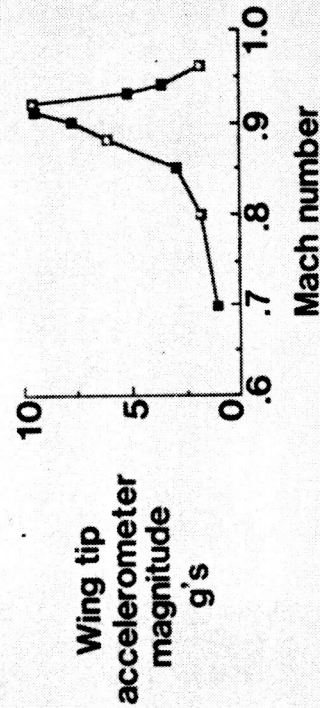
# FLEXIBLE WING RESPONSE CORRELATES WITH CHANGING FLOW CONDITIONS IN TRANSONIC RANGE



**FLEXIBLE SUPERCritical WING**

- Designed for cruise at  $M = 0.80$
- Drag rise at  $M = 0.81 - 0.83$

## WING DYNAMIC RESPONSE



## PRESSURE TIME HISTORIES

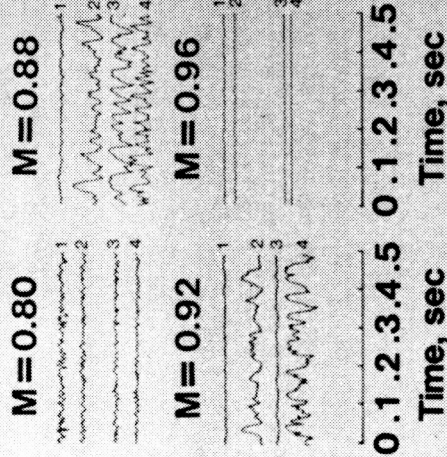


Figure 33(b).

# AEROSERVOELASTIC ANALYSIS VALIDATED BY WIND-TUNNEL TESTS

Boyd Perry, III  
Aeroservoelasticity Branch  
Extension 3323  
RTOP 505-63-21

**Research Objective** - The objective was to synthesize and test an active roll control system for the Active Flexible Wing wind-tunnel model shown in the attached figure. The research was conducted jointly by the Aeroservoelasticity Branch and the Configuration Aeroelasticity Branch.

**Approach** - The Active Flexible Wing concept was developed by Rockwell International, who designed and built the wind-tunnel model. The model is aeroelastically scaled with a span of about nine feet and is mounted on a sting in such a manner that the model is free to roll. It was tested in the Langley Transonic Dynamics Tunnel during February 1987. The model has two leading-edge and two trailing-edge control surfaces on each wing panel. At each test condition the two most effective pairs of control surfaces are chosen for roll control; for the present example these are the trailing-edge-inboard pair ( $\delta_1$ ) and the leading-edge-outboard pair ( $\delta_2$ ).

**Accomplishment Description** - Active Roll Control (ARC) control laws were synthesized such that the roll performance of the model and the stability robustness (i.e. gain and phase margins) of the model are constant for all combinations of feedback parameter,  $\kappa$ , and input-scaling parameter,  $\kappa_c$ . A block diagram of the ARC system is shown in the lower left corner of figure 34(b). The plot in the upper right corner of the figure presents analytical predictions of a 0.3-second ramp-hold roll-rate command (input to the block diagram) and the corresponding roll-rate response (output from the block diagram) time histories. For the given roll-rate command the roll-rate response is predicted to always have the shape shown (and although not shown, the stability robustness of the model is predicted to always be the same) for all values of  $\kappa$  and  $\kappa_c$ . The quantities which do change with  $\kappa$  and  $\kappa_c$  are the deflections of the control surfaces,  $\delta_1$  and  $\delta_2$ . The plot in the lower right corner of the figure presents a comparison of analytical predictions and corresponding experimental results of control surface deflections. Plotted are the values of  $\delta_1$  and  $\delta_2$  which result from applying the given roll-rate command to the model for a constant value of  $\kappa_c$  and several values of  $\kappa$ . The agreement between analysis and experiment is very good.

**Significance** - The control law parameterization plot indicates that constant performance and stability may be obtained for a range of control surface deflections. Control surface deflection is a measure of the effort required by the control system to perform its function and this plot, and others like it, allow the engineer to minimize that effort by proper choice of  $\kappa$  and  $\kappa_c$ . In a similar manner, wing loads (shears, bending and torsion moments) during the roll maneuver may be minimized by an alternate choice of  $\kappa$  and  $\kappa_c$ .

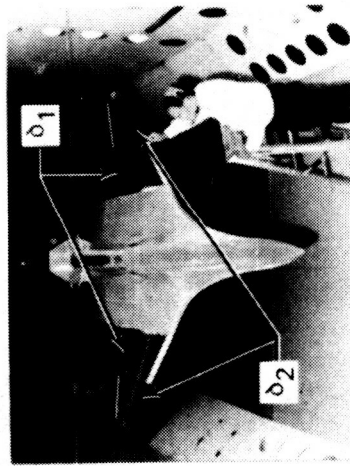
**Future Plans** - Data reduction from the wind-tunnel test will be completed and a NASA-TP will be prepared. Present plans call for an additional TDT tunnel entry in the spring of 1989, focusing on active flutter suppression and maneuver load control.

Figure 34(a).

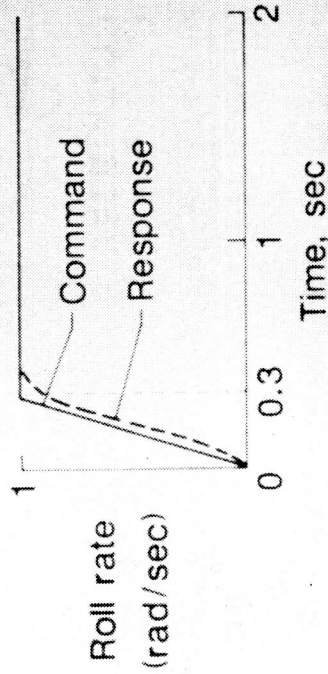
# AEROSERVOELASTIC ANALYSIS VALIDATED BY WIND TUNNEL TESTS

Mach = 0.90 Dynamic pressure = 250 psf

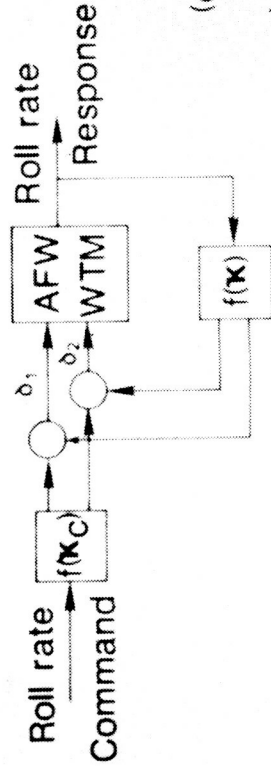
Active flexible wing model



Roll rate performance



Block diagram of active roll control system



Control law parameterization

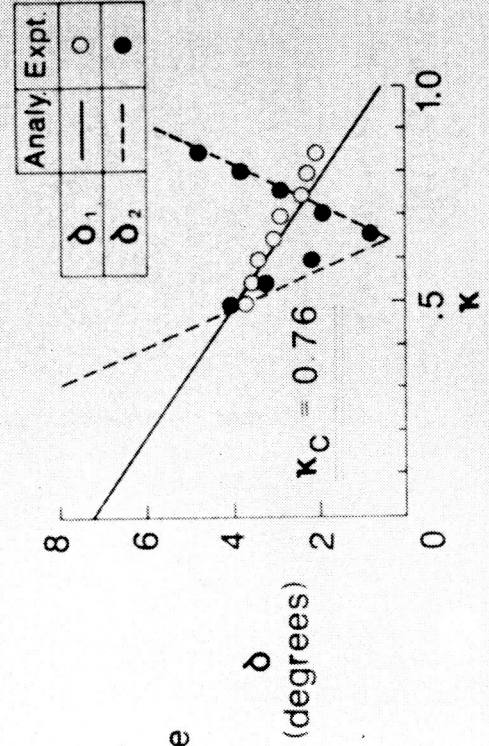
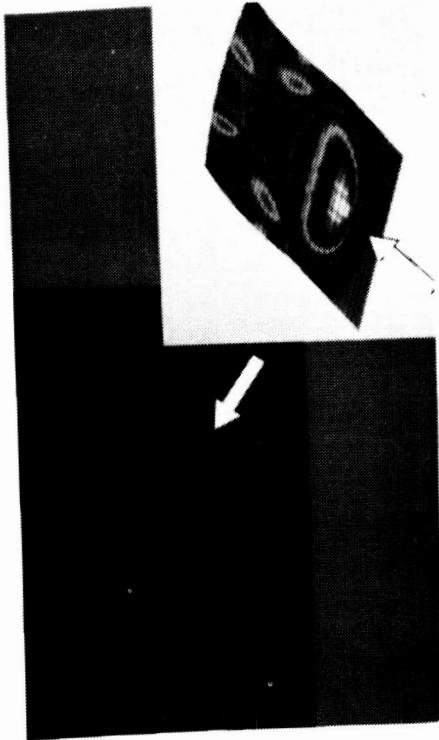


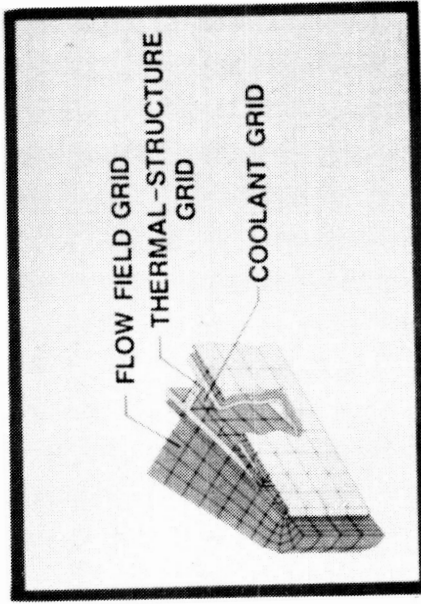
Figure 34(b).

# AEROTHERMAL LOADS

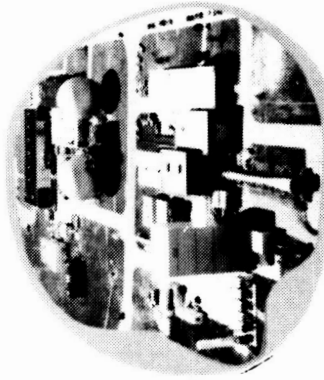
**DETAILED  
AEROTHERMAL LOADS**



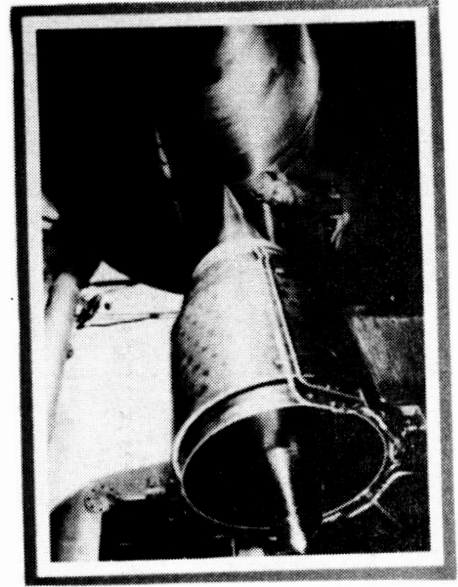
**INTEGRATED  
ANALYTICAL METHODS**



**FACILITIES**



**VALIDATION TESTS**



**STRUCTURAL CONCEPTS  
TESTS**



Figure 35.

# AEROTHERMAL LOADS FIVE YEAR PLAN

DISCIPLINARY THRUSTS	FY 87	FY 88	FY 89	FY 90	FY 91	EXPECTED RESULTS	
EXPERIMENTAL	GAPS/WAVY SURFACES/ PROTUBERANCES					DETAILED DESIGN DATA BASE	
	COMPRESSION/AXIAL CORNER FLOWS						
	SHOCK-SHOCK/SHOCK BOUNDARY LAYER						
	MASS ADDITION FLOW EFFECTS						
	NASA/INDUSTRY/DOD COOPERATIVE STUDIES						
	EULER ALGORITHMS						VALIDATED CODES
	ANALYTICAL	NAVIER-STOKES COMPRESSIBLE VISCOUS FLOW ALGORITHM					
ADAPTIVE TECHNIQUES							
INTEGRATED FLUID-THERMAL-STRUCTURAL							
INTEGRATED ANALYSIS CAPABILITY							
FACILITIES AND TEST TECHNIQUES	FACILITY OPERATION, MAINTENANCE, AND ENHANCEMENT					EFFICIENT RELIABLE FACILITIES AND TEST TECHNIQUES	
	HIGH TEMPERATURE INSTRUMENTATION						
	8' HTT MODIFICATION/AIR BREATHING PROPULSION						

Figure 36.



# AEROTHERMAL STUDY OF SIMULATED SHUTTLE CHINE TILE-GAPS IN THE 8-FOOT HIGH TEMPERATURE TUNNEL

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RTOP 506-43-31

**Research Objective** - Experimentally determine the pressures and heat transfer rates in the gaps between the reusable surface insulation tiles typical of the shuttle chine area. Extensive studies on flat surfaces have shown gap width and length and flow angularity relative to the gap to be important. Chined or curved surfaces - unlike flat surfaces - result in pressure gradients that cause greater flow ingestion into the tile gaps and hence augment the aerothermal loads. Because of the limited data base, many of the Shuttle tile chine gaps are filled to reduce the loads at a sacrifice in weight and labor.

**Approach** - The Curved Surface Test Apparatus has been developed as a generalized test bed, which is representative of the forward portion of a lifting body, for the Langley 8-foot High Temperature Tunnel (8' HTT). Chine radii of 4 and 10 inches are used to vary the surface pressure gradient. The large, 10-inch chine configuration is shown in figure 37(b). Thin-wall, metallic heat-transfer tiles were installed in selected locations adjacent to pressure instrumented solid tiles to determine the pressures and heat transfer rates in the tile gaps.

**Accomplishment Description** - Aerothermal tests in the 8' HTT at Mach 6.8, a total temperature of 3400 °R, Reynolds numbers of 0.4 to  $1.5 \times 10^6$  per foot, and angles of attack of 7, 10, and 13 degrees are complete.

**Significance** - Preliminary results indicate that the gap heating is driven by intertile steps and the gap orientation to the flow just as was the case for flat surfaces. The maximum heating occurs at the junction of the longitudinal gap terminating at a transverse gap (T-gap) as shown in the figure. From T-gap 1 to 2 the gap impingement heating decreased because of increased local flow angularity relative to the gap. Also, forward facing steps caused by a downstream tile being raised above the upstream tile augmented the local gap heating. For both configurations, selectively placed gap flow-stoppers were found to be effective in reducing the gap heat loads.

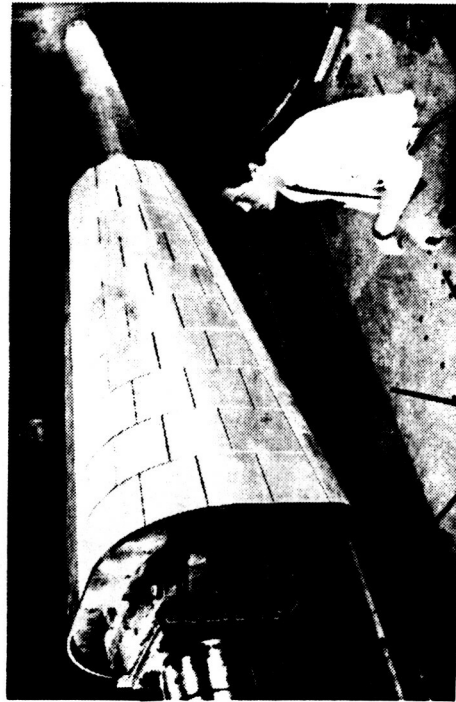
**Future Plans** - Test results are being analyzed for formal NASA publication.

Figure 37(a).

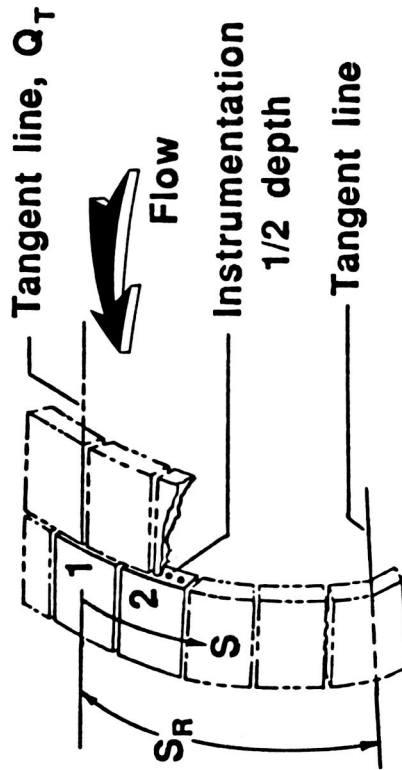
# INTERTILE STEP AND GAP ORIENTATION DRIVES CHINE GAP HEATING

8'HTT, M = 6.8, Angle of attack =  $-10^\circ$

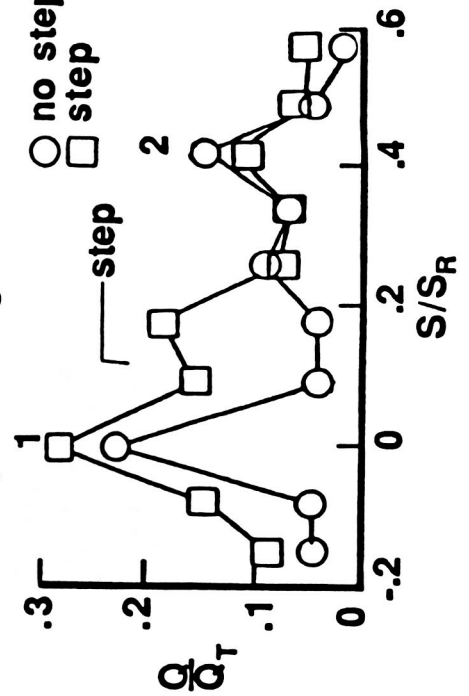
Model



Tile Array



Gap Heating Distributions



Drivers

- Maximum Q at T-Gaps
- Forward facing steps augment Q
- Q is a function of local flow angle

Figure 37(b).

# LEADING EDGE SWEEP REDUCES PEAK PRESSURE AND HEAT TRANSFER RATE

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RTOP 506-43-31

**Research Objective** - Shock-wave interference heating on the cowl-lip of an airbreathing hypersonic engine is a critical thermal structural design problem. A generic 0° sweep wedge (incident-shock generator) - cylinder (bow-shock generator) shock-wave interference model has been tested, and the experimental results show that extremely high pressure and heat transfer rate amplification occurs in highly localized regions on the cylinder surface due to shock-wave interference. A swept wedge-cylinder shock-wave interference model was tested to provide experimental data to compare with existing unswept data and show reductions, if any, in surface pressure and heat transfer rate as a means of decreasing the extremely high aerothermal loads that occur in localized regions on the cylinder surface.

**Approach** - A two-dimensional 12.5° wedge-cylinder shock-wave interference model was modified to sweep at angles of 15° and 30°. The swept model was tested in the CALSPAN 48-Inch Hypersonic Shock Tunnel (48" HST) at a Mach number = 8, unit Reynolds Number per foot of 1.4 million, dynamic pressure of 800 psf, and a total temperature of 2800 °R. The 3-inch diameter cylinder of the swept model was located at various positions relative to the wedge to produce the different shock-wave interference patterns, and detailed surface loads on the cylinder were measured.

**Accomplishment Description** - The swept wedge-cylinder model tests provide experimental pressure and heat transfer rate measurements on a highly localized region of the cylinder. Although schlieren photography of the various swept shock-wave interference patterns was not available in the 48" HST, the 55 sensor measurements do show the surface loads produced by these interference patterns. A preliminary assessment of the experimental data has been accomplished, and the locus of normalized peak pressure and heat transfer rate amplifications for sweep angles of 15° and 30° at various cylinder positions are plotted on figure 38(b) and are compared with experimental data taken from the 0° sweep test results.

**Significance** - The preliminary results of the 15° and 30° swept model tests show the peak pressure and heat transfer rate amplification decreases by 25% and 40% with increasing sweep angle. Therefore, the extremely high localized aerothermal loads produced by shock-wave interference patterns can be reduced by sweeping a configuration into the flowfield. However, the plotted results do show data scatter and more data analysis is required.

**Future Plans** - Limited experimental results obtained from the swept model tests show general data trends; however, further testing is needed on this model to fully describe the locus of peak pressure and heat transfer rate amplification and to obtain schlieren photography of the skewed shock-wave interference pattern (CALSPAN 96" HST has this capability). Also, an analytical effort is being planned for the three-dimensional swept model configuration to supplement the experimental results by providing flow field shock interference patterns.

Figure 38(a).

# LEADING EDGE SWEEP REDUCES PEAK PRESSURE AND HEAT TRANSFER RATE AMPLIFICATIONS

CALSPAN 48" HST, MACH 8, WEDGE ANGLE = 12.5°

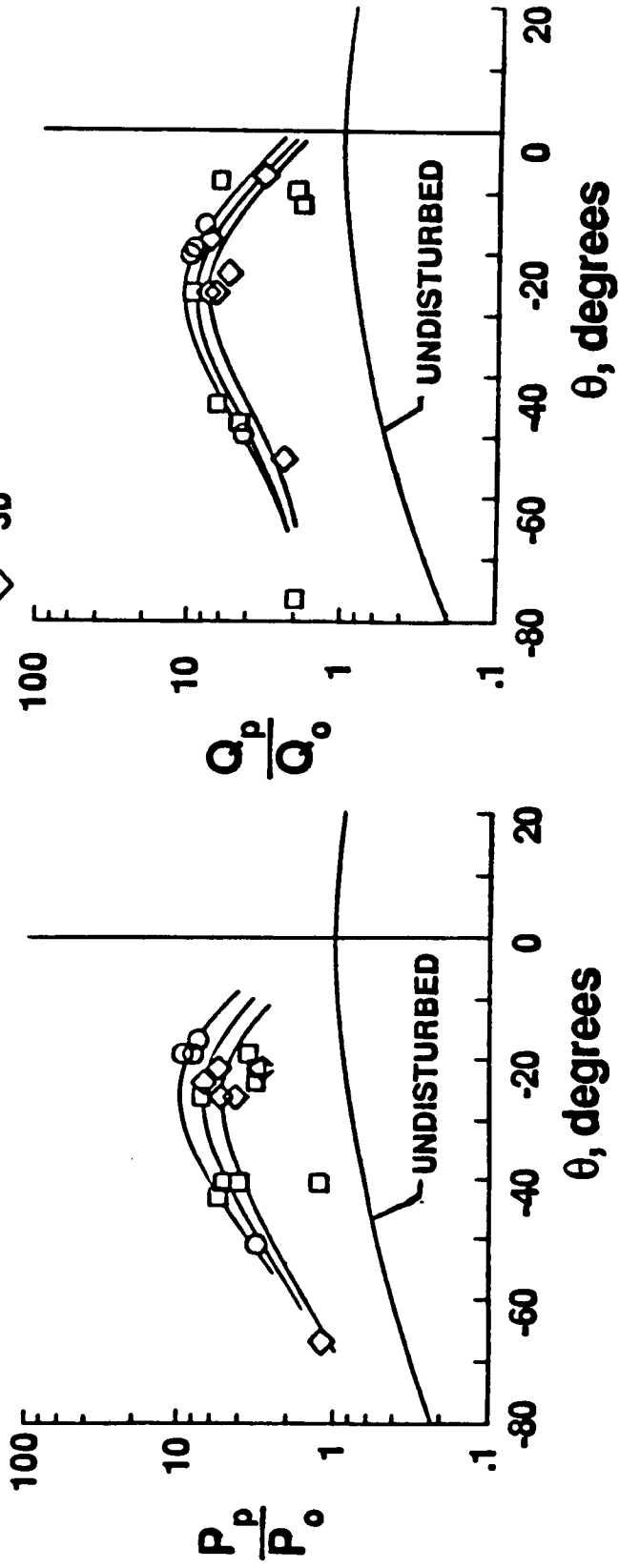


Figure 38(b).

# ADVANCED ACTIVELY COOLED RADOME PERFORMANCE DEMONSTRATED IN THE LaRC 8-FT HTT

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RTOP 506-43-31

**Research Objective** - The objective was to Support the Army Strategic Defense Command in verification of an actively cooled radome for a hypersonic interceptor.

**Approach** - Experimentally verify the performance of the transpiration cooling system and the ability of the radome to transmit and receive with low RADAR signal loss. The model, shown in the figure installed in the Langley 8-Foot High Temperature Tunnel (8' HTT) test section, was tested at Mach 6.8 at a dynamic pressure of 1400 psf and a stagnation temperature of 3500 °R. The radome consisted of discrete cylindrical waveguides, which are the white areas shown in figure 39(b). The radome was cooled by water transpired through discrete orifices upstream of each waveguide. The water vaporized at the surface absorbing the aerodynamic heating and thus protecting the radome. An uncooled replicate cone was used to obtain baseline heating rates to assess cooling performance and establish coolant flow rates for the experiment. Boundary layer trips insured that the boundary layer was always turbulent. RADAR signals were transmitted from the model to receiving horns just outside the flowfield with the signal attenuation measured.

**Accomplishment Description** - The tests verified the radome RADAR and the thermostructural performance. The RADAR signal loss increased with the coolant flow above the ideal rate. The ideal flow rate is the maximum coolant flow rate which results in all of the coolant vaporizing in the boundary layer. Complete blackout of the signal would occur when the coolant flow exceeded about 70% of ideal. The coolant distribution manifold requires redesign to account for non-uniform heat flux and to minimize heat gain by the coolant before injection. This redesign is needed to avoid vaporization in the coolant passages and thus coolant starvation in the higher heat flux regions near the front of the cone.

**Significance** - The test demonstrated a major technical advance for radomes in that they can function at extreme values of Mach number and enthalpy. Good signal propagation is now possible which is extremely important for control/guidance of this type of interceptor. Radome survival and structural performance will require some additional work to completely validate the design.

**Future Plans** - Tests of a modified radome may be requested by the Army Strategic Defense Command in the future.

Figure 39(a).

ACTIVELY COOLED RADOME PERFORMANCE  
VALIDATED IN LaRC 8' HIGH TEMPERATURE TUNNEL AT M=7

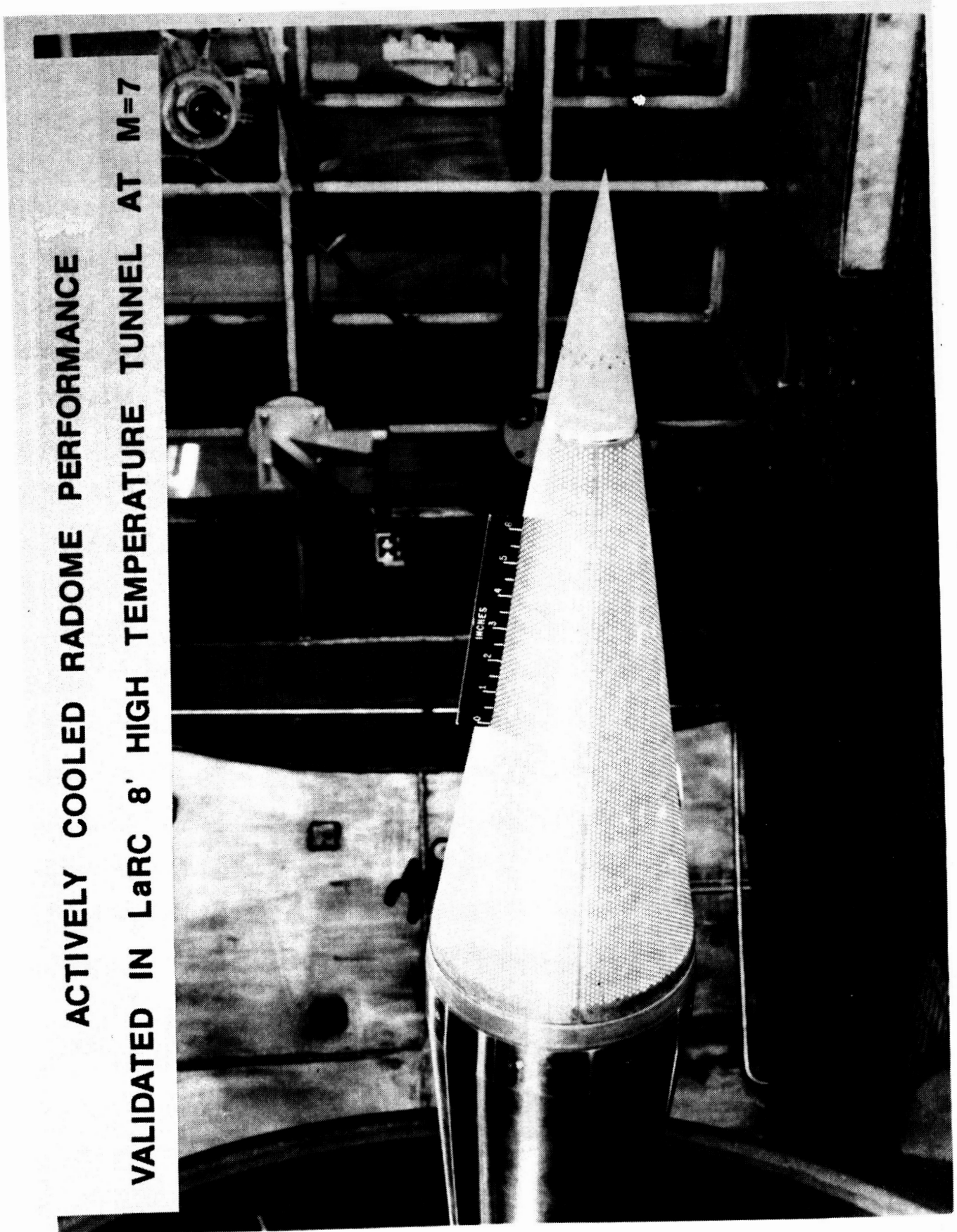


Figure 39(b).

# ADAPTIVE UNSTRUCTURED REMESHING IMPROVES COMPUTATIONAL EFFICIENCY

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RTOP 506-40-21

**Research Objective** - The objective is to develop an adaptive unstructured mesh generator for efficient fluid flow analysis.

**Approach** - Finite element methods can be implemented on unstructured grids, a feature which can be exploited in both mesh generation and adaptivity. A sophisticated mesh generator has been developed to solve 2-D computational domains of arbitrary shape with a mesh of triangular elements. The generator allows element stretching in the direction of significant one-dimensional flow features. As with most adaptive processes, the objective is equal distribution of the error. The major modification in this research is that the local directionality of the error is taken into account. This permits a local direction of stretching in order to better align elements with the flow (bottom of figure 40(b)). The adaptive process is thus able to produce an improvement in the solution quality without dramatically increasing the number of nodes in the mesh. For an arbitrary 2-D region the mesh generation process begins by constructing a 'background grid' of linear triangular elements which completely encompasses the solution domain. The necessary mesh generation parameters are the local mesh spacing, the local direction of stretching, and the local amount of stretching. Nodal values of these parameters on the background grid must be specified. These are user selected for an initial mesh or interpolated from a solution on a previous mesh. Gradients of density, mach number or other flow variables or a combination of these can be used for the nodal parameters. The boundaries are defined in terms of Bezier polynomials. The generator begins by placing nodes on these boundaries, according to the interpolated mesh spacing. The lines joining successive boundary nodes form the initial 'front' (the collection of sides available to form new triangular elements). One side in the front is selected and a triangle constructed by either placing a new node or selecting an existing node in the front, depending on interpolated values and geometric constraints. The front is progressively updated and the process continued until the entire domain has been discretized.

**Accomplishment Description** - The remeshing technique has been very successfully applied to Euler and Navier-Stokes solutions of shock impingement on a blunt body used to simulate the cowl lip of a scramjet on a hypersonic vehicle as shown in the figure. Predictions compare very well with experimental wall pressure and heating rate augmentations observed in the 8-foot High Temperature Tunnel and Calspan Hypersonic Shock Tunnels.

**Significance** - This remeshing approach, which permits adaptive remeshing based on flow solutions on previous meshes, by placing nodes at optimum locations in the flowfield, significantly reduces the number of unknowns in the solution domain resulting in improved computation efficiency.

**Future Plans** - The scheme is being extended to 3-D. This includes changes in strategy for more efficient searching and updating algorithms.

Figure 40(a)-

# ADAPTIVE UNSTRUCTURED REMESHING IMPROVES COMPUTATIONAL EFFICIENCY

SCHLIEREN PHOTOGRAPH OF FLOW PHENOMENA TO PREDICT



FINAL UNSTRUCTURED MESH ADAPTED TO FLOW PHYSICS



$$\frac{\text{SMALLEST}}{\text{LARGEST}} = \frac{1}{1000}$$

- INITIAL MESH UNIFORM
- 3-5 REMESHES FOR CONVERGENCE

SHOCK FRONT

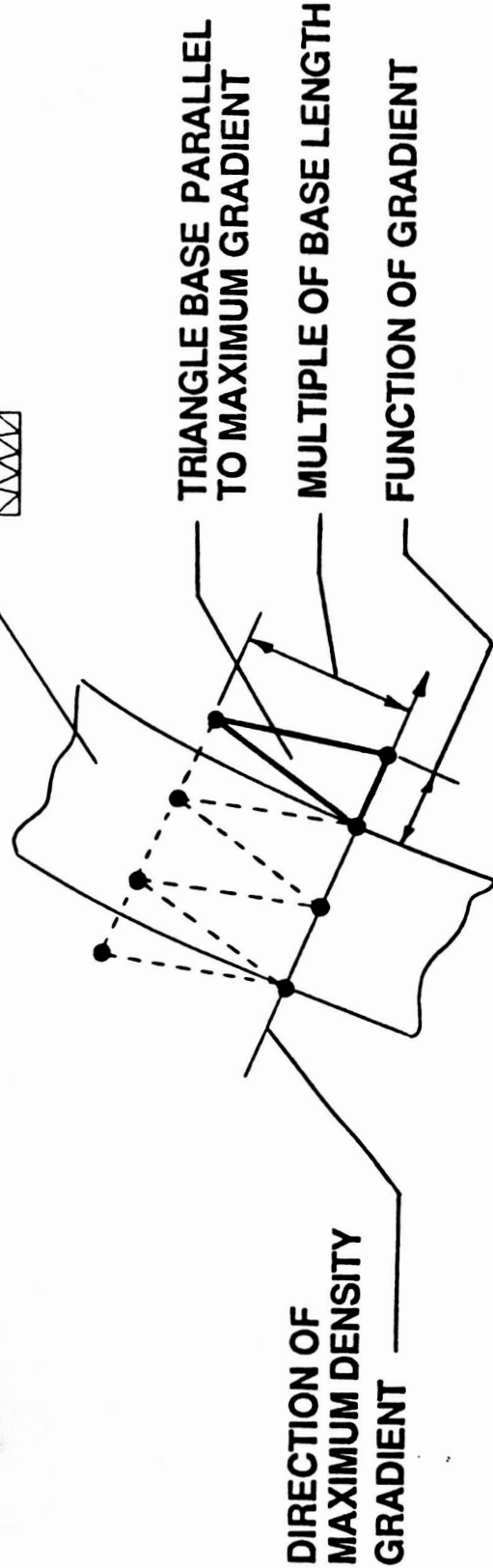


Figure 40(b).



# FINITE ELEMENT INVISCID RESULTS IN GOOD AGREEMENT WITH EXPERIMENTAL RESULTS

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Extension 2229  
RTOP 506-40-21

**Research Objective** - To demonstrate that the inviscid features of a "Type IV" shock interference pattern and the resulting surface pressure distribution can be accurately and efficiently predicted by a finite element model coupled with an adaptive unstructured remeshing technique.

**Approach** - A "Type IV" shock interference pattern can occur when a planar oblique shock impinges the leading edge bow shock of a cylindrical leading edge representative of a hypersonic engine inlet cowl. This flow field is mathematically modeled using the finite element method to solve the compressible Euler equations for a two-dimensional computational domain. The cowl leading edge is represented by a 3"-diameter cylinder and the impinging oblique shock is generated by a 12.5° wedge. Solutions are obtained on adaptive unstructured meshes of triangular elements. The initial mesh is approximately uniform, while subsequent improved meshes are adapted to the physics of the flow based on the density gradients in each element. Four meshes were adequate to resolve the "Type IV" supersonic jet shock interference pattern. The final mesh had approximately 4000 grid points. The solution is converged on the final mesh where approximately 60% of the computational effort occurs.

**Accomplishment Description** - The finite element method is shown through comparison with experimental data to accurately capture the flow physics and predict the surface pressure distributions. The adaptive unstructured remeshing technique concentrates elements in regions of large gradients and removes elements in regions of small gradients to minimize computational and model generation efforts. Predicted flow field density contours for a Mach 8.0 flow at a stagnation temperature of 2790 °R are compared with the experimental schlieren photograph shown in figure 41(b). Due to the calorically perfect gas assumption, the predicted bow shock is slightly further from the cylinder. The length of the transmitted shock and the jet appear to be predicted within the resolution of the schlieren photograph. The predicted surface pressure distribution, also shown in the figure, is in excellent agreement with the experimental data.

**Significance** - This analysis demonstrates that complex inviscid flow problems can be efficiently solved using the finite element method combined with an adaptive remeshing procedure. The benefit is a significant savings in computational time and costs.

**Future Plans** - Problems for the future include: (1) viscous calculations including smaller cylinder diameters, (2) multiple impinging shocks, and (3) three dimensional swept cylinder configurations.

Figure 41(a).

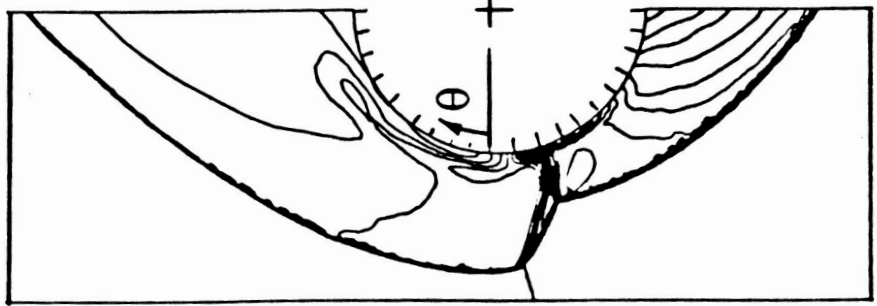
# FINITE ELEMENT INVISCID RESULTS IN GOOD AGREEMENT WITH EXPERIMENTAL RESULTS

TAYLOR GALERKIN ALGORITHM - ADAPTIVE UNSTRUCTURED REMESHING  
CALORICALLY PERFECT GAS

SCHLIEREN PHOTOGRAPH



PREDICTED DENSITY CONTOURS



PREDICTED SURFACE PRESSURE DISTRIBUTION

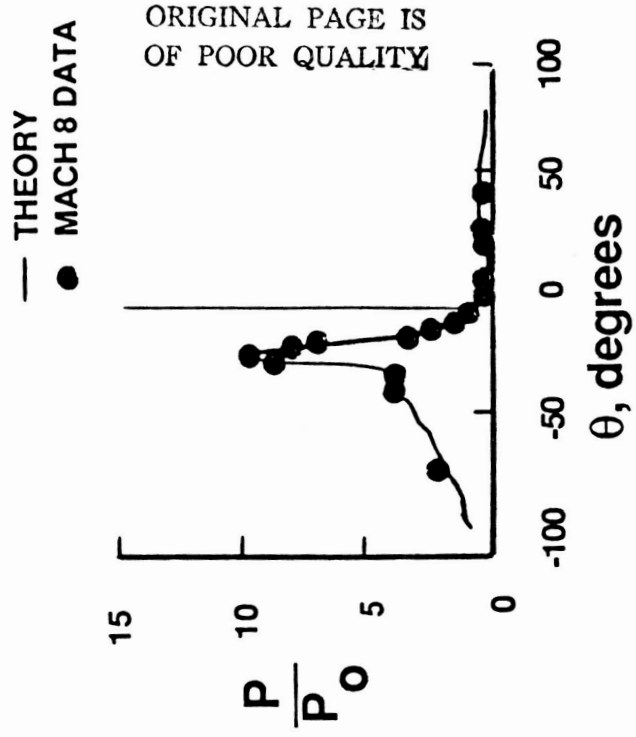


Figure 41(b).

# LANGLEY INTEGRATED FLUID THERMAL STRUCTURAL ANALYSIS OF BLUNT LEADING EDGE

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RTOP 506-43-31

**Research Objective** - Develop a fully integrated fluid-thermal-structural analysis capability for accurate prediction of the aerothermal loads, structural temperatures, deformations and stresses for aerospace applications.

**Approach** - The most efficient approach for the analysis of interdisciplinary problems is to use a single model and a single algorithm for all three disciplines. In this approach, the Navier-Stokes equations for laminar, compressible flow are solved together with the energy equation and the equilibrium equations for the structure using a Taylor-Galerkin finite element formulation. Radiation heat transfer at the fluid-structure interface is included, and nonlinear inelastic deformation of the structure is permitted in the analysis.

**Accomplishment Description** - The integrated analysis capability has been developed and exists in the Langley Integrated Fluid Thermal Structural (LIFTS) analyzer. The capability of the LIFTS analyzer to solve all three disciplines is illustrated in figure 42(b) of a blunt leading edge exposed to Mach 6.5 flow where experimental data are available for comparison. The code accurately predicts the flow solution which includes the bow shock, the aerodynamic pressure and heat flux. With these aerothermal loads, the leading edge temperature distributions, deformations and stresses are then computed as shown by the unquantified contours on quarter-segments of the cylinder. Two versions of the LIFTS analyzer now exist: (1) a vectorized version for the Langley VPS-32, and (2) a scalar version for the Ames NAS Cray-2.

**Significance** - The results illustrated in the figure demonstrate the capability of LIFTS to use a single methodology in a single code to perform analyses for all three disciplines. This capability, to our knowledge, is the first of its kind and is an important milestone toward coupled flow-thermal-structural analysis for more complicated aerospace applications.

**Future Plans** - Future plans include implementation of an efficient adaptive mesh refinement procedure for improved solution accuracy, an implicit/explicit solution algorithm for faster solution convergence, and computational algorithms for prediction of the thermal plasticity behavior of metallic and composite structures.

Figure 42(a).

# LANGLEY INTEGRATED FLOW THERMAL STRUCTURAL (LIFTS) ANALYSIS OF BLUNT LEADING EDGE

## FLOW THERMAL STRUCTURAL

SHOCK  
SHAPE

AEROTHERMAL  
LOADS

TEMPERATURE  
CONTOURS

STRESS  
CONTOURS

— THEORY  
● EXPERIMENT

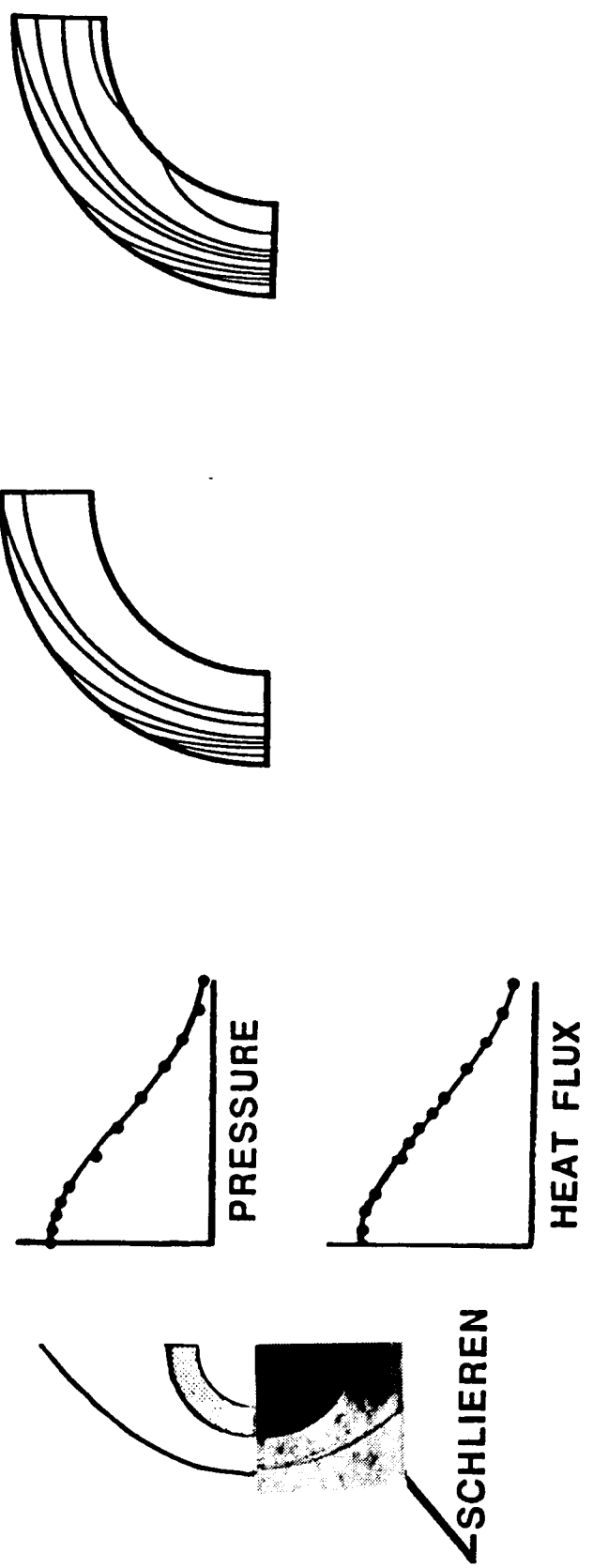
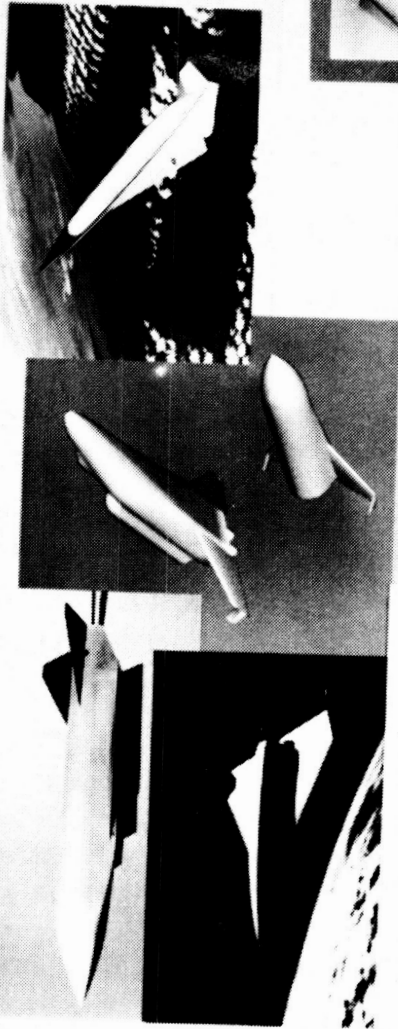


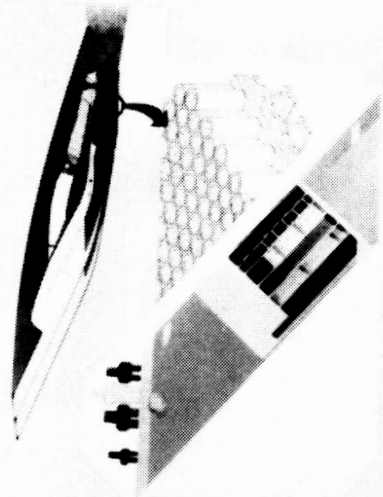
Figure 42(b).

# THERMAL STRUCTURES

Concept Development  
and Verification



System Studies



Thermal Structural  
Analysis

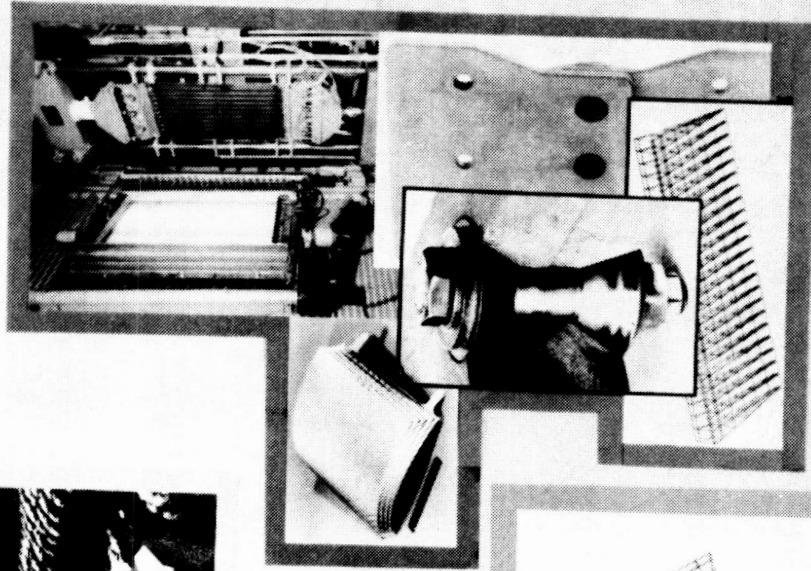


Figure 43.

# THERMAL STRUCTURES FIVE YEAR PLAN

DISCIPLINARY THRUSTS	FY 87	FY 88	FY 89	FY 90	FY 91	EXPECTED RESULTS
<b>COOLED STRUCTURES</b>	<b>SCRAMJET STRUT TESTS</b>					<b>DEMONSTRATE ADVANCED CONCEPTS</b>
	<div style="border: 1px solid black; padding: 5px; display: inline-block; margin-bottom: 5px;">STRUT LIFE CYCLE ASSESSMENT</div>					
	<div style="border: 1px solid black; padding: 5px; display: inline-block; margin-bottom: 5px;">ACTIVELY COOLED STRUCTURES</div> <div style="border: 1px solid black; padding: 5px; display: inline-block; margin-left: 100px; margin-top: 5px;">H<sub>2</sub> COOLED VEHICLE ANALYSIS DEMO</div>					
<b>HOT/CRYO STRUCTURES</b>	<b>C/C TECHNOLOGY</b>					<b>DEMONSTRATE MASS EFFICIENT TPS/STRUCTURE/TANK CONCEPTS</b>
	<b>METALLIC TPS DEVELOPMENT</b>					
	<div style="border: 1px solid black; padding: 5px; display: inline-block; margin-bottom: 5px;">STRUCTURE/CRYO TANK DEVELOPMENT/ALS</div> <div style="border: 1px solid black; padding: 5px; display: inline-block; margin-left: 100px; margin-top: 5px;">SHUTTLE TPS/STRUCTURE</div> <div style="border: 1px solid black; padding: 5px; display: inline-block; margin-left: 100px; margin-top: 5px;">CONTROL SURFACES</div> <div style="border: 1px solid black; padding: 5px; display: inline-block; margin-left: 100px; margin-top: 5px;">INTEGRATED TPS/STR/TANK DEMONSTRATION</div>					
<b>ANALYSIS &amp; SYNTHESIS METHODS</b>	<b>INTEGRATED T-S ANALYSIS</b>					<b>INTEGRATED ANALYSIS METHODS</b>
	<b>DESIGN ORIENTED ANALYSIS TECHNIQUES</b>					
	<div style="border: 1px solid black; padding: 5px; display: inline-block; margin-bottom: 5px;">PREMIER INTEGRATED DESIGN/ANALYSIS CAPABILITY</div> <div style="border: 1px solid black; padding: 5px; display: inline-block; margin-left: 100px; margin-top: 5px;">INTERDISCIPLINARY DESIGN/ANALYSIS METHODS/AEROTHERMOELASTICITY</div>					

Figure 44.

# METHODOLOGY FOR DESIGNING COOLED ENGINE WALL STRUCTURE

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RTOP 505-63-31

**Research Objective** - The resurgence of research into hypersonics has spurred a renewed interest in high temperature structures. Because peak heating rates in the combustor of a scramjet can exceed 2000 Btu/ft<sup>2</sup>-s, an active system that cools the walls using surface heat exchangers (cooling jackets) is required. A large number of parameters must be considered simultaneously when designing an active cooling system to meet the following: structural criteria, such as limiting stresses, temperatures, and fatigue life, thermal criteria, such as temperature gradients, pressure drops, and coolant Mach number; and performance requirements, such as minimizing the required coolant. The objective of this research has been to develop methodology that considers the above factors to design "optimal" surface heat exchangers.

**Approach** - The criteria discussed above can be described mathematically as a constrained optimization problem. Numerical optimization techniques are fairly well developed and may be the only practical approach to the optimum design of a cooling jacket system because of the many design parameters. Recently, these techniques have been incorporated into a computer language (Sizing and Optimization Language) that greatly simplifies their usage. This computer language has been chosen as a test bed for the methodology development. The design must satisfy the design constraints which are determined by simple models such as one dimensional heat transfer analysis, handbook level stress analysis, and simple coolant film and friction coefficient relationships.

**Accomplishment Description** - Models appropriate to both channel fin and pin-fin cooling jackets have been developed. Specific results have been generated for a system of eleven engine panels of a scramjet engine. Two distinct design cases for the cooling system have been optimized (see figure 45(b)): a nickel cooling jacket and titanium cooling jacket case. The nickel cooling jacket system required less coolant than the titanium due to its higher thermal conductivity, but was heavier. Another optimum design was evaluated which added a design criteria that limited the weight of the nickel design to the weight of the titanium design. The best nickel design with this weight limit required only a small increase in the coolant flow. The parametric analysis was performed with the addition of only one computer statement to the cooling system model. This method of changing constraints to determine parametric sensitivities has been found to be very useful.

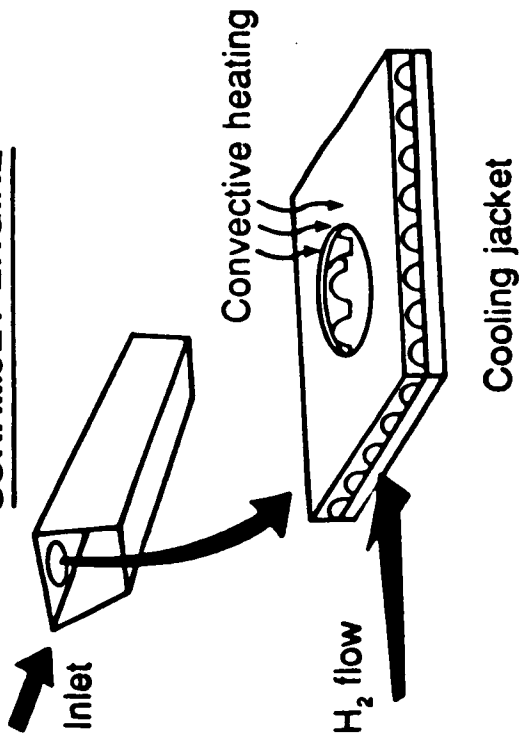
**Significance** - There is little design experience in designing flight-weight active cooling systems. The only guidance for designers is the results of their analyses which may be for conditions far from the optimum design or may neglect important design considerations. The methodology developed considers most of the important design requirements simultaneously in designing active cooling systems, and can quickly address sensitivity issues, such as effects of material changes or different limits on various design parameters.

**Future Plans** - Future work will focus on the feasibility of including thermal stresses of the engine primary structure as a constraint imposed on the design.

Figure 45(a).

# METHODOLOGY FOR DESIGNING COOLED ENGINE WALL STRUCTURE

## SCRAMJET ENGINE



## OBJECTIVE

- Define cooling jacket geometry for engine walls
- Calculate H<sub>2</sub> flow rates and pressures
- Determine minimum H<sub>2</sub> flow or minimum jacket weight
- Satisfy multiple constraints (e.g. wall temperature, stress, H<sub>2</sub> Mach no., and pressure drop)

## NEW ANALYSIS CAPABILITY

- Developed sizing and optimization computer language
- Analyzes 100 variables simultaneously
- Performs sensitivity studies
- Discovers common errors automatically

## RESULTS

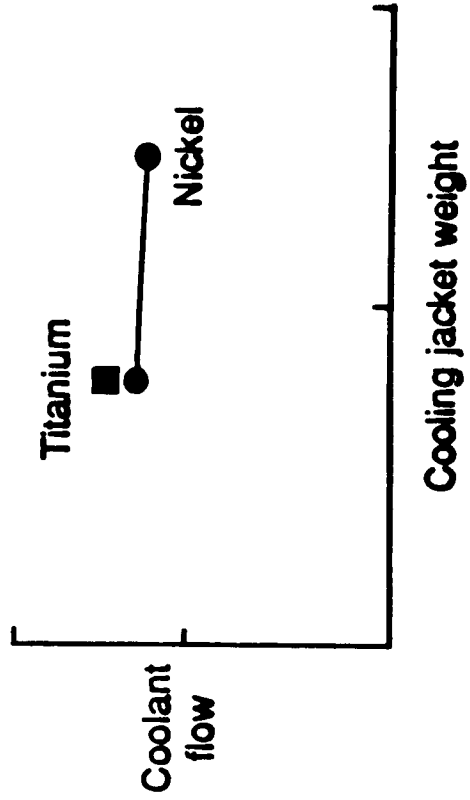


Figure 45(b).



# CARBON-CARBON/REFRACTORY-METAL HEAT-PIPE STRUCTURE EXTENDS THE USE OF CARBON-CARBON IN WITHSTANDING HIGH HEAT LOADS

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Extensions 3665 and 4412

RTOP 506-49-11

**Research Objective** - Future hypersonic vehicles such as the National Aero-Space Plane (NASP) will experience high heating rates which will cause temperatures to exceed the maximum structural reuse temperature of carbon-carbon (currently carbon-carbon is limited by its oxidation protective coating to temperatures below 2800 °F). This study is to extend the maximum heat flux capability of carbon-carbon structures by using embedded heat pipes and thereby provide a simple, reliable, fail/safe, and potentially lighter alternative for the nose cap and wing leading edge of NASP.

**Approach** - The design philosophy of the present concept utilizes the high specific strength of carbon-carbon at elevated temperatures to accommodate the thermal/structural loads and use very thin refractory-metal D-shaped, heat pipes ( 5 mills thick), embedded within the structure (see figure 46(b)) to transport the stagnation heat aft, where it can be rejected by radiation. The heat pipes could be sized and spaced close enough so that in the event of a failure, the ablation protection afforded by the carbon-carbon would be sufficient to enable a safe reentry. The preferred method of construction would be to fabricate, fill, and checkout the individual heat pipes, embed them in a 2-D layup of carbon-carbon prepreg and process (pyrolyze and density) as you would a typical carbon-carbon component.

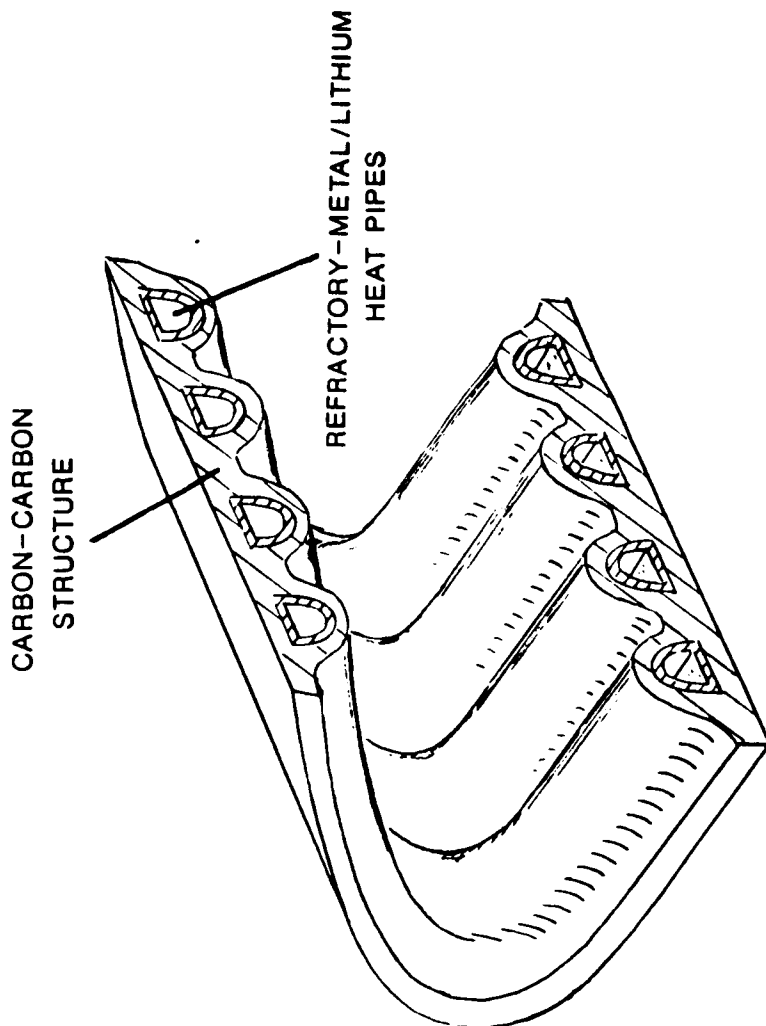
**Accomplishment Description** - 2-D thermal/structural analysis indicates the feasibility of the concept. Reaction rate studies of refractory metals, (molybdenum, tungsten, and tantalum), indicate that tungsten is the least reactive with carbon, and because of its very high thermal conductivity and low coefficient of thermal expansion, it is the leading candidate for a container for the lithium working fluid. Tungsten tubes (0.03 in. thick) have been successfully embedded within 2-D carbon-carbon prepreg, pyrolyzed and densified three times without any noticeable damage to the composite.

**Significance** - The proposed concept can extend the use of conventional carbon-carbon structure to withstand much higher local heating rates, such as those predicted for the wing leading edge of NASP, by significantly reducing peak stagnation temperatures. Also, the current concept offers an ablative/fail-safe capability in the event of a heat pipe failure.

**Future Plans** - Three-dimensional finite-element thermal/structural analyses will be used to design/synthesize a carbon-carbon/refractory-metal heat-pipe leading edge for NASP. Several flat specimens (three heat pipes wide) will be fabricated and tested, thermally and mechanically, to verify fabrication methods. A six-inch span segment of the leading edge will then be fabricated and tested to verify concept performance.

Figure 46(a).

# CARBON-CARBON/REFRACTORY-METAL HEAT-PIPE CONCEPT



## DESIGN FEATURES

- C-C PRIMARY STRUCTURE  
STRONG AND LIGHTWEIGHT AT 3000 F
- THIN REFRACTORY-METAL HEAT PIPES  
REDUCE LOCAL STAGNATION HOT SPOTS  
TRANSPORT AND RADIATE HEAT
- MAXIMIZE HEAT PIPE OPERATING TEMPERATURE  
TO MINIMIZE SIZE/MASS

## MATERIALS

CARBON-CARBON  
2-D PREPREG  
3-D WEAVE

REFRACTORY METALS  
TUNGSTEN  
MOLYBDENUM  
COLUMBIUM

CARBON-CARBON/REFRACTORY-METAL  
HEAT-PIPE LEADING EDGE

## BLADE-STIFFENED CARBON-CARBON COMPRESSION PANELS

James Wayne Sawyer  
Thermal Structures Branch  
Extension 4201

RTOP 506-43-31

**Research Objective** - Carbon-carbon composite materials are being considered for use as hot structures on advanced aerospace vehicles. Stiffened panels are basic to most aerospace vehicle structures and understanding their behavior in compression is essential for the design of efficient carbon-carbon structural components. This objective is to develop the technology to design and fabricate efficient carbon-carbon stiffened compression panels.

**Approach** - Use the structural panel analysis and sizing code PASCO to determine efficient carbon-carbon panel designs. Fabricate full size carbon-carbon panels of one of the designs to develop the fabrication techniques and to provide test specimens. Instrument and load the panel specimens to failure in compression to determine the failure modes and strength of the fabricated panels. Analyze the panels using the finite element code EAL and compare the predicted results with the test results. Investigate differences between the analytical and experimental results.

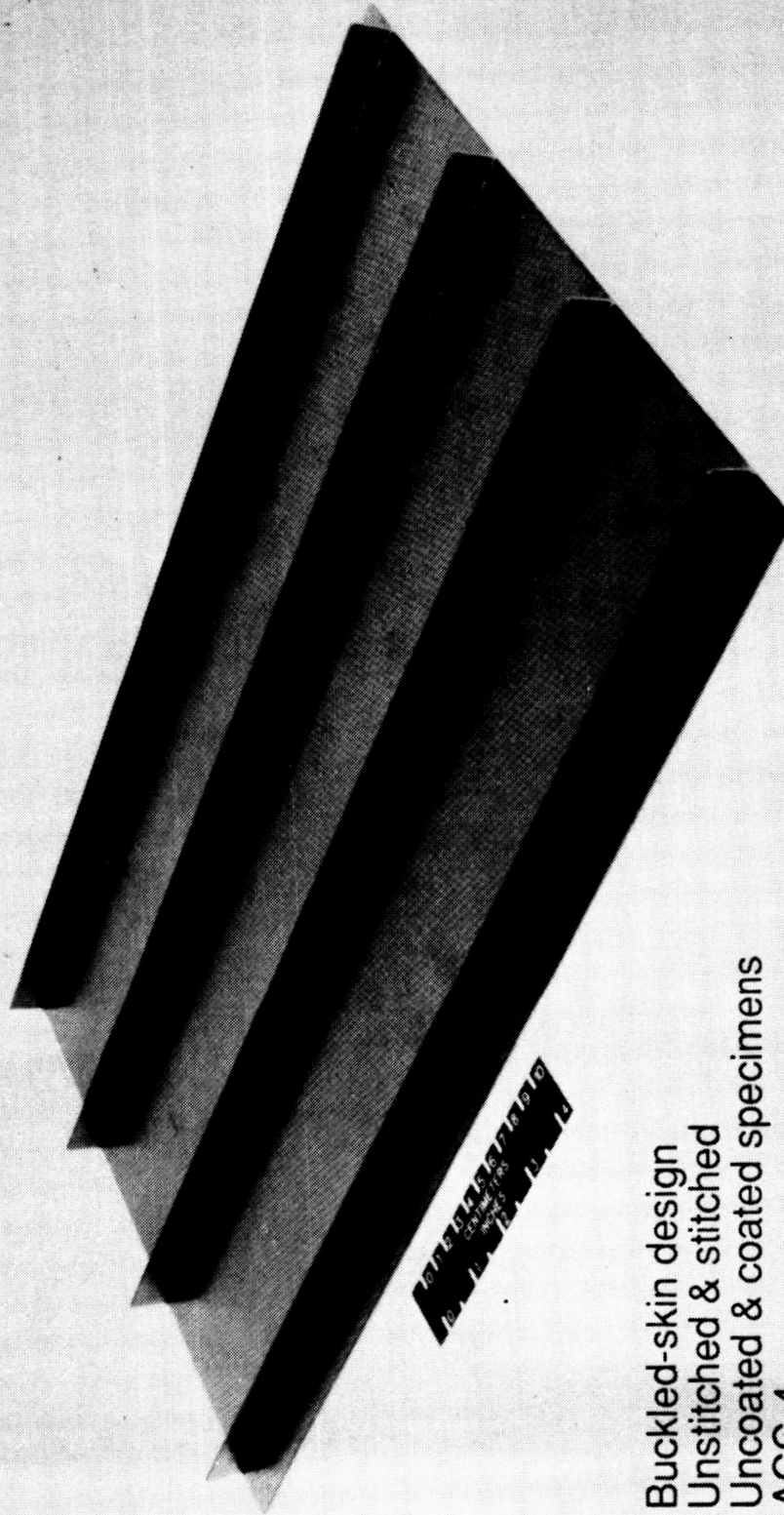
**Accomplishment Description** - Several blade and T-stiffened carbon-carbon panel concept designs have been analyzed and evaluated. A blade stiffened minimum gage design has been selected and four uncoated ACC-4 carbon-carbon panels fabricated. One of the panels is shown in figure 47(b). The panels are 12.5 inches wide by 20 inches long, and have four blade stiffeners 0.75 inches high spaced 3.50 inches apart. The panel skins and stiffeners are composed of 6-ply of T-300 fibers woven into eight harness satin weave fabric. Two of the panels have a single row of stitching along each side of the stiffeners to determine if stitching will improve the failure strength of the panels. The panels have been analyzed and are expected to have buckled skins when loaded to failure.

**Significance** - The capability to design, fabricate, and analytically predict the behavior of carbon-carbon stiffened compression panels will supply some of the technology required for the structural applications of carbon-carbon materials.

**Future Plans** - The panels will be tested and the results compared with the analytical predictions. Two additional panels of the same design are being fabricated but will be given an oxidation protection coating before testing. The coated panels will also be tested and the results compared with analytical predictions by the end of FY 1988.

Figure 47(a).

# BLADE-STIFFENED CARBON-CARBON COMPRESSION PANELS



RM-08

Buckled-skin design  
Unstitched & stitched  
Uncoated & coated specimens  
ACC-4  
6-Ply satin weave (70 mil)  
2-Dimensional construction

Figure 47(b).

# BLATT'S EQUATION USED TO CHARACTERIZE EFFECTS OF THERMAL STRATIFICATION FOR NASP VEHICLE DESIGN

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RTOP 505-63-31

**Research Objective** - Much of the weight and volume of hypersonic vehicles such as the National Aero-Space Plane (NASP) is associated with the cryogenic fuel (liquid hydrogen), tank structure and insulation. Cryo-tanks exhibit thermal stratification, which is a non-uniform temperature distribution within the liquid, such as shown on the left of figure 48(b). It results from external wall heating and internal liquid natural convection and evaporation, and causes a thermally-stable situation wherein the warmed fluid from the boundary layer flows to the top of cooler liquid. The tank pressure, determined by the vapor pressure of the warmer liquid, can rise dramatically. Fuel may need to be vented (and thus wasted) to prevent tank over-pressurization. This objective is to characterize the effect of thermal stratification on cryo-tank pressurization, design and performance.

**Approach** - A critical review of the literature on thermal stratification was conducted and showed that most of the previous experimental and analytical work was directed towards vertical launch cryo-tanks and was not appropriate to NASP-type horizontal tanks. However, one relatively simple equation was identified which appeared to be applicable to various tank geometries and sizes. This equation was proposed by Blatt (J. Spacecraft, Vol.5, No.6, June 1968), who reasoned that the pressure rise rate (e.g. psi/hr) in a closed container is proportional to the heating rate (Q), and inversely proportional to the fluid mass (M) and ullage (vapor space) fraction (S). Blatt correlated data from five liquid hydrogen tanks, both spherical and cylindrical (vertical), varying in size from 13 gal to 50,000 gal capacity.

**Accomplishment Description** - The literature search revealed results from 21 additional stratification tests, including those from four liquid hydrogen horizontal cryo-tanks, and were added to Blatt's data. This new data and the original data are shown on the right of figure 48(b). Much of the additional data fell within the region applicable to NASP and permitted greater confidence in the use of the correlation on cryo-tank designs for NASP-type trajectories, geometries and conditions.

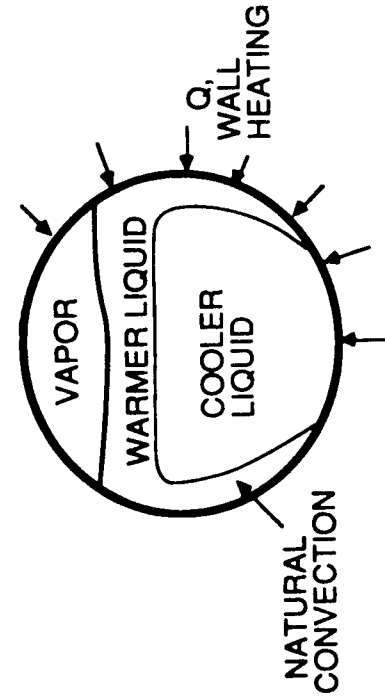
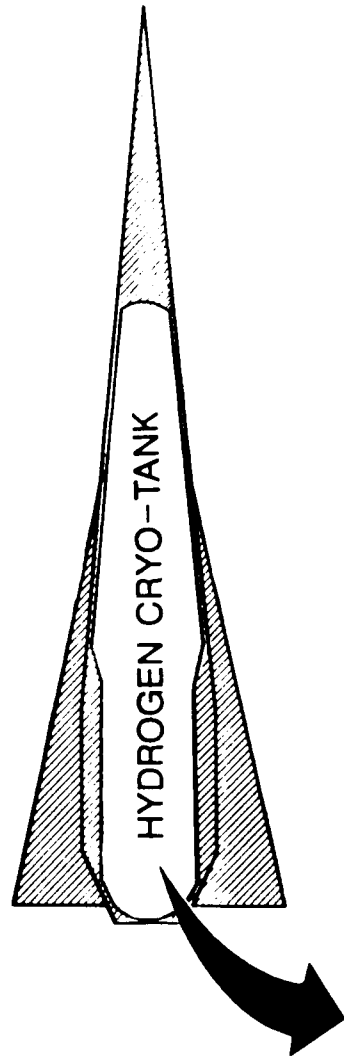
The effect of thermal stratification was investigated for a generic NASP vehicle during an ascent trajectory. Tank pressure response to fuel withdrawal and heating for two identical tank designs were examined. The first assumed that the tank contents are homogeneous and thus being heated uniformly, while the second applied Blatt's correlation to the tank. Results (figure 48(c)) show that a tank design (including insulation thickness) based on the homogeneous model is nonconservative and that stratification will require venting relatively early in the flight trajectory.

**Significance** - Hypersonic vehicle designs are very sensitive to insulation weight and fuel volume. The modified Blatt's equation provides a tool for the designer to use to examine thermal stratification and its effect on tank pressurization, fuel venting and cryo-tank design.

**Future Plans** - Future work will focus on the inclusion of temperature distributions in the ullage so that the fuel mass lost during venting and its effect on vehicle gross take-off weight can be determined. In addition, experimental work may be required to determine the effect of tank draining on the correlation parameters.

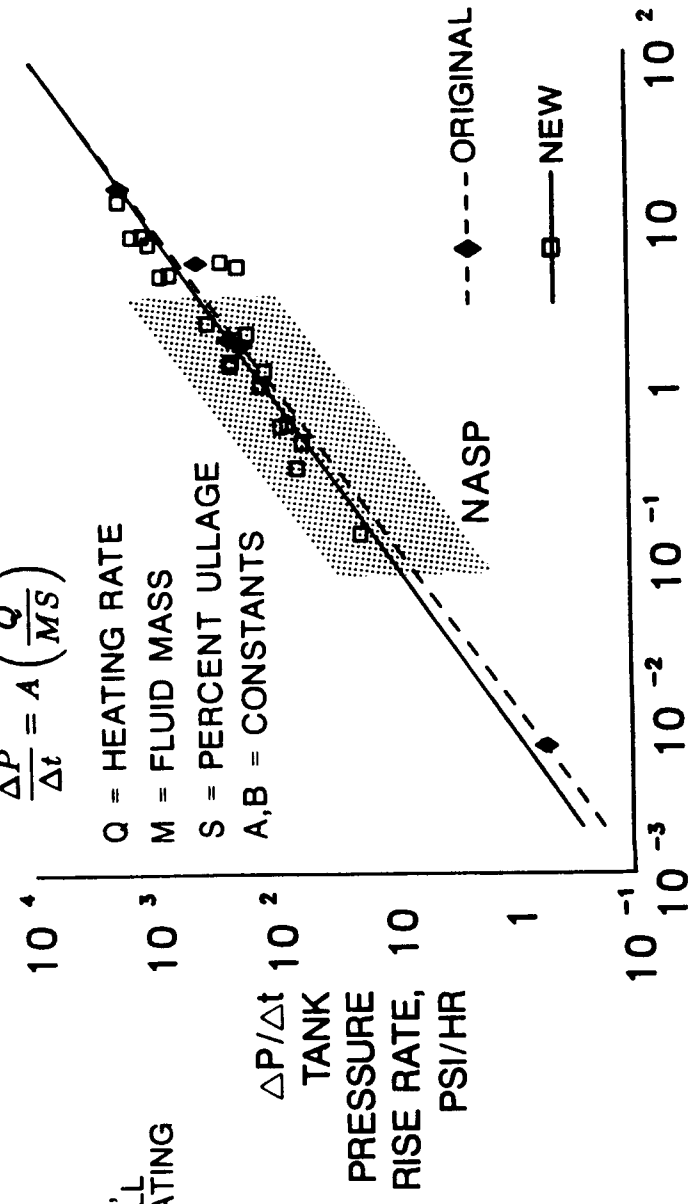
Figure 48(a).

# BLATT'S EQUATION USED TO CHARACTERIZE EFFECTS OF THERMAL STRATIFICATION FOR NASP VEHICLE DESIGN



$$\frac{\Delta P}{\Delta t} = A \left( \frac{Q}{MS} \right)^B$$

- Q = HEATING RATE
- M = FLUID MASS
- S = PERCENT ULLAGE
- A, B = CONSTANTS



BLATT'S PARAMETER, Q/MS

Figure 48(b).

# BLATT'S EQUATION USED TO CHARACTERIZE EFFECTS OF THERMAL STRATIFICATION FOR NASP VEHICLE DESIGN

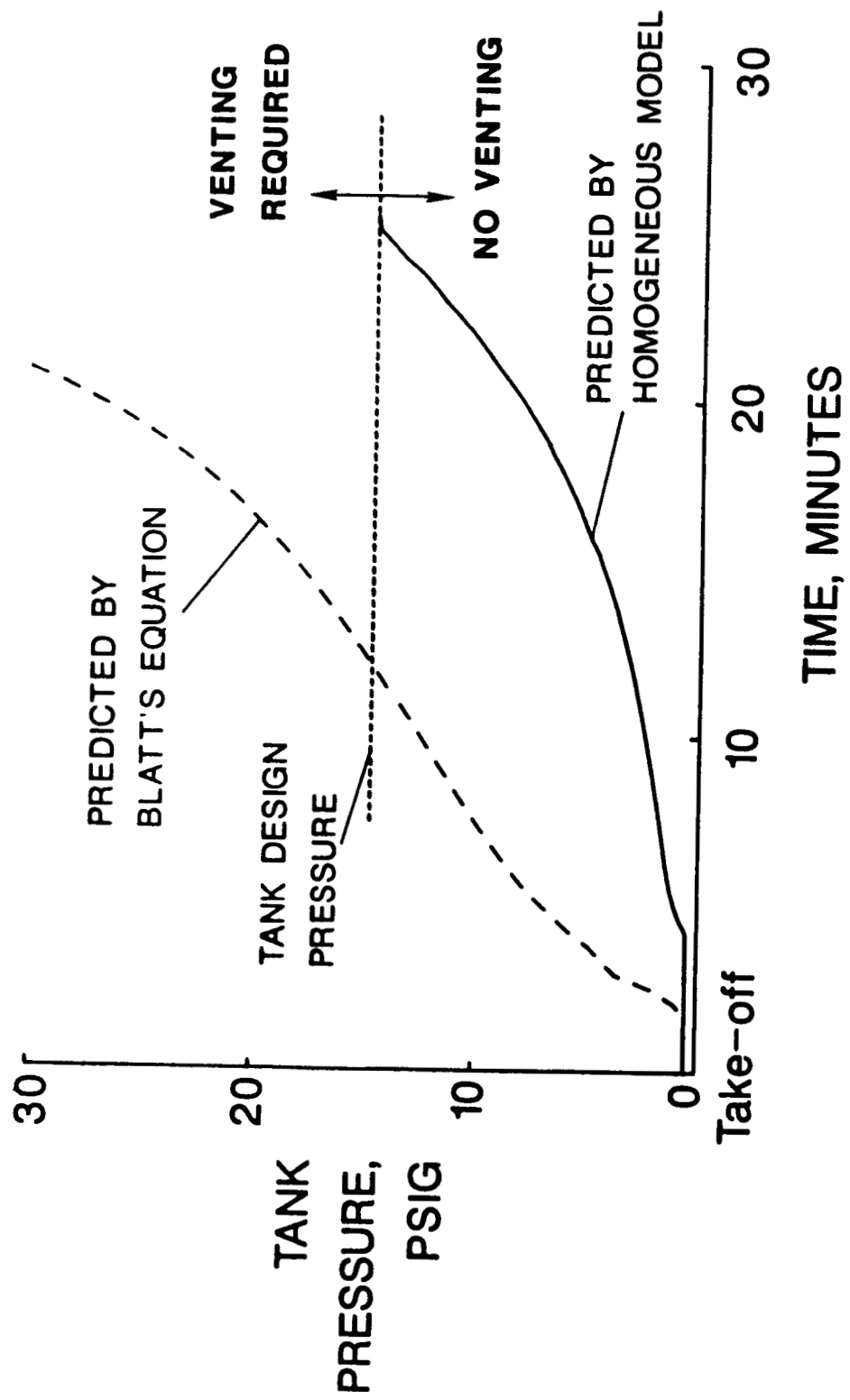


Figure 48(c).

# TRANSIENT FINITE-ELEMENT HEAT PIPE ANALYSIS PREDICTS STARTUP FROM A FROZEN STATE

Gene T. Colwell and Charles J. Camarda  
Georgia Institute of Technology and Thermal Structures Branch  
Extension 3665

RTOP 506-49-11

**Research Objective** - High temperature heat pipes have been proposed as a means of cooling the stagnation regions of hypersonic vehicles. Heat pipes offer a simple, passive, reliable, and possibly lighter-weight alternative to other, actively-cooled concepts. Since high-temperature heat pipes necessitate the use of liquid-metal working fluids, the working fluid will most likely be solid during the initial phase of the trajectory. The purpose of the present study was to develop a reliable means of solving the coupled thermal/fluid heat-pipe equations and predict heat pipe transient performance, including startup from the frozen state.

**Approach** - The heat pipe experiences distinct phases of operation during startup from the frozen state which complicates the numerical solution of the problem; (1) initially, the working fluid is solid, vapor density is so low molecular flow conditions prevail and axial heat transport by the vapor is negligible; (2) in the region of heat input, the vapor space establishes a region where continuum flow conditions prevail, the axial rate of heat transfer is extremely high, vapor velocities can reach sonic conditions and vapor compressibility effects become important; (3) the continuum front grows from the region of heat input (evaporator region) toward the condenser or heat output end of the heat pipe and eventually continuum flow conditions exist along the entire length of the heat pipe. A finite-element analysis method was chosen to solve the coupled thermal/fluid heat pipe problem because of its natural capability to interface with existing finite-element codes for combined thermal/structural analysis of heat-pipe-cooled structures.

**Accomplishment Description** - A finite-element analysis code has been developed which solves the coupled thermal-fluid heat pipe problem and is capable of accurately predicting heat pipe temperatures, temperature gradients as well as liquid and vapor pressures and velocities. Numerical results for startup of a liquid-metal heat pipe subject to a stepped heat input and representative of radiant-heat test of a shuttle-type heat-pipe-cooled leading edge are shown in figure 49(b). Although not shown on the figure, the results compare well with experiment.

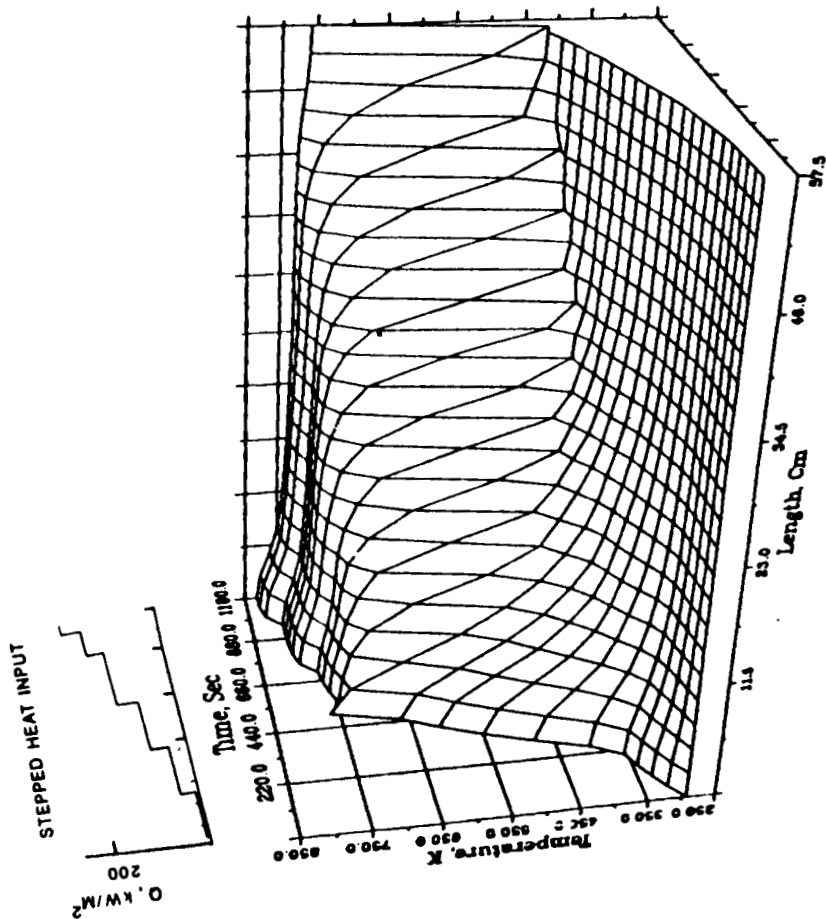
**Significance** - The present capability enables detailed analysis of nominal and off-design heat pipe performance and allows confidence in the development of heat-pipe solutions for highly-heated aerospace structures. Future vehicles such as NASP demand the use of efficient and reliable methods for alleviating local thermal loads such as heat pipes and confidence, developed through both analytical and experimental verification, is essential for their use.

**Future Plans** - Enhancements to the finite-element code are being made to enable the prediction of limiting heat pipe conditions and a user-friendly version of the code will be sent to Cosmic. Further, experimental verification of the code will take place in the spring of 1988, when a laboratory-type radiant heat test of an internally-instrumented, six-foot long sodium filled heat pipe will be tested and compared with analysis.

Figure 49(a).



# DEVELOPED FINITE-ELEMENT ANALYSIS CAPABILITY FOR TRANSIENT HEAT PIPE OPERATION



TRANSIENT HEAT-PIPE STARTUP  
( TEMP. HIST./PROFILE )

## MOTIVATION

- HEAT PIPES PROPOSED FOR NASP L.E.'S AND NOSE CAPS
- LIMITED ANALYSIS CAPABILITY

## SOLUTION

- DEVELOP F.E. SOLUTION TO COUPLED PROBLEM (THERMAL/FLUID)
- INCLUDES:
- 2-D COMP. VAP. FLOW
  - FREEZING/THAWING LIQUID
  - STARTUP FROM FROZEN STATE
- PREDICTS TEMPS. AND PRESSURES

## STATUS

- RESULTS COMPARE WELL WITH EXPERIMENT

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## IMPROVED INTRACELL BUCKLING COEFFICIENT LEADS TO WEIGHT REDUCTION FOR HONEYCOMB PANELS

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RTOP 505-63-31

**Research Objective** - Honeycomb panels play an important role in aircraft structures. For areas with low loading indexes, the compressive load-carrying capability of minimum gage face sheets becomes an issue. This situation arises when high strength materials are used by the structural designer to satisfy strength criteria at the cell level shown in the upper left of figure 50(b). This local buckling phenomenon is called intracell buckling. Existing work indicates that the commonly used design equations for intracell buckling are very conservative for low ratios of face sheet thickness to cell size ( $t/s$ ). This objective is to find realistic prediction methods for intracell buckling at low  $t/s$  ratios.

**Approach** - Finite element models of honeycomb panels were developed which included the face sheets and honeycomb core. Buckling loads of the structure were then determined for various ratios of face sheet thickness to cell size. This approach allowed calculation of the intracell buckling coefficient based upon critical load and known geometry, fixity, and material constants.

**Accomplishment Description** - Results indicated that the buckling coefficient is not constant for varying  $t/s$  ratios as is assumed by the design equations. Rather, the results showed an obvious relationship between buckling coefficient and the relative bending stiffness between face sheet and core structure. As the bending stiffness of the face sheet decreases with smaller thickness, the amount of edge restraint provided by the core increases. Therefore, the buckling coefficient  $K$  is increasing which results in higher critical loads. The same trend of increasing  $K$  with decreasing  $t/s$  ratio is observed in the experimental data shown in the upper right of the figure for  $t/s$  less than 0.04.

**Significance** - Current intracell buckling design equations, such as the one shown from MIL-HDBK-23A, ignore the interaction between face sheet and core when predicting buckling loads. The analytical results of the lower part of the figure show that reductions of up to 30 percent on face sheet weight are possible when using buckling coefficients based on relative stiffness between the face sheet and the core.

**Future Plans** - The majority of experimental work which is the basis for design equations was carried out in the early 1950's. Additional tests are needed on modern honeycomb materials to validate a design approach for low  $t/s$  ratios.

Figure 50(a).

# IMPROVED INTRACELL BUCKLING COEFFICIENT LEADS TO WEIGHT REDUCTION FOR HONEYCOMB PANELS

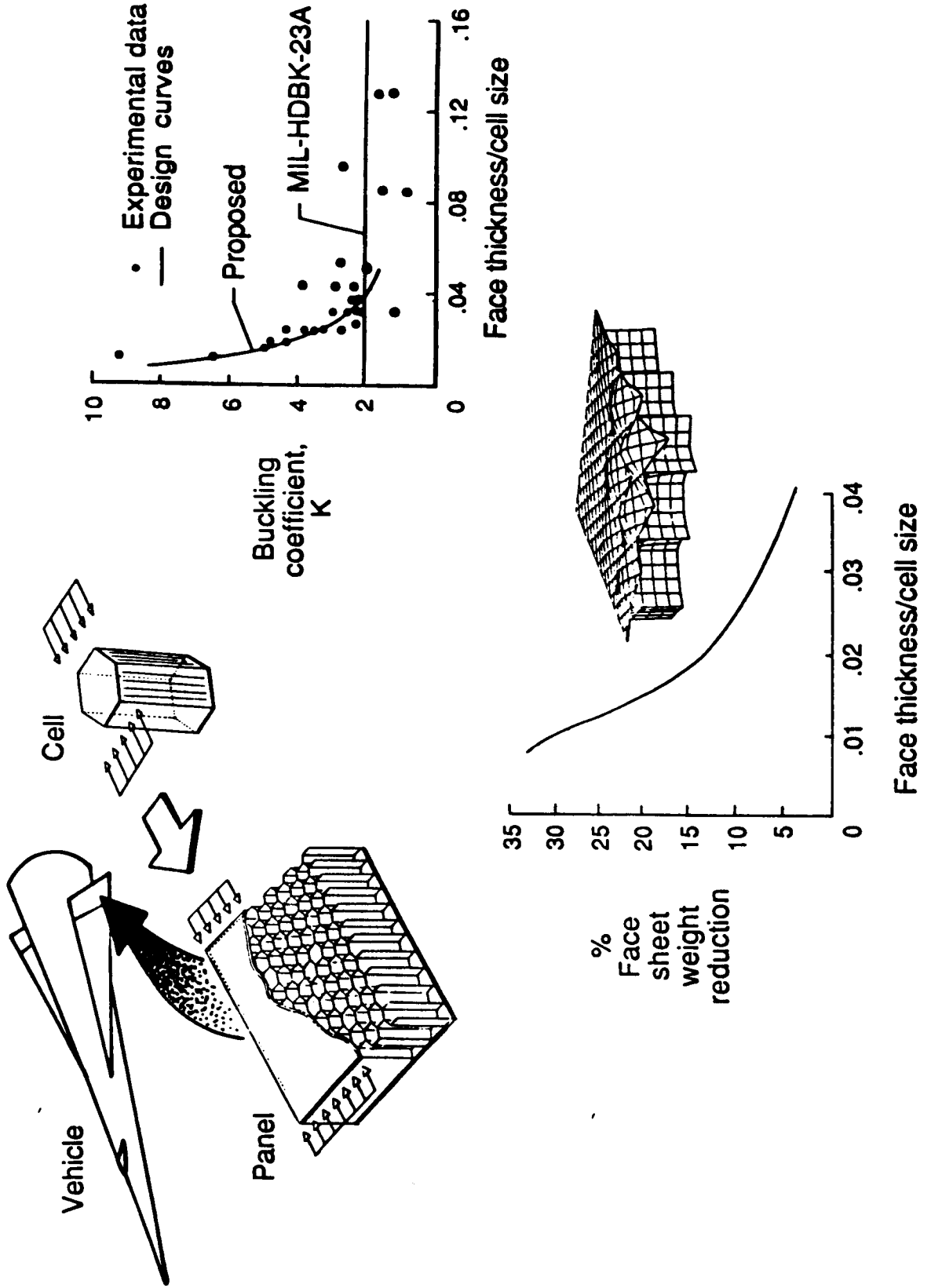


Figure 50(b).

# NEW DIRECT SOLUTION DEVELOPED FOR THERMAL STRESSES IN A SPHERICAL NOSE CAP UNDER AN ARBITRARY TEMPERATURE DISTRIBUTION

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RTOP 506-43-31

**Research Objective** - The process to design a nose cap for a hypersonic vehicle begins with a trajectory analysis followed by a stagnation point heating rate analysis. The heating rate distribution is used in a transient thermal analysis to find the time-dependent temperatures throughout the structure. Thermally induced stresses are computed for the worst case temperature profile and used to size the structure. Since the design process as outlined is iterated to arrive at a final design, it becomes essential to use a rapid, easy to use and accurate stress determination technique. Therefore, the research objective is to develop a direct solution to the thermal stress equations for the nose cap of a hypersonic vehicle that would provide just such a technique.

**Approach** - The nose cap structure is treated as a thin spherical shell with an axisymmetric temperature distribution. By expressing the stress solution to the thermoelastic equations as a series of Legendre polynomials, the governing differential equations are solved. The process of finding the coefficients for the series solution in terms of the temperature distribution, by using the orthogonality property of the Legendre polynomials, may be generalized when the temperature along the shell and through the thickness are expressed as polynomials. Under this assumption, the orthogonality condition leads to a sequence of integrals involving powers of the spherical shell coordinate and the Legendre polynomials. The coefficients of the temperature polynomial appear outside of these integrals and thus the integrals are evaluated once and for all and their values tabulated for use with any arbitrary polynomial temperature distribution.

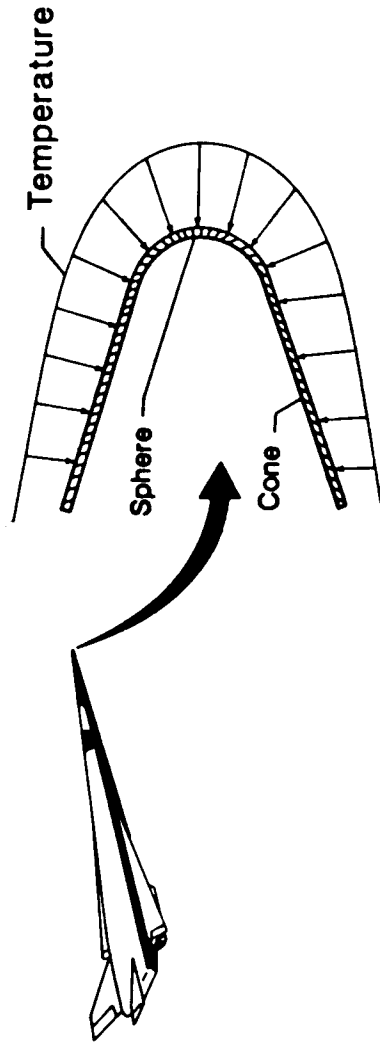
**Accomplishment Description** - A solution technique has been developed that solves the thermal stress equations directly. It has been programmed on a personal computer and receives its input interactively. The program is user friendly such that the user need not be concerned with the solution technique. Other than specifying the temperature as a polynomial, the user's only concern is computing stresses and plotting results of parameter studies for any material and spherical shell geometries specified. The results shown figure 51(b) compare the direct solution technique with results from a detailed finite element analysis. The stresses are representative of those in a ceramic nose cap with a typical temperature distribution.

**Significance** - The solution technique significantly reduces the iteration time to arrive at an acceptable design. Consequently, stress determination is no longer the most time consuming of the design steps by a considerable margin. The more efficient stress computations it provides could be readily included as a routine in an overall design synthesis program.

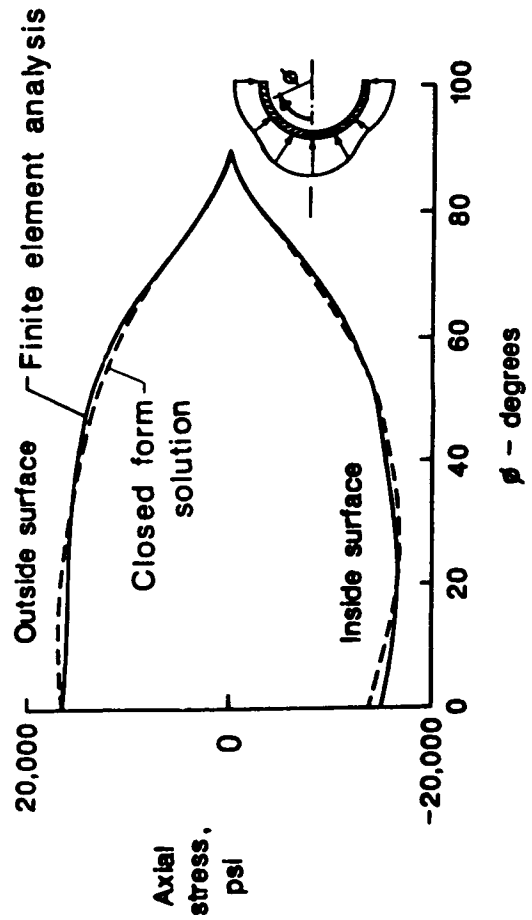
**Future Plans** - The solution technique for spherical shells will be extended to include conical shells. In addition, influence coefficient methods will be incorporated to allow combined shell geometries representing complex structural configurations.

Figure 51(a).

# NEW DIRECT SOLUTION DEVELOPED FOR THERMAL STRESSES IN A SPHERICAL NOSE CAP UNDER AN ARBITRARY TEMPERATURE DISTRIBUTION



## AXIAL STRESS DISTRIBUTION



- Closed form solution using Legendre polynomials
- Programmable on a desk top computer; user friendly
- Accurate, fast and efficient

Figure 51(b).

# CONFIGURATION AEROELASTICITY

## FY88 PLANS

- COMPLETE FLUTTER CLEARANCE STUDY OF F-16 WITH MODIFIED WING AND NEW PYLON
- COMPLETE PARAMETRIC INVESTIGATION OF ARROW WING FLUTTER CHARACTERISTICS
- COMPLETE TDT STUDY OF TAPERED ROTOR BLADES WITH ADVANCED AIRFOIL SECTIONS
- COMPLETE PARAMETRIC TRACK AND BALANCE STUDY FOR FORWARD FLIGHT CONDITIONS USING GENERIC ROTOR BLADES
- COMPLETE AH-1G DIFFICULT COMPONENTS STUDY
- COMPLETE INVESTIGATION OF TENSION-TORSION COUPLING ON DYNAMICS OF ROTATING CIRCULAR TUBES
- NEW DAS FOR TDT OPERATIONAL FOR ON-LINE USE

## AIRCRAFT AEROELASTICITY

Maynard C. Sandford  
Configuration Aeroelasticity Branch  
Extension 2661

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 are illustrated in figure 53(b). This research is a combination of experimental and complementary and analytical studies. The experimental work focuses 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. Flutter clearance studies in support of the development of the advanced technology fighter (ATF) will be initiated. In the long term this work will require several test series in the TDT using two different designs. A buffet response study of a bulbous nose launch vehicle configuration is also scheduled for the TDT. Concern with launch vehicle buffet has been rekindled by the renewed interest in the use of an unmanned launch vehicle to place large mass payloads in orbit. A combined analytical and experimental study of the flutter characteristics of a generic arrow-wing configuration will be completed. In this study the effects on flutter of a variety of parametric changes in geometry, stiffness, and mass are being investigated.

Figure 53(a).

NASA  
L-83-9,853

# AIRCRAFT AEROELASTICITY

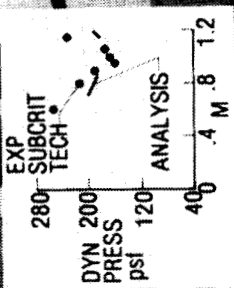
## CLEARANCE STUDIES



## RESEARCH AREAS

- FLUTTER
- DIVERGENCE
- ACTIVE/PASSIVE CONTROLS
- GUST RESPONSE
- AEROELASTIC TAILORING
- TEST TECHNIQUES

## CONFIGURATION STUDIES



## BASIC STUDIES

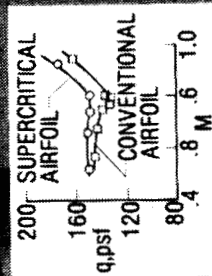
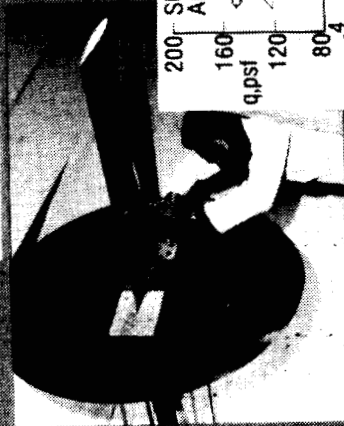


Figure 53(b).



# UPGRADING THE DATA ACQUISITION SYSTEM FOR THE LANGLEY TRANSONIC DYNAMICS TUNNEL

Bryce M. Kepley  
Configuration Aeroelasticity Branch  
Extension 2661

RTOP 505-63-21

**Research Objective** - The objective is to increase the productivity of the Langley Transonic Dynamics Tunnel (TDT) by replacing the existing computer-controlled data acquisition, display and control system with a new system that takes advantage of technological advances that have been made since the present system was designed over a decade ago. The new system will provide increased reliability, more flexibility, more data channels, faster data rates, and enhanced real time analysis as compared to the present system.

**Approach** - The conceptual design of the new system was developed by a Langley in-house team. The system consists of three central processor units which interface with an analog front-end that can accommodate 192 data channels. The three computer configuration provides the flexibility needed to perform multiple tasks. The configurations of the analog front-end provides for support of multiple tests. In addition to supporting tests in the TDT and its two calibration laboratories, the new system is configured to support tests in the General Rotor Aeroelastic Laboratory (GRAL), a helicopter hover test facility located in a building adjacent to the TDT. This facility is not supported by the present system.

**Status/Plans** - All major hardware components of the new system have been acquired and installed. A portion of this equipment is shown in figure 54(b). The design of the required operational software has been completed and most of the software written. The system is expected to be ready for on-line use in support of wind-tunnel testing in early calendar year 1988.

Figure 54(a).

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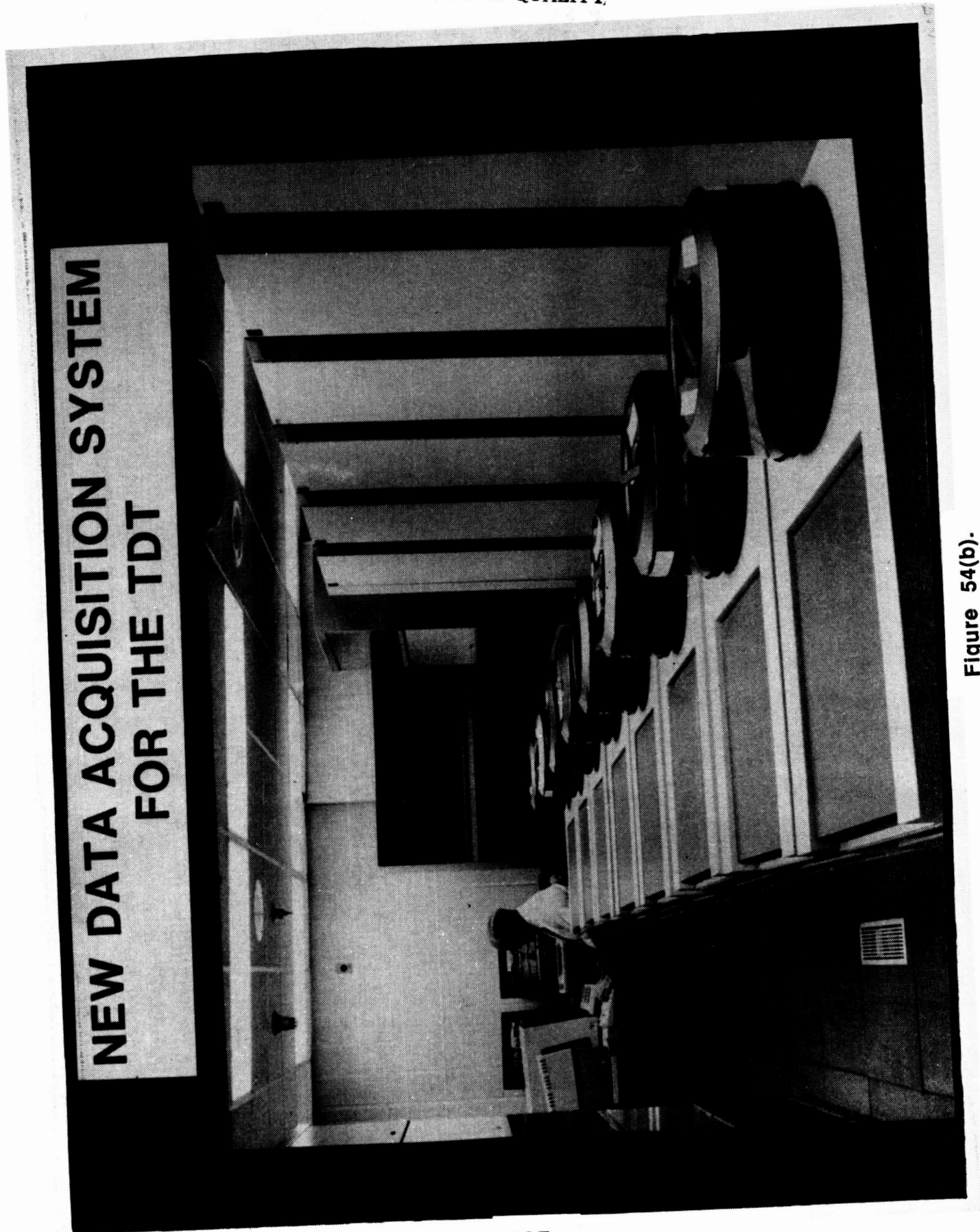


Figure 54(b).

## ROTORCRAFT DYNAMICS AND AEROELASTICITY

William T. Yeager, Jr.  
Configuration Aeroelasticity Branch  
Extension 2661

RTOP 505-63-51

**Research Objective** - The objectives as illustrated in figure 55(b) are (1) to conduct research in the aeroelastic, aerodynamic, and dynamic characteristics of rotors; (2) to support design of advanced performance rotorcraft in the areas of loads, vibration, and aeroelastic stability; and (3) to 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 LAD and the U. S. Army Aerostructures Directorate which is co-located at Langley. The work is a combination of experimental studies conducted 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. The in-house civil service research is supported and supplemented by industry contracts and university grants.

**Status/Plans** - The development of an advanced ARES will be continued. ARES II will provide a new dimension to TDT testing by simulating rigid body degrees of freedom as well as incorporating such features as individual blade pitch control. A second generation hingeless rotor for use in advanced research studies is being designed and obtaining a fully parametric bearingless hub will be pursued. The development of mathematical modeling techniques to analyze these new hardware configurations will continue. A grant is being used to modify the CAMRAD code to use a transfer matrix method to analyze trim, performance, loads, noise, stability and handling qualities, aeroelastic stability, vibration and gust response of bearingless rotors. Forward flight track and balance data will be obtained during a study planned for the TDT. Work will continue on the aerodynamically and dynamically advanced multi-speed (ADAM) rotor concept in preparation for TDT tests scheduled early next year. A passive, structurally tailored system will be developed on an advanced UH-60 Black Hawk rotor blade, that has demonstrated aerodynamic superiority over a current design, to minimize large fixed-system vibratory loads which occurred while maintaining the performance improvements.

Figure 55(a).

# ROTORCRAFT DYNAMICS AND AEROELASTICITY

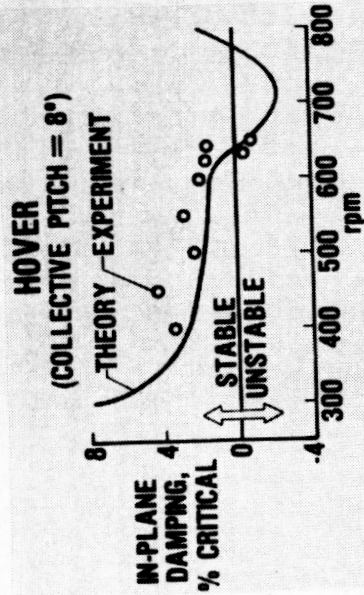
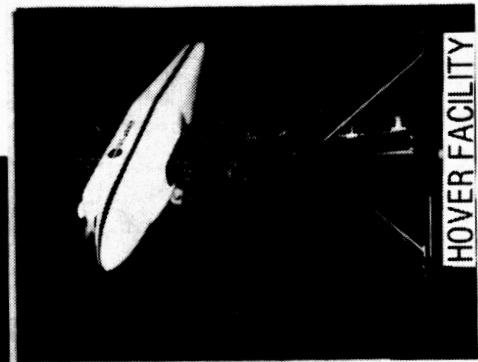
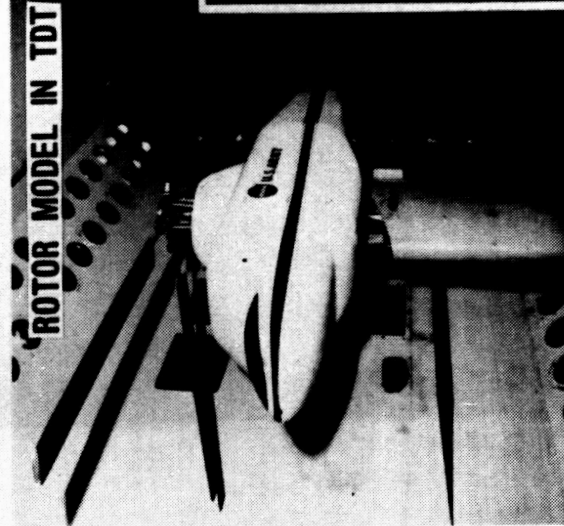
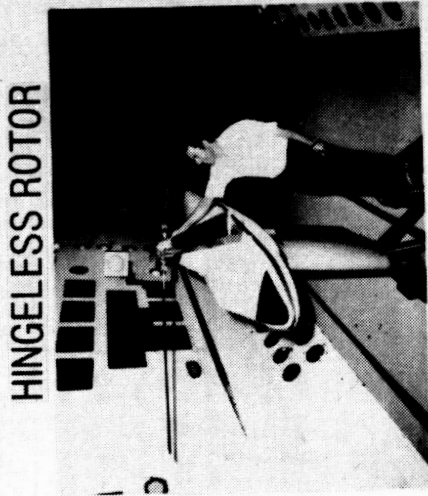
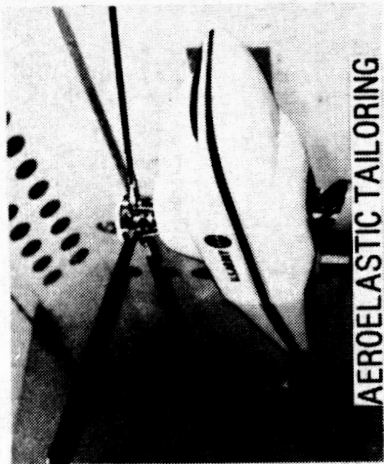
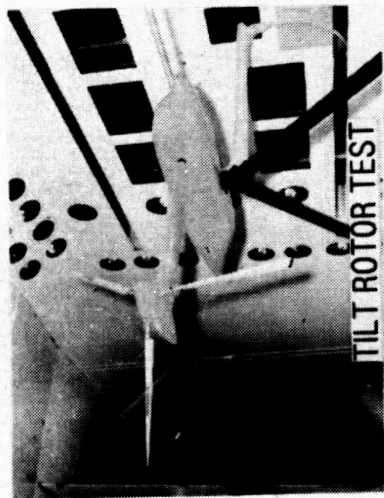


Figure 55(b).

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## A NATIONAL CAPABILITY TO ANALYZE VIBRATION AS PART OF HELICOPTER STRUCTURAL DESIGN

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Configuration Aeroelasticity Branch  
Extension 2661

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 in the development of new helicopters, vibration problems have not been identified and attacked until the flight test and operational stages. 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 replace in the United States a superior capability for design analysis of helicopter vibrations. The content of this effort is illustrated in figure 56(b).

**Approach/Status/Plans** - 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 Vertol, McDonnell Douglas Helicopter Co., and Sikorsky Aircraft Co. 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 analyses of a common vehicle. Because technology is a key ingredient in this program, annual meetings are held where the participants report on the status of their work. Perhaps a better grasp of the program can be obtained by mentioning a few of the program elements that are currently underway. At the next meeting, scheduled for early calendar year 1988, Bell, to cite just one industry participant, will be reporting on their work on a difficult components study wherein the effects of large mass components and secondary structure on airframe vibrations was assessed. In-house work in evaluating practical computational procedures for structural optimization of rotorcraft airframe structures subject to strength, frequency, and dynamic response constraints will be continued. Another significant in-house effort will focus on establishing a basic finite-element modeling methodology for composite rotor blades. In the university area, researchers at Rensselaer Polytechnic Institute are conducting a ground vibration test of the tail-boom structure of a Sikorsky S-55 helicopter and making comparisons with results from a companion finite-element analysis.

Figure 56(a).

# DESIGN ANALYSIS METHODS FOR VIBRATIONS PROGRAM (D A M V I B S)

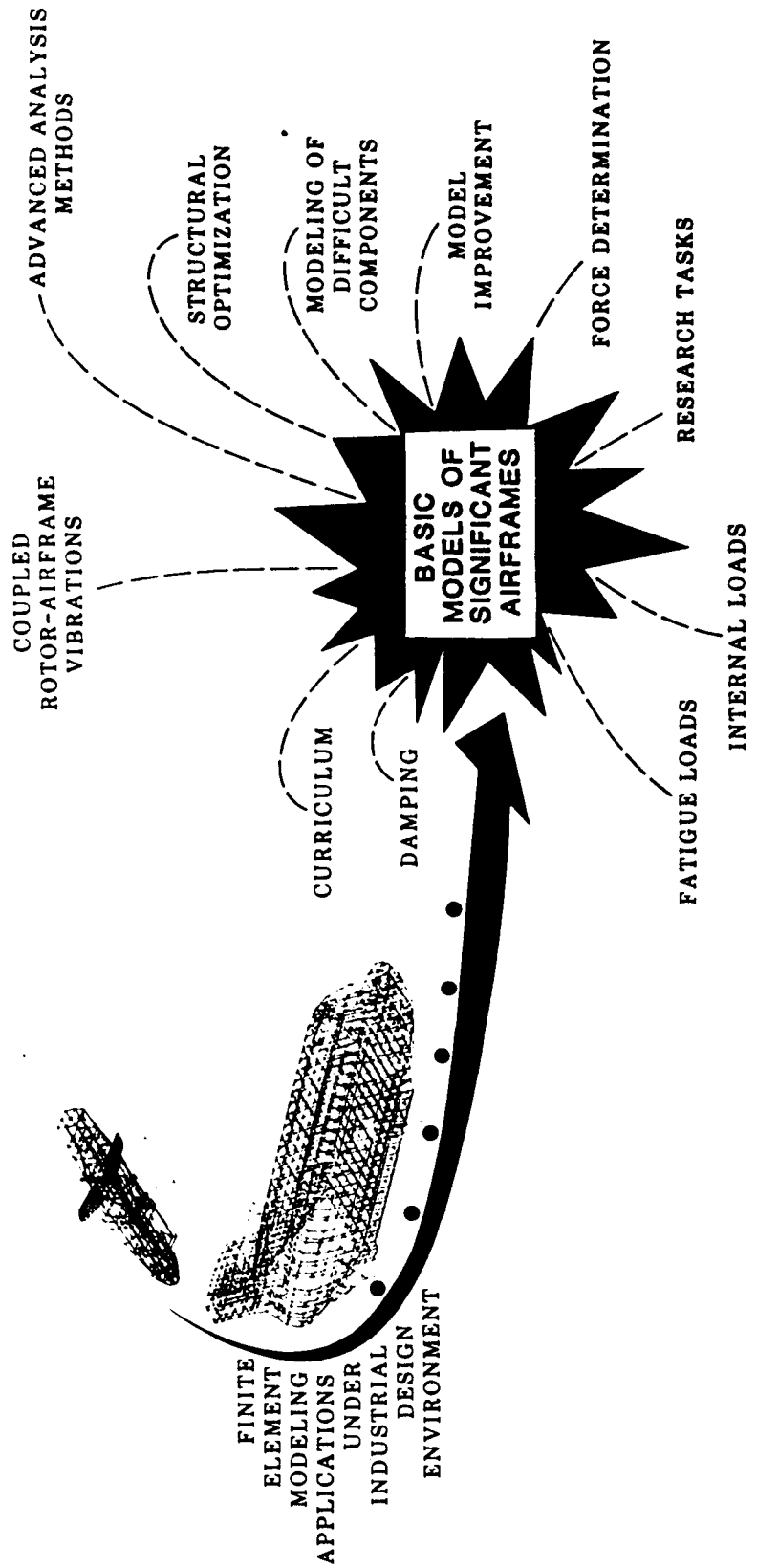
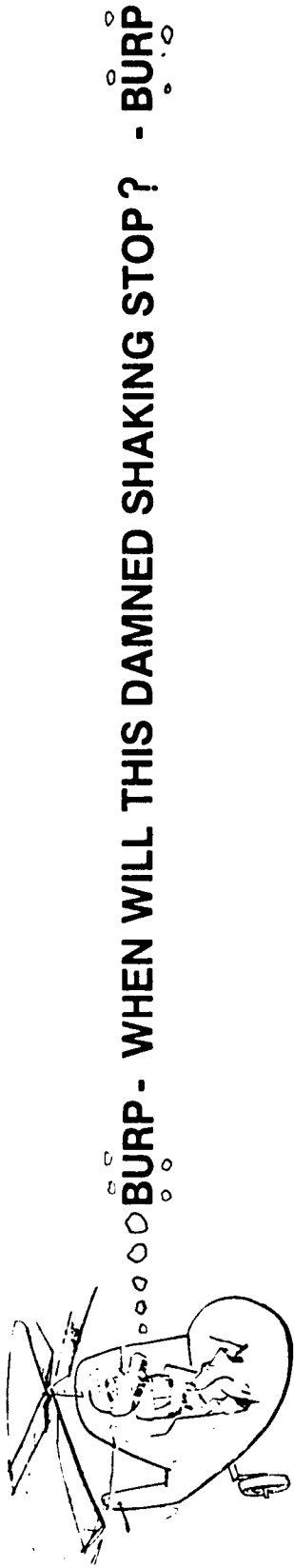


Figure 56(b).

# UNSTEADY AERODYNAMICS

## FY 88 PLANS

- VERIFICATION OF CAP-TSD FLIJTTER ANALYSIS FOR COMPLETE AIRCRAFT
  - COMPLETE TRANSFER OF TECHNOLOGY TO INDUSTRY
  - APPLICATIONS TO COMPLETE AIRCRAFT CONFIGURATIONS
  
- WING-ALONE FULL POTENTIAL CODE TO ASSESS GAP BETWEEN TSD AND THE EULER EQUATION
  
- EULER/NAVIER-STOKES COMPUTATIONAL AEROELASTICITY USING CFL3D AND CAP-TSD AEROELASTIC CAPABILITY
  
- INTERACTED VISCOUS BOUNDARY LAYER CAPABILITY
  - 2-D STRIP INTEGRAL METHOD
  - 3-D UNSTEADY FINITE DIFFERENCE METHOD
  
- VORTEX FLOW COMPUTATIONAL METHODS
  - INTEGRAL EQUATION/EULER EQUATION ZONAL METHOD
  - 3-D UNSTEADY, IMPLICIT EULER METHOD USING BODY FIXED GRID
  
- EXPERIMENTAL PROGRAM
  - CLIPPED DELTA CANARD-WING INTERFERENCE TEST
  - CANARD-ALONE HIGH-ALPHA DYNAMIC TEST

# AEROSERVOELASTICITY

## FY-88 PLANS

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- CONDUCT DYNAMIC LOADS ANALYSIS OF THE FIT STUDY AIRCRAFT
- COMPLETE CORRECTION FACTOR METHODS FOR UNSTEADY AERODYNAMICS
- DETERMINE SUITABILITY OF BRITISH PROPOSED SDG METHODOLOGY
- PREPARE MANUALS FOR ISAC DOCUMENTATION
- INVESTIGATE PROCEDURES TO INTEGRATE THERMAL EFFECTS INTO ASE ANALYSES
- COMPLETE MATHEMATICAL DEVELOPMENT OF OPTIMUM SENSITIVITY METHODS FOR DIRECT CONTROL LAW DESIGN
- DEVELOP METHODOLOGY TO DESIGN CONTROL LAWS BASED ON CONSTRAINED TIME HISTORY RESPONSES
- INITIATE COOPERATIVE ACTIVE CONTROLS TEST PROGRAM WITH ROCKWELL ON AFW MODEL
- COMPLETE ANALYSIS OF ARW-II DATA TO EVALUATE SHOCK INDUCED INSTABILITIES
- PERFORM ASE ANALYSIS OF STOPPED ROTOR X-WING AIRCRAFT



## DEVELOP METHODOLOGY TO DESIGN CONTROL LAWS BASED ON CONSTRAINED TIME HISTORY RESPONSES

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RTOP 505-63-21

**Research Objective** - In the existing control law synthesis methodologies, only the stochastic steady state responses are minimized. The transient time history responses are not treated directly as design constraints. The present objective is to develop this capability for a multi-input multi-output system, efficiently.

**Approach** - The transient responses due to a step command input and their gradients with respect to the control law parameters are obtained analytically and computed from the state transition matrices. The violation of the desired response bounds (fig. 59(b)) will be treated as a set of cumulative constraints and will be minimized using the method of feasible direction.

**Status/Plans** - The analytical gradient expressions have been derived. These will be validated against a pilot problem with a known analytical solution. The efficient way to handle these constraints and the feasibility of the procedure will then be determined. A large order problem with a known numerical solution will be used for validation. The procedure will be incorporated into the optimization overlay of the control law synthesis software PADLOCS in an interactive and batch mode.

# Develop Procedures to Design Control Laws with Constrained Time History Responses

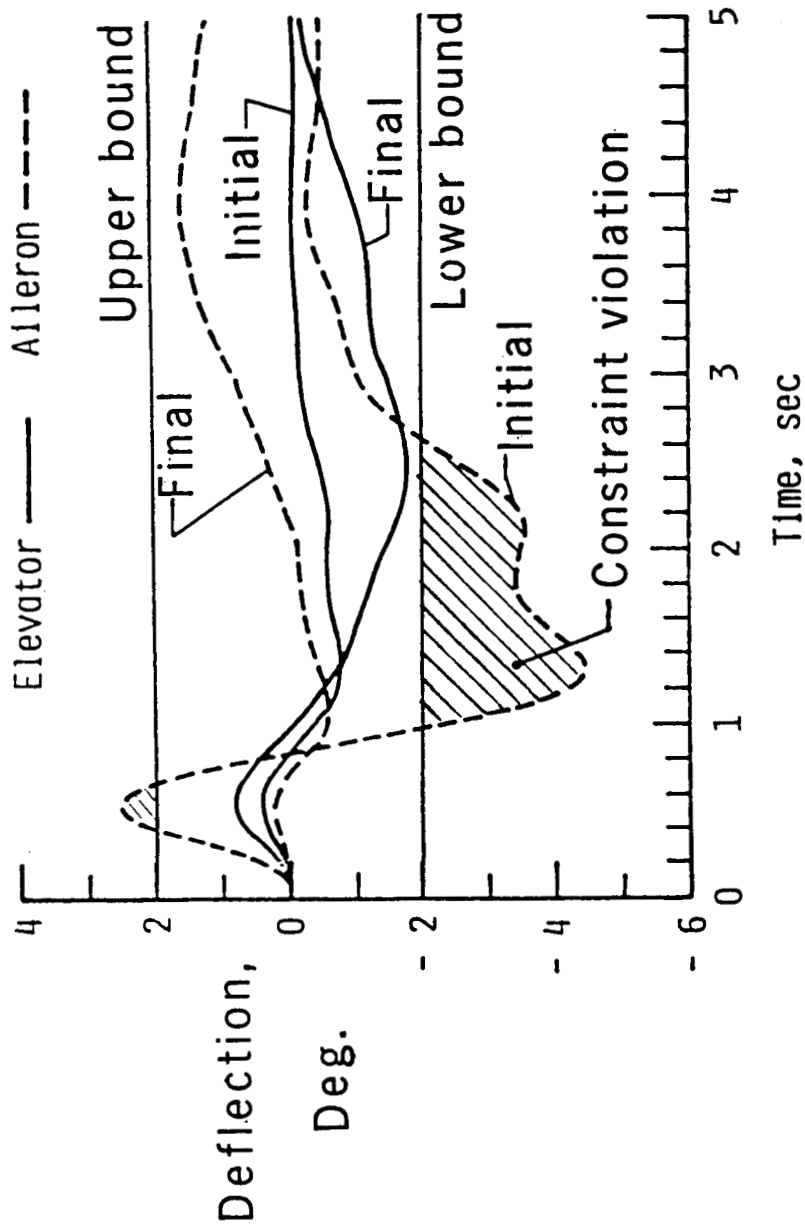


Figure 59(b).

# INITIATE COOPERATIVE ACTIVE CONTROLS TEST PROGRAM WITH ROCKWELL ON AFW MODEL

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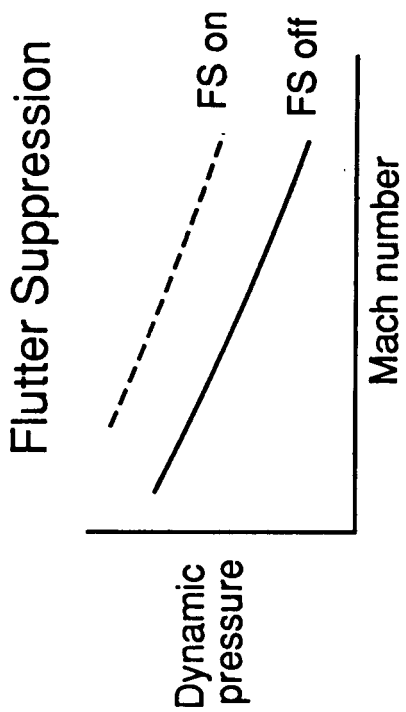
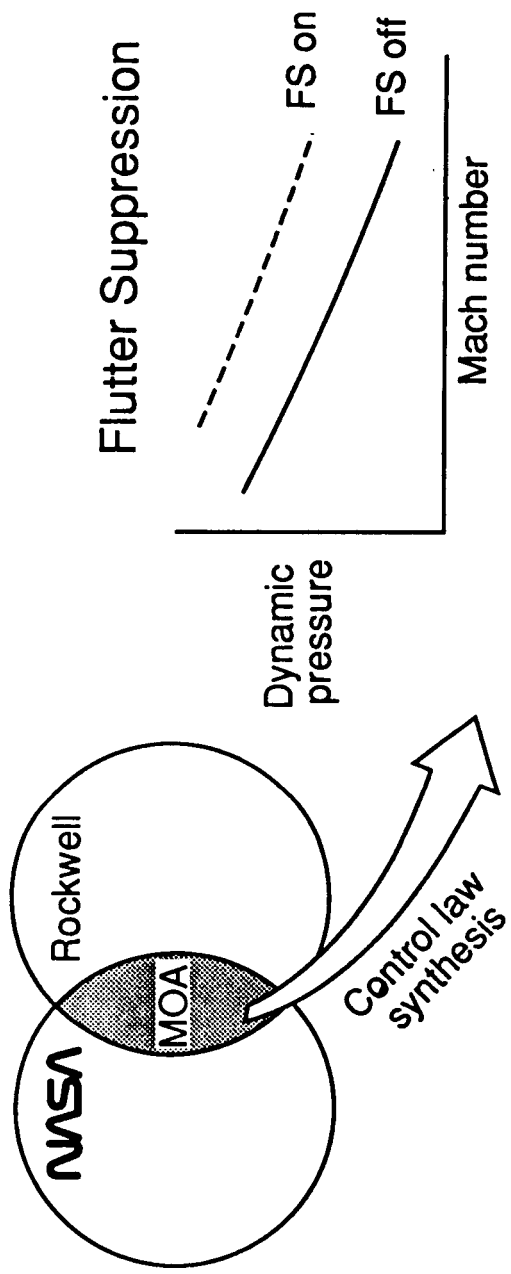
TOP 505-63-21

**Research Objective** - The evolution of advanced fighter technology has required that the disciplines of aerodynamics, control systems, and structures be integrated into a unified aeroservoelastic discipline. This new discipline must be evaluated by sophisticated analytical methods and validated through the testing of wind-tunnel models. NASA-LaRC and Rockwell International Corporation are concurrently developing advanced aeroservoelastic analysis and synthesis methods and, thus, have a mutual interest in cooperating in this effort. The objectives of this cooperative program are to synthesize and test advanced digital active control laws on a scaled aeroelastic wind tunnel model.

**Approach** - Rockwell has developed a concept referred to as the Active Flexible Wing (AFW) Concept. The concept utilizes wing flexibility and multiple active control surfaces to vary the wing shape, resulting in improved performance and reduced weight. A scaled aeroelastic model using the AFW Concept has been built and previously tested in the LaRC Transonic Dynamics Tunnel and will be tested again during this cooperative program. Through a Memorandum of Agreement, NASA-LaRC and Rockwell will jointly conduct this cooperative program. They will share methodology and related computer programs for the development of flutter-suppression and rolling-maneuver-load-alleviation control laws to control the aeroelastic responses of the AFW wind-tunnel model (fig. 60(b)).

**Status/Plans** - The Memorandum of Agreement was signed by NASA-LaRC and Rockwell in October of 1987 and has a term of three years, during which two wind-tunnel tests are planned.

# INITIATE COOPERATIVE ACTIVE CONTROLS TEST PROGRAM WITH ROCKWELL ON AFW MODEL



## Rolling Maneuver Load Alleviation

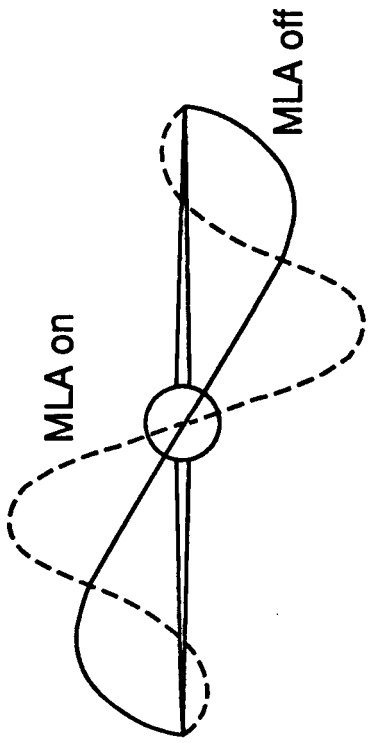


Figure 60(b).

## INTEGRATED AERO/THERMAL/ELASTIC ANALYSIS

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RTOP 505-63-21

**Research Objective** - The rigorous mission requirements imposed on advanced aerospace vehicles will drive the designs toward very efficient structures (i.e., reduced weight). The combination of reduced weight and efficient aerodynamic shape (i.e., low drag) for hypersonic flight has the potential to result in a relatively flexible structure. The flexible structure, high temperature, high Mach number and high dynamic pressure operating conditions provide for a severe aerothermoelastic (both static and dynamic) environment. The objective is to develop the analytical capability of determining aerothermoelastic characteristics and help resolve associated problems on hypersonic vehicles of national interest. The capability to predict quasi-steady state thermal and aerodynamic loads, deformations and required control settings for trim at selected flight conditions will enhance the design and analysis process.

**Approach** - The problem requires that the equilibrium between aerodynamic loading, thermal loading, gravity, inertial loading, deformed shape of the structure, and control settings be solved for at a given point and flight condition within a mission profile. Figure 61(b) depicts the current concept of the iterative method. The process involves predicting a rigid body trim condition, and updating deformations (due to aerodynamic, thermal, gravity and inertial loading) and control settings until equilibrium is achieved. State-of-the-art aerodynamic, structural and thermal analysis techniques will be assessed to determine current capabilities and weaknesses.

**Status/Plans** - Precise predictions of hypersonic pressures and heating and temperature distributions are impractical with present day methods. Subsonic and supersonic aerodynamics and elastic structural methods which analyze force and thermal loads are more precise, easier to implement, and are currently being used in an iterative aeroelastic trim method. Piston theory (hypersonic aerodynamics) has been implemented for approximating steady and unsteady aerodynamic forces. Advanced techniques will be integrated into the analysis system as they become available. Aeroelastic trim conditions for symmetric configurations at subsonic speeds have been determined using EAL and a modified version of a kernel function code by Atlee M. Cunningham, Jr., of General Dynamics. Maneuver conditions can include straight and level, steady climb/descent, pull-up/push-over and level, climbing or descending turns. Critical to the integrated aero/thermal/elastic analysis is aerodynamic & aerothermal codes for hypersonic flight being available.

Figure 61(a).

# INTEGRATED AERO/THERMAL/ELASTIC ANALYSIS

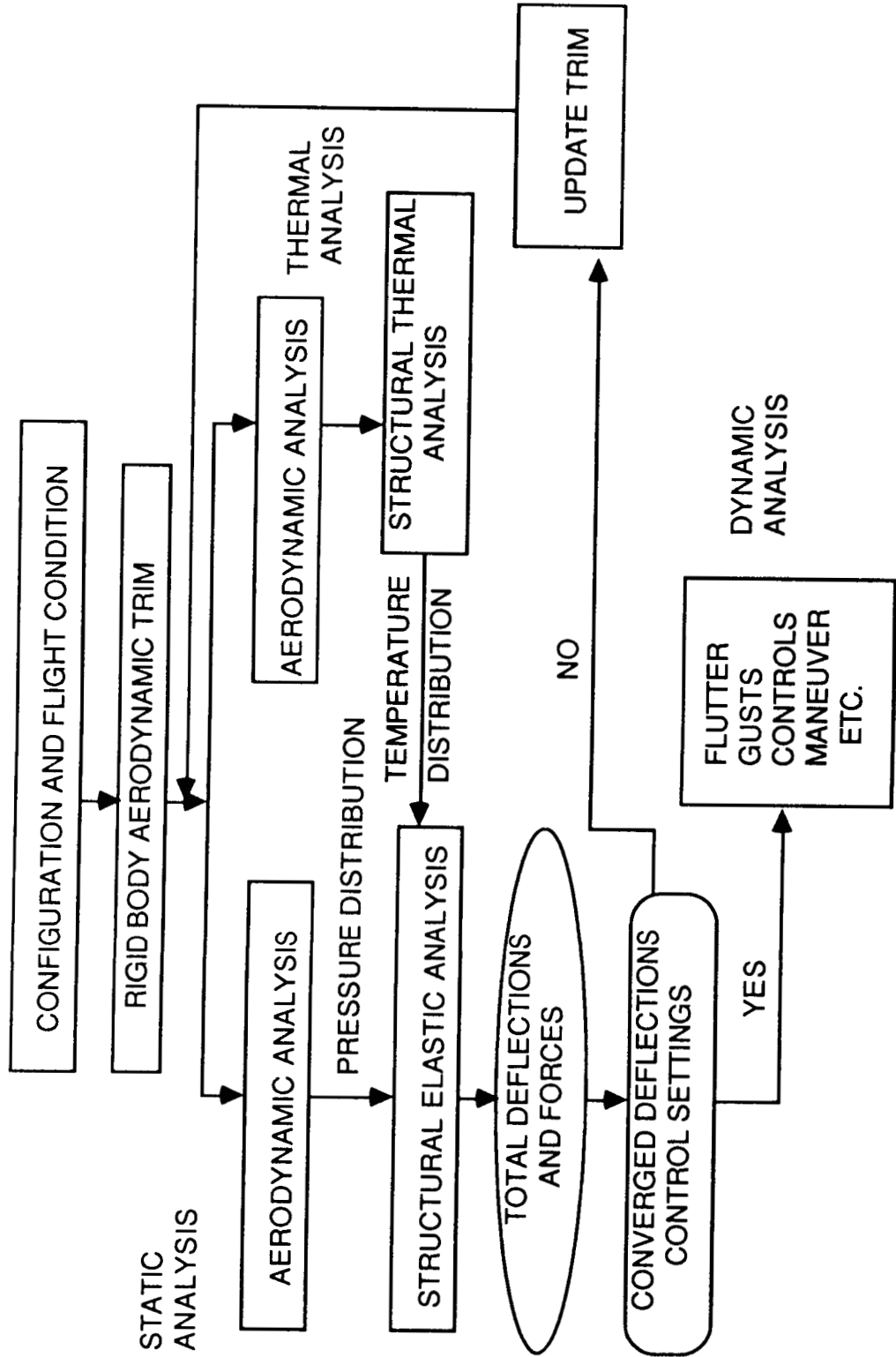


Figure 61(b).

# AEROTHERMAL LOADS

## FY 88 PLANS - EXPERIMENTAL

- o COMPLETE TESTING IN 8' HTT
  - CHINE GAP HEATING
  - 2ND GENERATION SHROUD INTEGRITY TESTS - AF/DNA/MDC
  - NASP BASELINE VEHICLE AERODYNAMICS - NASP/BOEING
  - TURBULENT BOUNDARY LAYER CHARACTERIZATION
- o COMPLETE TESTS IN CALSPAN HYPERSONIC SHOCK TUNNEL
  - SWEPT LEADING EDGE SHOCK-ON-LIP
  - MULTIPLE SHOCK-ON-LIP
- o UPGRADE Re & MACH NO. CAPABILITY OF CALSPAN 48" HST
- o DOCUMENT RESULTS
  - MACH 11-19 SHOCK-ON-LIP
  - MACH 8 SWEPT LEADING EDGE
  - CHINE GAP HEATING
  - MULTIPLE SHOCK-ON-LIP
- o FABRICATE, INSTRUMENT, AND TEST
  - SLOT COOLED MODEL WITH SHOCK WAVE INTERACTION
  - AXIAL CORNER MODEL WITH IMPINGING SHOCK WAVE

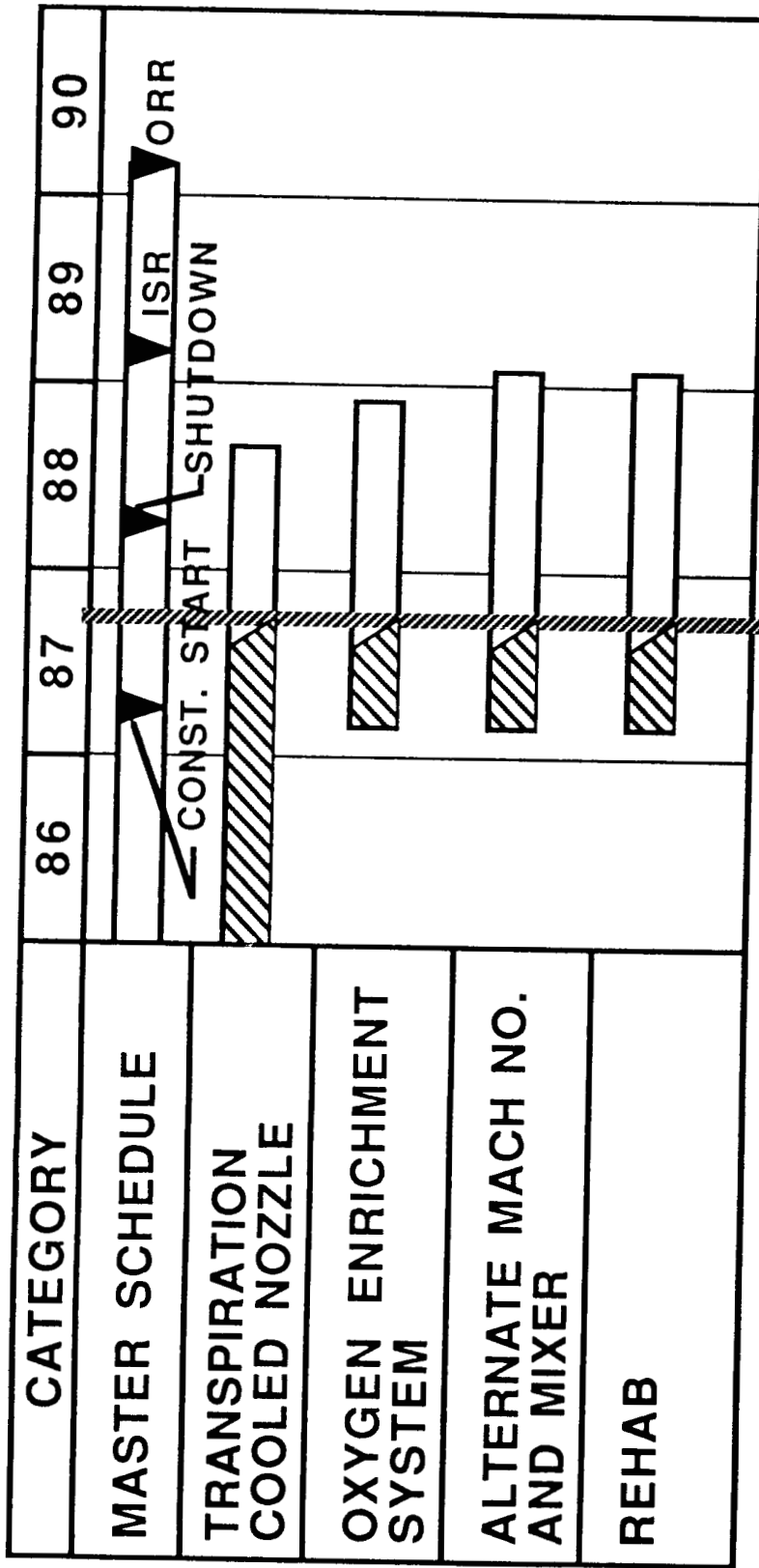
# AEROTHERMAL LOADS

## FY 88 PLANS - ANALYTICAL

- o VALIDATE ANALYSIS TOOLS WITH EXPERIMENTAL DATA
  - CYLINDRICAL LEADING EDGE
  - SHOCK-ON-LIP
  - COMPRESSION CORNER
  - HYPERSONIC VEHICLE (MACH 20) AND FIGHTER
- o DEVELOP IMPLICIT TIME MARCHING SCHEME
- o EXTEND TRIANGLE ADAPTIVE UNSTRUCTURED REMESHING SCHEME TO 3D
- o EXTEND ADAPTIVE UNSTRUCTURED GRID ENHANCEMENT TO THERMAL AND STRUCTURAL ANALYSIS
- o EVALUATE HIGHER ORDER ELEMENTS FOR CFD
- o INITIATE IMPLEMENTATION OF CONSTITUTIVE RELATIONSHIPS/ALGORITHMS FOR NONLINEAR STRUCTURAL BEHAVIOR



# 8'HTT MODIFICATION PROJECT SCHEDULE



KEY

PLANNED

ACTUAL

Figure 64.

## INCIDENT SHOCK / WALL JET INTERACTION

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RTOP 506-43-31

**Research Objective** - Investigate the effect of an incident oblique shock on the cooling effectiveness of a wall jet cooling layer.

**Approach** - Develop an experimental model with supersonic jets for mass injection parallel to the wall surface (fig. 65(b)). Characterize the cooling effectiveness of the slot, without an incident shock, for several different jet heights, free-stream to jet velocity ratios, and free stream to jet pressure ratios. Repeat the tests with an incident oblique shock of varying strengths interacting with the cooling layer downstream of the jet. Densely packed wall heat transfer rate and pressure instrumentation in the shock interaction region will resolve the highly localized phenomena. Tests will be conducted in both the LaRC 8-foot High Temperature Tunnel (8' HTT) and the Calspan 48-inch Hypersonic Shock Tunnel to provide a range of free stream Mach Numbers and Reynolds Numbers.

**Status/Plans** - The preliminary model for the 8' HTT tests has been completed and the final design is underway. Delivery of the completed model is scheduled for March 1988. Design and fabrication of the Calspan model is being accelerated by using existing model parts from earlier shock/boundary layer studies. The Calspan tests can begin early in 1988.

Figure 65(a).

# INCIDENT SHOCK / WALL-JET INTERACTION

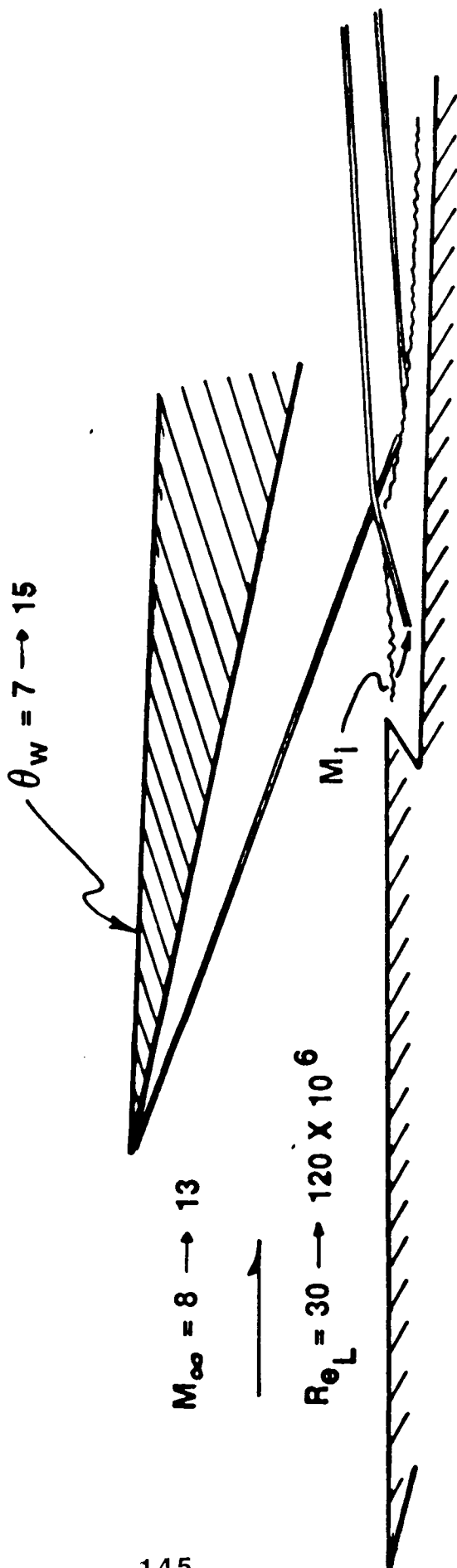


Figure 65(b).

## CORNER FLOW EXPERIMENT

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RTOP 506-43-31

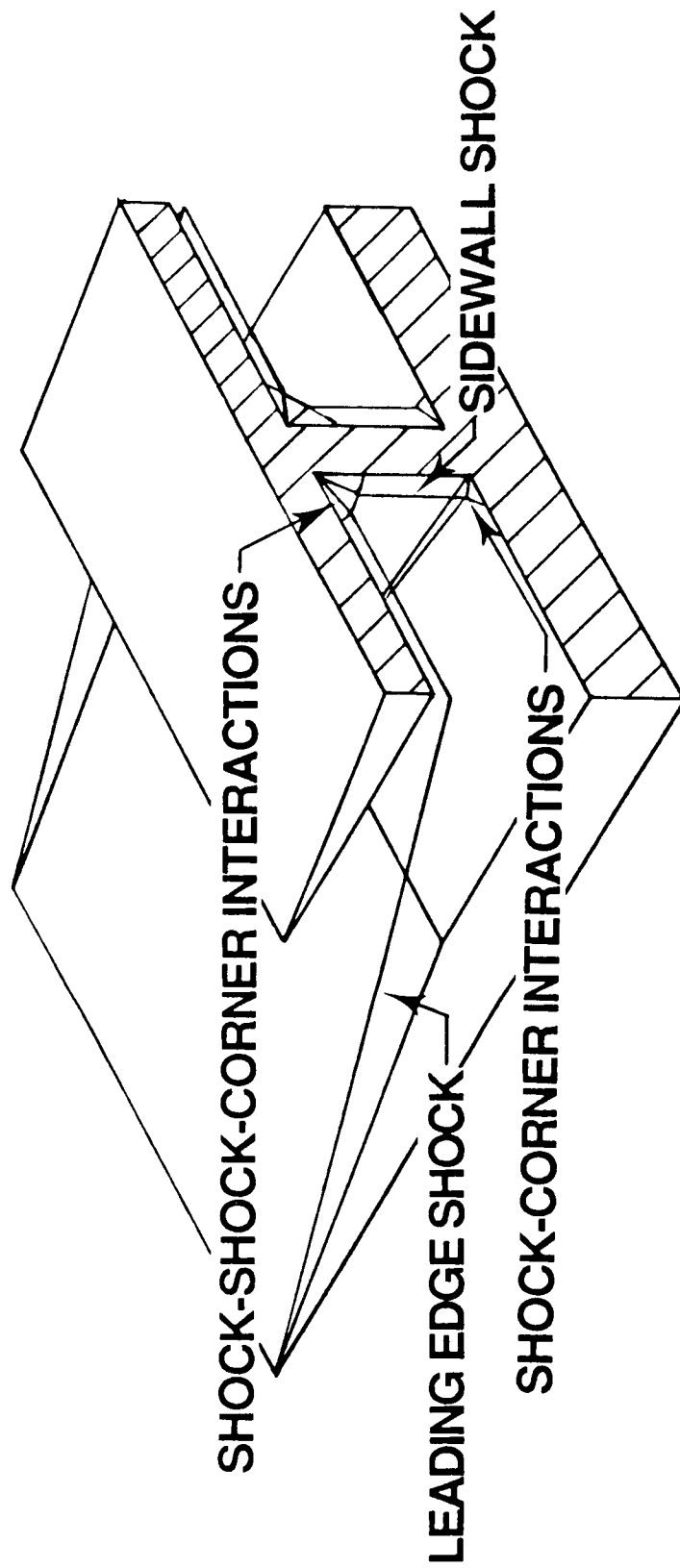
**Research Objective** - The objective of this research is to develop a data base on detailed aerothermal loads for a variety of corner geometries typical of those found on hypersonic vehicles. The heat transfer rate and pressure load data base will be utilized in the design of structures for future hypersonic vehicles. In addition, the data base will be useful in the validation of analytical codes used to predict loads for flight conditions that can not be simulated in ground based facilities.

**Approach** - This research will be conducted in two phases. In the first phase, a large number of relatively inexpensive two sided and three sided corner geometries (fig. 66(b)) will be constructed and tested in the Langley 20-Inch M-6 Tunnel. A phase change paint technique will be used to obtain both qualitative and quantitative heat transfer rates and distributions in the corners. Included in these geometries will be both simple corners representative of, for instance, a fin - fuselage corner as well as more complex corners with a shock impinging in the corner representative of, for instance, an inlet corner with the cowl bow shock wave intersecting the corner. In the second phase of the program, the information obtained in the first phase will serve as a guide for instrumentation layout and to identify critical corner geometries which produce intense localized heat transfer rates and pressure distributions. Highly instrumented models of the selected corner configurations will be constructed and tested in the Langley 8' HTT and the Calspan shock tunnels.

**Status/Plans** - The models for the first phase of the investigation are under construction. Tests in the 20-Inch M-6 Tunnel are scheduled for mid 1988.

Figure 66(a).

**CORNER FLOW EXPERIMENT  
DESIGN AND CODE VALIDATION DATA BASE**



**THREE-SIDED CORNER MODEL**

Figure 66(b).

# THERMAL STRUCTURES

## FY88 Plans

### Cooled Structures

- Test SCRAMJET struts in lab and combustion facility
- Develop actively cooled test apparatus at NBS; complete initial testing
- Develop very high temperature heat pipe
- Test water glycol cooled panels in ACTS

### Hot/Cryogenic Structures

- Develop carbon/carbon structure for control surfaces
- Test curved panel array in 8' HTT to resolve gap heating issues
- Develop methodology for cryo tank certification

### Analysis & Synthesis

- Investigate levels of modeling thermal/structural detail
- Determine thermal effects on structural stiffness
- Investigate new model-size reduction technique for thermal problems
- Extend transform-method-based finite elements to nonlinear problems
- Develop optimal radiation surface elements

Continue development of Thermal Structures Laboratory

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# Report Documentation Page

1. Report No. NASA TM-100534		2. Government Accession No.		3. Recipient's Catalog No.	
4. Title and Subtitle LOADS AND AEROELASTICITY DIVISION RESEARCH AND TECHNOLOGY ACCOMPLISHMENTS FOR FY 1987 AND PLANS FOR FY 1988				5. Report Date January 1988	
				6. Performing Organization Code	
7. Author(s) S. C. Dixon and James E. Gardner				8. Performing Organization Report No.	
				10. Work Unit No. 505-63-21-02	
9. Performing Organization Name and Address Langley Research Center Hampton, Virginia 23665-5225				11. Contract or Grant No.	
				13. Type of Report and Period Covered Technical Memorandum	
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, DC 20546				14. Sponsoring Agency Code	
15. Supplementary Notes					
16. Abstract <p>The purpose of this paper is to present the Loads and Aeroelasticity Division's research accomplishments for FY 87 and research plans for FY 88. The work under each Branch (technical area) will be 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.</p>					
17. Key Words (Suggested by Author(s)) Accomplishments, research plans, Loads and Aeroelasticity			18. Distribution Statement Unclassified - Unlimited Subject Category - 02		
19. Security Classif. (of this report) Unclassified		20. Security Classif. (of this page) Unclassified		21. No. of pages 150	22. Price A07