NASA-CR-201006

JOINT INSTITUTE FOR AERONAUTICS AND ACOUSTICS
STANFORD UNIVERSITY

AERO NO. 42-95

NASA GRANT NCC 2-55 #27

THE RESEARCH AND TRAINING ACTIVITIES FOR THE JOINT INSTITUTE FOR AERONAUTICS AND ACOUSTICS

Submitted to the

NASA Ames Research Center
Moffett Field, CA 94035

For a period of One Year
Commencing October 1, 1995

by the

Department of Aeronautics and Astronautics
Stanford University
Stanford, California 94305

Principal Investigator

Professor Brian Cantwell

July 1995

JOINT INSTITUTE FOR AERONAUTICS AND ACOUSTICS
STANFORD UNIVERSITY

AERO NO. 42-95

NASA GRANT NCC 2-55

THE RESEARCH AND TRAINING ACTIVITIES FOR THE JOINT INSTITUTE FOR AERONAUTICS AND ACOUSTICS

Submitted to the

NASA Ames Research Center
Moffett Field, CA 94035

For a period of One Year
Commencing October 1, 1995

by the

Department of Aeronautics and Astronautics
Stanford University
Stanford, California 94305

Principal Investigator

Professor Brian Cantwell

July 1995

TABLE OF CONTENTS

ABSTRACT 3

I. JOINT INSTITUTE PROGRAM OVERVIEW 4

A. Introduction 4

B. Research Project Summaries 4

C. Institutional Support 6

D. Training Activities 6

E. Research Participation 7

II. DETAILED PROJECT DESCRIPTIONS 8

Project 1. Active flow control by tangential forebody blowing 8

Project 2. Large eddy simulation as a tool for studying jet aero-acoustics . 23

Project 3. Research on a lifting wing-flap combination with application to airframe noise generation 28

Project 4. Application of luminescent paint sensors to fluid physics problems 35

Project 5. Prediction of wing maximum lift for preliminary design methods 38

III. PERSONNEL 46

IV. FUNDING 46

ABSTRACT

This proposal requests continued support for the program of activities to be undertaken by the Ames-Stanford Joint Institute for Aeronautics and Acoustics during the period October 1, 1995 to September 30, 1996. The emphasis in this program is on training and research in experimental and computational methods with application to aerodynamics, acoustics and the important interactions between them. The program comprises activities in active flow control, Large Eddy Simulation of jet noise, flap aerodynamics and acoustics and high lift modeling studies. During the proposed period there will be a continued emphasis on the interaction between NASA Ames, Stanford University and Industry, particularly in connection with the high lift activities.

The program will be conducted within the general framework of the Memorandum of Understanding (1976) establishing the Institute, as updated in 1993. As outlined in the agreement, the purposes of the institute include the following:

- To conduct basic and applied research.
- To promote joint endeavors between Center scientists and those in the academic community.
- To provide training to graduate students in specialized areas of aeronautics and acoustics while participating in the research programs of the Institute.
- To provide opportunities for Post-Doctoral Fellows to collaborate in research programs of the Institute.
- To disseminate information about important aeronautical topics and to enable scientists and engineers of the Center to stay abreast of new advances through symposia, seminars and publications.

The program described above is designed to address future needs of NASA Ames and has been the basis of discussion among Profs. B. Cantwell, I. Kroo, S. Lele and S. Rock from the Stanford faculty and several members from NASA Ames including Dr. R. Moser, Dr. S. Davis, Dr. Dochan Kwak, Dr. S. Smith, Dr. J. Ross and Dr. L. Olsen. Coordination of this activity at Ames is the responsibility of the Institute Associate Director for Center Affairs, Dr. C. A. Smith.

I. JOINT INSTITUTE PROGRAM OVERVIEW

A. Introduction

Experimental and computational aerodynamics have for many years played an important role in the basic and applied research programs of Ames Research Center and in the research and training activities of Stanford University. Recently computational tools have been brought to bear on the difficult problem of flow generated noise. The coordinated use of a combination of experimental and computational tools has long been recognized as an essential part of a comprehensive approach to improving our fundamental understanding of complex flow phenomena. Developments in computational capabilities, in flow visualization, in measurement and in new kinds of wind-tunnel instrumentation will constitute a major step forward in the ability of scientists and engineers to advance the state of the art in aerodynamic design technology.

It is therefore the general character of the proposed program that it involves both experiment and computation and that these are used in complimentary ways. This approach can be undertaken only if highly qualified personnel and good research facilities are available. In this regard the blending of resources from Stanford and Ames is an important ingredient and was one of the motivating reasons behind the establishment of the Ames-Stanford Joint Institute.

In the experimental parts of the program described below, smaller scale investigations undertaken at Stanford are coordinated with both computations and experiments carried out in the more powerful facilities at Ames Research Center.

B. Research Project Summaries

The research directions summarized here, and further elaborated in the Program Description, are the result of several discussions with research management and staff at Ames Research Center. The activities are consistent with the emphasis on acoustics and high-lift in current NASA programs.

Project 1. Active flow control by tangential forebody blowing

This is a continuing program in the use of active control as a means of controlling aircraft at high angle of attack. The combined roll and yaw control of a generic aircraft with thin delta wing using forebody tangential blowing is being investigated. Techniques for developing nonlinear optimum control laws are being developed using experimentally derived results obtained on a unique free-to-roll, free-to-yaw support system which has been developed under this program. The natural behavior of this system consists of oscillations in roll and yaw. Wind tunnel data, plus numerical computation are being used to provide the aerodynamic information necessary for the formulation of control laws for this configuration.

Project 2. Large eddy simulation as a tool for studying jet aero-acoustics

New subsonic and supersonic aircraft are required to meet more stringent environmental noise regulations. Current design/analysis tools for estimating the noise generated by an aircraft configuration rely strongly on empirical data. With recent advances in computational technology it seems possible that important components of aircraft noise could be predicted by a first principles approach. Since noise is generated by unsteady flow it becomes necessary to accurately predict the unsteady flow. The proposed research seeks to evaluate and develop Large Eddy Simulation (LES) as a computational technology for predicting jet-noise.

Project 3. Study of a lifting wing-flap combination with application to airframe noise generation

The adoption of increasingly stringent international, national and local noise rules and the advent of large, high lift commercial aircraft has led to a renewed interest in noise generation by airframe components. Recent studies of airframe noise have identified noise sources associated with wing flap trailing edges and flap side edges. In this project the fluid dynamic processes associated with this type of noise source are being investigated. An NACA 63-215 Mod B airfoil section has been used by NASA Ames investigators for high Reynolds number studies in the Ames 7x10 tunnel. These experiments also included noise studies carried out by Boeing and Ames investigators working in collaboration using Boeing-developed phased array instrumentation. This same geometry is also being studied in CFD computations by Ames and Stanford investigators and in small scale experiments at Stanford University. The emphasis in the Stanford experiments is on visualization and measurement of unsteady aspects of the flow which can not be easily studied in either the computations or the 7x10 experiments. Fluorescent dye visualization will be carried out in a low speed water facility and unsteady pressure measurements on the flap and other sections of interest along with selected wake measurements will be taken in the Stanford low speed wind tunnel.

Project 4. Application of luminescent paint sensors to fluid physics problems

The use of a luminescent (pressure sensitive) paint to measure the spatial pressure distribution on a wind tunnel model is being studied. These paints are based on a class of chemicals known as porforins and make use of a surface reaction which, under illumination with ultraviolet light, causes the scattered light intensity to be proportional to the partial pressure of oxygen at the painted surface. This can be used to infer surface pressure over an extended area using a video camera and associated image processing system. With further development, these paints as well as similar systems capable of measuring wall shear stress, promise to revolutionize wind tunnel testing techniques. In particular the high cost of pressure instrumentation for wind tunnel models can be greatly reduced both because the need for pressure taps is greatly reduced and also because the cost of illumination can be amortized over a large number of experiments. The surface reaction is fast and in principle it should be possible to measure the time varying pressure. Initial, proof-of-concept, experiments are under way at Stanford using a jet impinging on a plate to examine the time dependent response of the paint.

Project 5 Prediction of wing maximum lift for preliminary design methods

The high lift characteristics of wings have important effects on aircraft noise, cost and performance. The goal of this project is to improve our understanding of the flow regime for high aspect ratio swept wings in the context of preliminary analysis and design. A computational experiment using a 3-D Navier-Stokes code is being developed to investigate the viscous flow along a swept wing segment. A 2-D grid is used with sweep simulated by imposing sideslip through boundary conditions at the edges with subsonic exit boundary conditions specified. The calculation will be run on representative airfoil sections in order to identify the effect of changing the character of the pressure distribution. This research will provide information on the capability of using appropriate 2-D C_{Lmax} data to infer the 3-D highlift characteristics of swept 3-D wings.

C. Institutional Support

Institutional support involves administrative, secretarial and technical salaries, travel, university equipment and services including communication, expendable supplies, computer services, engineering services, etc, and capital equipment. This support provides all of the basic services necessary for continuing operations of the Institute including its small-scale experimental and computational facilities, instrumentation and equipment and thereby supports all of the research and training activities summarized earlier.

D. Training activities

The training role of the Institute is accomplished through 6 units of coursework in acoustics offered by the Aero/Astro department including AA 201A (Fundamentals of Acoustics) and AA 201B (Topics in Aeroacoustics). In addition the Aero/Astro Lab course, AA131, incorporates an experiment based on an analogy which can be drawn between two-dimensional sound waves in air and surface waves generated in shallow water (of a depth chosen to provide nondispersive waves) to introduce students to the problem of flow generated noise. For the first time facilities at the Fluid Mechanics Laboratory at NASA Ames were used to carry out an experimental project for AA131 (Spring quarter 1995). The project involved measuring the effect of velocity difference on the strength of streamwise vortex structures in a plane mixing layer.

E. Research Participation

The research programs summarized in Item B above will be undertaken by Stanford faculty, staff and graduate students within the Department of Aeronautics and Astronautics with the involvement of 4 Professors, 2 Research Associates and 4 Ph.D. students. This group has experience in Aerodynamics and Acoustics and is familiar with NASA's wind tunnel and computational facilities. The strong collaboration between Stanford and Ames researchers which has been the hallmark of Joint Institute research in the past will be continued and enhanced in the coming year. The program activity at Ames will be coordinated by the Institute Associate Director for Center Affairs, Dr. C.A. Smith.

II. DETAILED PROGRAM DESCRIPTION

The research program proposed for the period October 1995 through September 1996 is described in detail below. It has been discussed with the cognizant personnel staff members at Ames and agreement reached on the general scope of the programs.

Project 1. Active flow control by tangential forebody blowing

Research participants: Prof. S. Rock, Dr. Z. Celik, graduate student

Ames Technical Contact: Dr. C.A. Smith

1.1 Introduction

The aerodynamics of an aircraft flying at high angles of attack is characterized by such phenomena as flow separation and vortex breakdown. Under these conditions the separation is usually asymmetric and the vehicle is subject to large lateral loads which cannot be overcome by conventional control surfaces. It is therefore necessary to find alternate means of augmenting the aircraft flight control system. Several methods of flow control have been studied and are of interest as a means to alter the flow structure in a such way that post-stall control of the aircraft becomes possible [Ref. 1-8].

The focus of this research is active flow control by the injection of a thin layer of air tangential to the forebody of the aircraft. The method is known as Forebody Tangential Blowing (FTB) has been proposed as an effective means of altering the flow over the forebody of the vehicle[Refs. 2,5]. By using this method, the flow asymmetries are changed and consequently the aerodynamic loads are modified. Static and dynamic experiments performed at the Department of Aeronautics and Astronautics at Stanford University under the NASA-JIAA program have shown that significant side force, roll and yaw moments as well as normal force and pitching moment [Refs. 5,8-10] can be generated using a small amount of blowing. This is important given that the implementation on a real aircraft would provide a limited amount of air. It has also been demonstrated that the FTB could successfully be used to suppress wing rock and to roll the model to a desired bank angle. In addition, it was shown that FTB could provide sufficient aerodynamic force and moments to use in a control scheme in two degrees of freedom [Refs. 9,10].

During this past year, dynamic experiments have been performed using a model support system that allows the wind tunnel model to move in two degrees of freedom, roll and yaw [Ref. 11]. This configuration is of interest because it better approximates the characteristics of the lateral-directional dynamics of an aircraft. It has been demonstrated that FTB could be used in a closed loop control logic to stabilize the two degree of freedom system [Ref. 12]. All these experiments have been performed in the low speed wind tunnel of the Department of Aeronautics and Astronautics at Stanford University.

1.2 Research Objectives

The overall objective of this research is to understand better the mechanisms through which tangential forebody blowing works and to demonstrate that the lateral control of a wind tunnel model is possible in two degrees of freedom. The ultimate purpose is to determine the feasibility of its use to control and/or improve the motion of an aircraft at high angles of attack.

Significant progress has been made during 1994-95. One of the fundamental extensions is to further develop the mathematical modeling and the control approach to encompass large roll and yaw angle control and to add a degree of freedom in the pitch direction. It is also our intention to improve the aerodynamic predictibility of the motion of the wind tunnel model. This will be achieved by investigating a tip geometry which would enable us to modify the flow over the forebody in a controllable manner. Blockage and wind tunnel wall interference effects should also be explored. The control algorithm which uses FTB in two degrees of freedom should be extended to include the vertical stabilizer and the rudder. Investigation of the effect of FTB on rudder buffeting is also proposed. These are the proposed future directions for the research.

1.3 Research Program

Experimental investigations are being conducted with a wind tunnel model provided with roll and yaw degrees of freedom. Static and dynamic measurements of the aerodynamic loads are being used to characterize the natural behavior of the system and the effect of blowing. A mathematical model of the system is being generated for use in the synthesis of control laws for the two degree of freedom system. Past work [Ref. 5] and the present experiments have shown that the effects of roll angle, yaw angle and blowing on the pitching moment and normal force are significant. It is also known that the flow characteristics will change when the angle of attack is changed [Ref.4,5,8]. Therefore investigation of the coupling between lateral and longitudinal modes will be conducted.

For the system with two degrees of freedom the regulator problem has been solved using a closed loop control based on a linearized model of the system. Research will be conducted to address the problem of commanding large roll and yaw angles, ϕ and γ respectively.

Due to the characteristics of the aerodynamic phenomena, non-linear control laws will be investigated as a means to incorporate this blowing "effector" into a flight control system while minimizing the amount of air used for control.

1.4 Summary of Activities Completed During 1994-1995

The following activities were completed during the 1994-1995 period:

- It was shown that the wing rock of the wind tunnel model with 60 and 70 degree swept wings and with and without the vertical tail could be successfully suppressed. The model with a 70 degree swept wing could be rolled up to 30 degrees by FTB in either direction and be successfully held at a desired roll angle.

- A suspension system that allows a model two degrees of freedom in the wind tunnel has been implemented. In particular, the sub-system that actively cancels external effects due to the supporting structure has been completed.

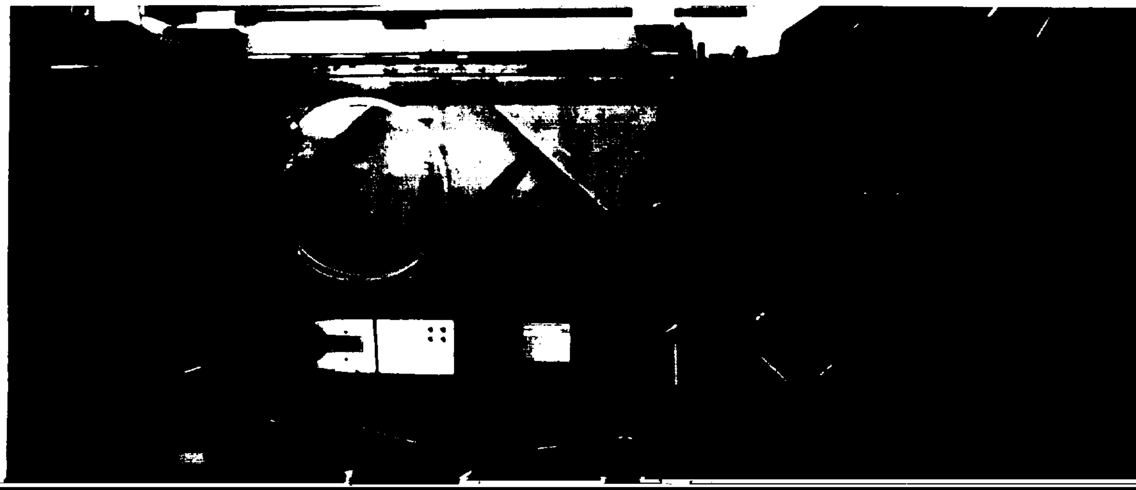
- A discrete vortex model has been developed that captures the interactions between forebody and wing vortices. The model correctly predicts the overall structure of the flow and the trends in the static loads.

- The mathematical structure of an unsteady aerodynamic model has been proposed that captures the main features of the flow. The model is coupled with the dynamic equations of motion of the system and will be used in the design of closed loop control laws.

- The feasibility of using forebody tangential blowing to control the roll-yaw motion of a wind tunnel model at high angle of attack has been experimentally demonstrated within the constraints of the experimental apparatus.

In the next section each of the completed activities is discussed in more detail and data are presented to support the results.

1.5 Research Activities Completed



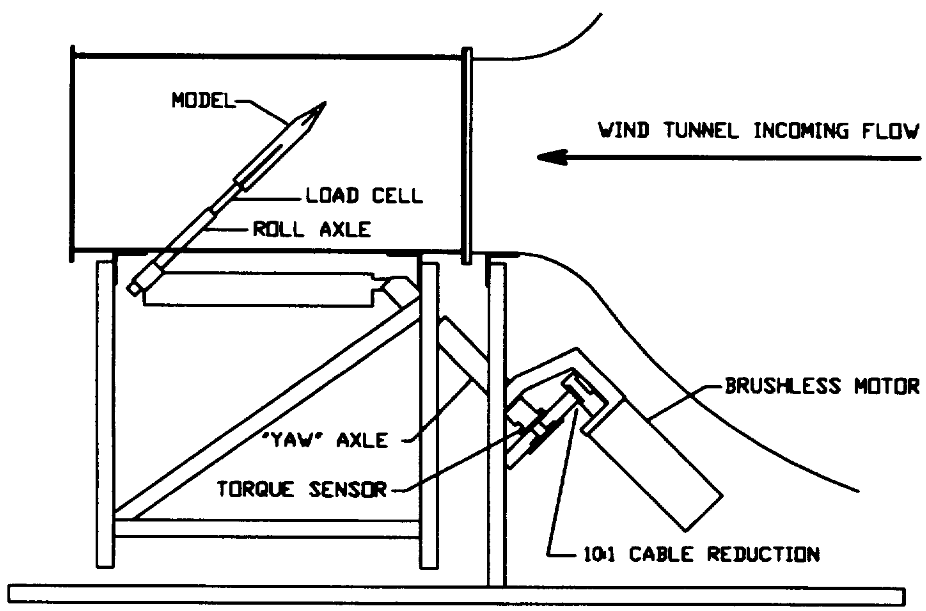


Figure 1.2: Side View of the Test Section

Design and construction of the experimental apparatus has been completed. In particular the active cancellation of external effects has been addressed during the 1994-95 period:

- A system to actively cancel the effects of the suspension has been designed, built and implemented. An overall objective in building the experimental apparatus was that the dynamic properties of the suspension should not dominate the dynamic response of the overall system. Examples of these added dynamic effects include the inertia and gravity restoring moment due to the geometry and configuration of the supporting structure, the friction on the bearings and the spring effect of the tubing used to supply the air for the blowing system.

- It has been verified that the sub-system which implements the roll degree of freedom, has a minimum adverse effect on the dynamic response of the system in roll. This was demonstrated by comparing the aerodynamic roll moment during wing rock with the friction effect of the bearings and the spring effect of the tubing [Ref.9,13].

- Simulations and preliminary tests have indicated that for the yaw sub-system the apparatus inertia and the gravity restoring moment have a large effect on the motion. Therefore a system has been designed and installed that provides active cancellation of these effects [Refs.11,12]. The operation of all the hardware and software components of this system has been verified and the system has been used in dynamic experiments with two degrees of freedom. Figure 1.3 shows the implementation of the active cancellation loop in a block diagram form.

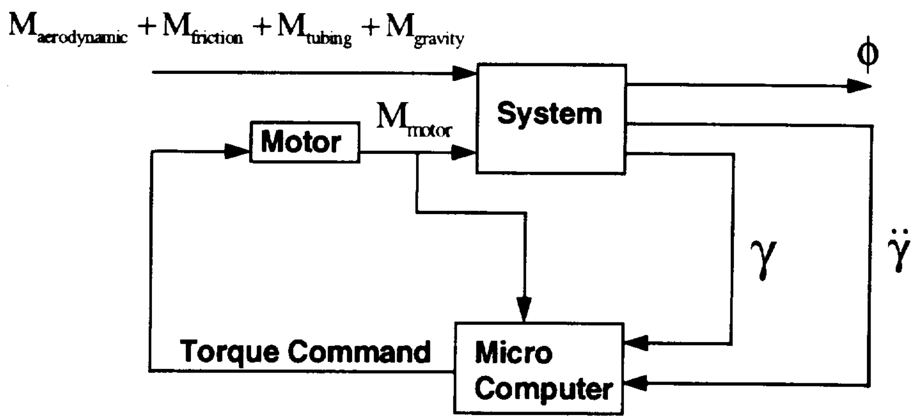


Figure 1.3: Concept and Implementation of Active Cancellation Loop

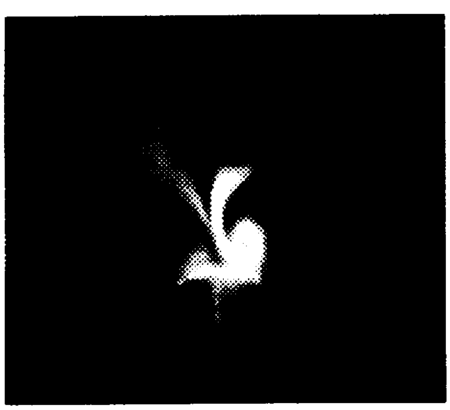
Progress has been made in the understanding of the aerodynamics of the phenomena and the use of blowing to control the roll-yaw motion of an aircraft at high angle of attack. The following results were achieved in the 1994-95 period.

1. Flow visualization experiments

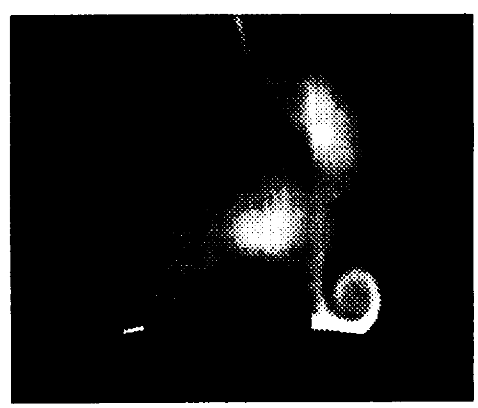
Past and present smoke and surface oil-flow experiments have revealed the basic structure of the flow [Ref.3,8]. Figure 1.4a shows the flow structure for a cross-section on the forebody of the model. The structure is clearly asymmetric with the left vortex (readers view) close to the fuselage and the right one away from it. The asymmetry scales up as one moves towards the rear portion of the model. Figure 1.4b shows the flow structure for a cross-section further downstream where the wing is present. On the left side only one large vortex is observed while on the right side two vortical structures are identified. The left wing vortex merges with the left forebody vortex due to its proximity to the fuselage. Although four main vortices are expected, two from the forebody and two from the wing leading-edges, in some experiments, particularly dynamic tests, it was observed that only three vortices could be clearly identified even for a symmetric condition in which roll and yaw angles are zero and no blowing is applied (Figure 1.4b). This is attributed to the location of the smoke relative to the burst vortex and resulting lack of suction to attract smoke, and the rapid dispersion of smoke at relatively high velocities.

The effect of asymmetric blowing, i.e. blowing applied on one side only, is mainly to increase the asymmetry or reflect it to its mirror image depending on which side the blowing is applied. Blowing moves the primary and the secondary separation lines on the forebody as reported in Refs. 5 and 8 and can cause a change in the amount of vorticity that is shed as shown computationally [Ref. 14]. As a consequence, the strength and positions of the vortices are affected by blowing.

The application of symmetric blowing has the effect of changing the flow structure to a more symmetric one. It also has a similar effect on the flow structure over a delta wing model [Refs. 3,4,7] such that the model would act as if it were at a lower angle of attack. For high values of symmetric blowing the flow on the forebody can be considered attached and its structure is symmetric even on the stations where the wing is present. This observation indicates a possible control strategy to be employed on the two degrees of freedom system: Application of symmetric blowing to remove or minimize flow asymmetries and an additional time varying asymmetric blowing to maintain stability of the vehicle.



(1.4a) Station 1 - Forebody



(1.4b) Station 2 - Wing-Body

Figure 1.4: Smoke Flow Visualization Results - $\phi=\gamma=0$, no blowing is applied.

2. Static Aerodynamic Loads

A quasi-three-dimensional potential discrete vortex model has been developed to provide insight on the interaction between forebody and wing vortices. The inclusion of blowing is currently been investigated and should be fully incorporated with the aerodynamic model by October 1995.

One concern regards whether a potential vortex method is appropriate, given that vortex breakdown may occur. To address this issue, a reference is made to the results of flow visualizations, for example Figure 1.4b. These experiments show that a vortical structure is present even on the side where the wing vortex merged with the forebody vortex. The causes of vortex breakdown are the object of discussion among researchers, but it is well accepted that the consequences are a diffusion of the vorticity and lower velocities in the vortex core. Therefore such effects can be modeled by increasing the vortex core radius and/or weakening the vortex as it moves along the vehicle. These effects are being studied using a discrete vortex program developed by Pedreiro [Ref. 15]. However the location of the vortex breakdown is dependent on various factors such as sweep angle, flow turbulence, Reynolds number, wall interference and blockage effects which can not be modelled accurately by potential flow solvers.

Preliminary results demonstrate that the method correctly captures the trends of the roll, yaw and pitching moments, Figure 1.5. Current work is being directed at modifying parameters such as vortex core radius and vortex strength factor, that requires experience with the code and a comparison with experiments so that the optimum values can be selected.

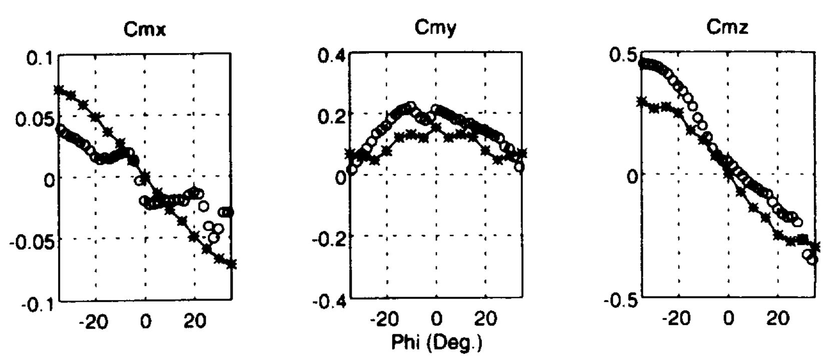


Figure 1.5: Results from the discrete vortex model. Comparison with experimental measurements. Cmx, Cmy and Cmz are the roll, pitch and yaw moment coefficients.

Figure 1.6 shows the result of the flow visualization and the positions of the discrete vortices calculated by the program at a wing-body station for ϕ=20 deg., γ=0, no blowing. The vortex cluster is too away from the model in the simulation and the wing vortex seems more tightly formed in the experiment, but the overall flow structure is well represented.

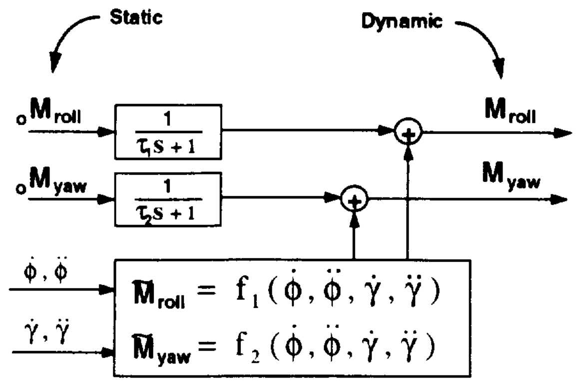
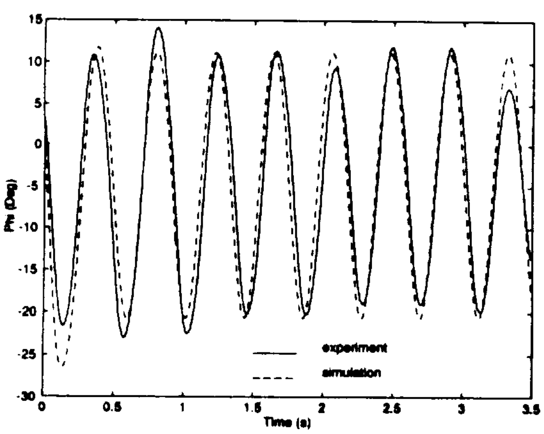
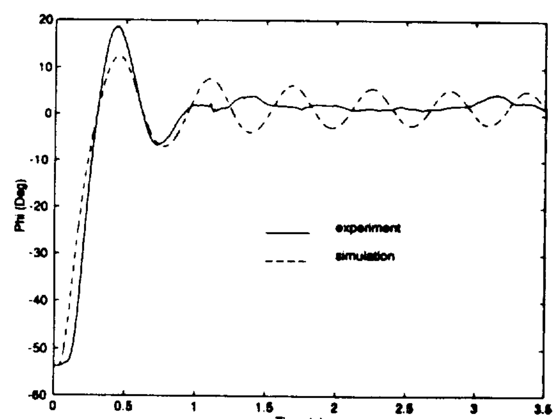


Figure 1.7: Block diagram of the aerodynamic model.

The time constants as well as the coefficients of the linear functions f1 and f2 are determined by using a minimum least squares fit of the simulated time histories of roll and yaw angles to the measured ones. Results of the mathematical model are compared to experimental data in Figure 1.8 for the cases with and without blowing and the wind tunnel model constrained to motion in one degree of freedom, roll.



(1.8a) No Blowing



(1.8b) Blowing Applied

Figure 1.8: Roll and yaw angle time histories. Comparison of simulation and experiment.

4. Closed Loop Control

The feasibility of using FTB to control the roll-yaw motion of a wind tunnel model at a high angle of attack has been demonstrated. The natural motion of the two degrees of freedom system is divergent as shown in Figure 1.9a. The system is therefore unstable and an initial objective was set to investigate the possibility of using FTB to stabilize the system dynamically since it was already shown in the static experiments that the FTB could generate sufficient side force and yawing moment to alleviate the asymmetries on the model [Ref. 8,10]. The unsteady aerodynamic model was coupled with the dynamic equations of motion to provide a description of the system. The equations were linearized about small roll and yaw angles and written in a form suitable to the design of closed loop control. A linear quadratic regulator design was performed and implemented in the experiment. The results are shown in Figure 1.9b. As can be seen the closed loop control stabilizes the system.

(1.9a) Natural Motion (1.9b) Closed Loop Control

Figure 1.9: Two Degree of Freedom System: Natural Motion and Closed Loop Control.

The offset in both roll and yaw angles is due to the fact that zero roll and yaw is not an equilibrium point. The offsets can probably be made smaller by adding some integral control. Refinements of the control logic are currently being investigated.

1.6 Research Activities Proposed for 1995-1996

Based on the results obtained to date, two main areas of research are proposed for 1995-1996.

1. The stabilization of the system with two degrees of freedom has been demonstrated (see Figure 1.9). A logical next step consists of investigating the possibility of commanding large roll and yaw angles. This problem is of utmost interest because it applies directly to the ability to point the vehicle to a desired direction. The issue associated with large angle maneuvers can be appreciated by comparing the effect of blowing at different roll and yaw angles, shown in Figure 1.10. The characteristics of the system vary for large roll and yaw angles making the control problem extremely difficult. Thus, the challenge is to model these variations so that control logic can be effectively applied.

The proposed approach to accomplish this modeling is to augment the present technique using indicial response methods to examine the functions of roll and yaw rates and their angular accelerations with respect to aircraft attitude. These methods will allow for a better understanding of the unsteady nature of the flowfield, which varies for large roll and yaw angles [Refs.16,17]. It should be noted that these investigations do not require any modifications of the apparatus or instrumentation available.

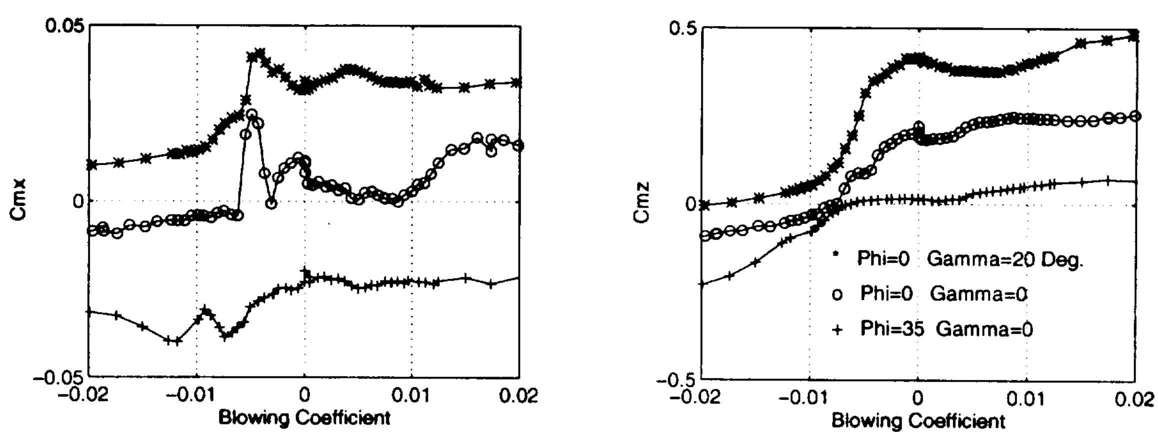
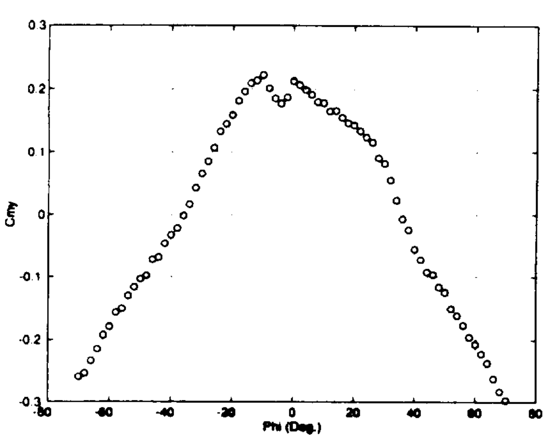
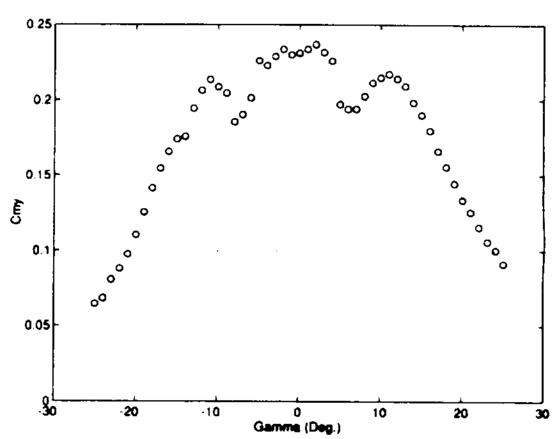


Figure 1.10: Effect of Blowing for Different Roll and Yaw Angles.

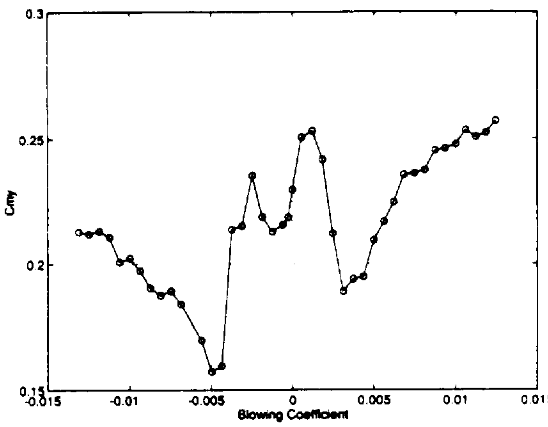
2. Measurements of the aerodynamic loads have shown that the effect of roll angle, yaw angle and blowing on pitching moment are considerable, Figure 1.11. It is also well known that as the angle of attack increases, the originally symmetric flow becomes asymmetric and as a consequence lateral loads occur even at symmetric flight conditions. Since the ultimate application of FTB is control augmentation for free-flying aircraft, it is important that the coupling between lateral and longitudinal modes be investigated. Research will be conducted to evaluate the coupling effects and to incorporate those in the design of the control logic for the vehicle. The modeling techniques that have successfully captured the main physics of the flow for roll-yaw motion will be extended to account for an additional degree of freedom in pitch. In addition, the nature of the angular rates and accelerations will be examined using functional analysis for insight into the dynamic interaction between aircraft and flowfield. The control laws will ultimately be demonstrated in the wind tunnel through an experiment in which the model is allowed three degrees of freedom. It is possible to add pitch as a third degree of freedom with some modifications on the existing apparatus[Ref. 18].



(1.11a) Roll Angle Effect



(1.11b) Yaw Angle Effect



(1.11c) Blowing Effect

Figure 1.11: Effect of Roll Angle, Yaw angle and Blowing on Pitch Moment.

Project 1 References

1. Ericsson, L.E. and Reding, J.P.,"Alleviation of Vortex Induced Asymmetric Loads", Journal of Spacecraft and Rockets, Vol 17, Nov.-Dec 1980, pp.533-548.

2. Skow, A.M., Moore, W.A. and Lorincz, D. J.,"Forebody Vortex Blowing-A Novel Concept to Enhance the Departure/Spin Recovery Characteristics of Fighter Aircraft," AGARD CP-262, NO. 24, May 1979, pp 24-1 to 24-17.

3. Wood, N. J., Roberts, L. and Celik, Z. Z., "The Control of Asymmetric Vortical Flows over Delta Wings at High Angles of Attack,"AIAA Paper 89-3347, Aug. 1989.

4. Celik, Z.Z., Roberts, L. and Wood, N,"An Investigation of Asymmetric Vortical Flows over Delta Wings with Tangential Leading Edge Blowing at High Angles of Attack," AIAA Paper 90-0103 , Jan. 1990.

5. Celik, Z.Z. and Roberts, L., "Vortical Flow Control on a Slender Body at High Angles of Attack,"AIAA Paper 91-2868,Atmospheric Flight Mechanics Conference, August 1991.

6. Rom, J., "High Angle of Attack Aerodynamics," Springer-Verlag, 1992

7. Wong, G.S., "Experiments in the Control of Wing Rock at High Angles of Attack Using Tangential Leading Edge Blowing," Ph.D. Dissertation, Aeronautics Department, Stanford University, 1991.

8. Celik, Z.Z. and Roberts, L., "Vortical Flow Control on a Wing-Body Combination Using Tangential Blowing," AIAA Atmospheric Flight Mechanics Conference, August 10-12, 1992, Hilton Head Island, SC.

9. Celik, Z.Z. and Pedreiro, N. and Roberts, L.,"The Control of Wing Rock by Forebody Blowing," AIAA Paper 93-4386, Aug. 1993.

10. Celik, Z.Z. and Pedreiro, N. and Roberts, L.,"Dynamic Roll and Yaw Control by Tangential Forebody Blowing," AIAA Paper 94-1853, June 1994.

11. Pedreiro, N., "Design of a Suspension System for Wind Tunnel Dynamic Experiments with Two Degrees of Freedom," Internal Report, Aeronautics Department, Stanford University, 1993. To be published as part of Doctoral Thesis.

12. Pedreiro, N., "Roll-Yaw Control at High Angle of Attack Using Forebody Tangential Blowing - LQR Design," Internal Report, Aeronautics Department, Stanford University, 1995. To be published as part of Doctoral Thesis.

13. Pedreiro, N., "Effects of Friction, Tubing and Extra Inertia on the Free to Roll System," Internal Report, Aeronautics Department, Stanford University, 1993. To be published as part of Doctoral Thesis.

14. Font, G. I., "Tangential Fuselage Blowing on an Ogive Cylinder," Stanford University Doctoral Thesis, Department of Aeronautics and Astronautics, June, 1992.

15. Pedreiro, N., "VTX6A - A Discrete Vortex Program for Wing-Body Combination at High Angle of Attack," Stanford University, 1994-95. To be published as part of Doctoral Thesis.

16. Tobak, M., Chapman, G. T. and Schiff, L.B., "Mathematical Modeling of the Aerodynamic Characteristics in Flight Dynamics," NASA Technical Memorandum 85880, January 1984.

17. Jenkins, J. E., "Relationships Among Nonlinear Aerodynamic Indicial response Models, Oscillatory Motion Data and Stability derivatives," AIAA Paper 89-3351-CP, 1989.

18. Rock, S.M., Private communication. April 1995.

Project 2 - Large Eddy Simulation as a Tool for Studying Jet Aero-Acoustics

Research Participants: Prof. S. Lele, graduate student

Ames Technical Contact: Dr. R. Moser

2.1 Introduction

The prediction and control of aerodynamically generated noise is important to the design of less noisy subsonic and supersonic aircraft. Current prediction methods strongly rely upon empirically obtained data and use some form of acoustic analogy (pioneered by Lighthill) for scaling the data. The predictive capabilities of such tools is limited by their empirical origins. With the recent advances in computational algorithms and computer hardware, a new generation of analysis/design tools can be developed which are based upon the underlying physical principles. In the area of aero-acoustic analysis/prediction, Large Eddy Simulation (LES) holds the promise of predicting, at least, the dominant features of the noise radiated to the far-field by a flow such as a jet issuing from a nozzle.

Predicting the far-field noise via LES is far more challenging than an overall prediction of the near-field aerodynamic flow. Acoustic predictions are dependent upon a two-point space-time scales have been developed to allow an overall statistical prediction of the aerodynamic near-field. The models are typically designed to provide correct energy transfer between the resolved and unresolved scales (Ghosal et al. 1992). While these models seem quite promising, their impact on far-field acoustics has not been considered. Since the far-field noise is a very small by-product of the flow it is necessary to ensure that the subgrid models do not behave as a low order (and hence efficient), but spurious source of sound. As Crighton (1988) points out, sources of this type may be introduced via discretization errors and numerical boundary conditions, etc. It is necessary to examine subgrid scale models as well in this context. Furthermore for practical LES applications, the subgrid scale energy may be as much as 10-30% of the resolved energy and this may require that closer attention be paid to the acoustic sources implied by such models.

2.3 Research Program

We propose to carry out a program of research aimed at extending/developing the LES methodology from the point of view of far-field noise prediction. The fidelity of LES in predicting the unsteady flow and acoustic sources will be judged by making extensive comparisons with a Direct Numerical Simulation (DNS) of the same flow configuration. For this reason, the initial study is direct simulation of turbulent flow in simple geometries.

2.4 Research Activity Completed

A formulation capable of yielding a stationary turbulent jet flow while maintaining the efficiency and accuracy of spectral methods was developed. The method is an extension of Spalart's method (1988) for simulating boundary layers, to the case of a co-flowing jet or wake. The results resemble those obtained by Spalart (1986) for the sink-flow boundary layer. The method involves incorporating the slow spatial growth effects via a decomposition of the variables according to their multiple spatial scales and a suitable coordinate transformation (Timson et al. 1994). The derivation is more rigorous than the boundary layer analysis, due to the simplification introduced by explicitly considering the small deficit limit.

The result of the formulation is a modified set of equations consisting of the Navier Stokes equations with an additional set of small growth terms. The implied flow field is homogeneous in both the streamwise and spanwise directions, and was therefore implemented in the spectral code used by Rogers and Moser (1993). Testing of the code's ability to maintain a stationary flow was carried out, and a preliminary Direct Numerical Simulation was completed.

A flow field from a previously completed, temporally evolving wake simulation done by Moser and Roger (1994), was used as an initial condition for the DNS. Time histories of momentum thickness and turbulent kinetic energy during the simulation are shown in Figure 1. All

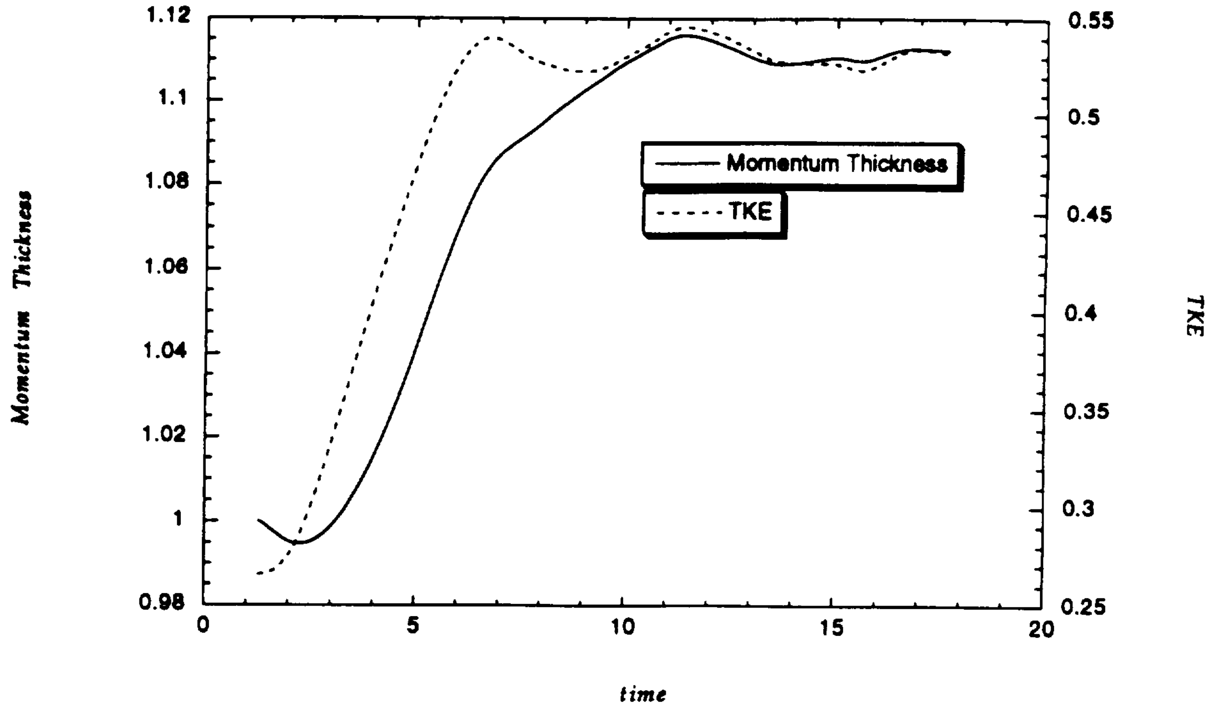


Figure 1: Time histories of momentum thickness and integrated turbulent kinetic energy (TKE).

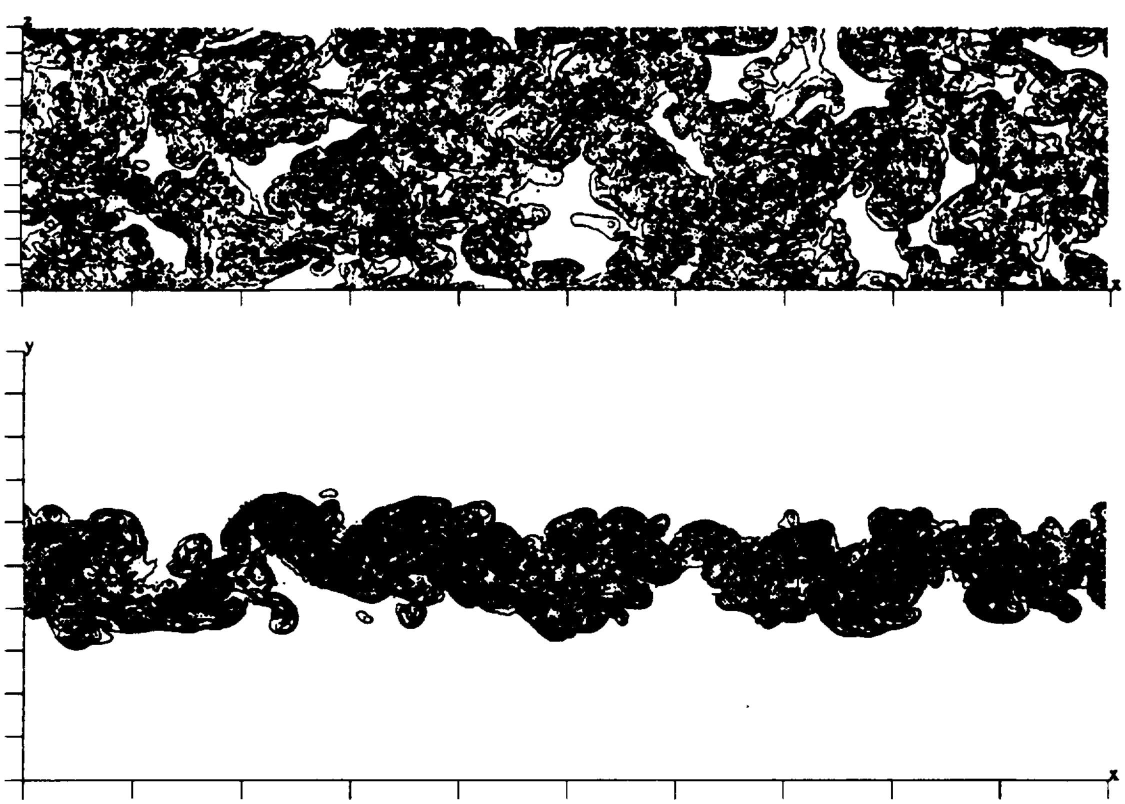


Figure 2: Contours of vorticity magnitude for the final time. Shown from the top view (top) and side view (bottom).

quantities are made dimensionless using the half-width of the average wake profile and its maximum velocity deficit. It is clear that a statistically steady evolution is achieved. There is some variation about the mean as would be expected as large scale structures are formed and destroyed. A snapshot of the vorticity magnitude at the final time in Figure 1 is shown in Figure 2 from both the top and edge view.

2.5 Research Activities Proposed for 1995-1996

The DNS will first be extended to collect the data necessary to make acoustic predictions. This will involve continuing the computation from the final condition of Figure 1 and computing the two point space-time correlations necessary for acoustic predictions. Concurrently the code will be modified to perform the Large Eddy Simulation of the same flow field. The LES calculations will begin with the simplest Smagorinsky type subgrid scale eddy viscosity models.

Comparison of the LES and DNS results will begin with one point statistics and move on to the two-point correlations mentioned previously. If the LES and DNS results compare well at this level it may be concluded that the dominant acoustic sources have been effectively modeled (at least in the context of an acoustic analogy).

2.6 Research Activities Planned Beyond 1995-1996

Once the initial tests of LES's fidelity have been carried out using a simple subgrid scale model, the direct impact of different subgrid scale models will be directly examined. This will require a study of the space time correlation of the model subgrid stresses and the resolved stresses in the DNS database. It is expected that this will involve reintegrating the DNS data from the coarsely spaced times available as restart files. When the subgrid scale energy is non-negligible these correlations may provide information about how much noise is radiated by the unresolved subgrid scale motions and how effectively it is captured by subgrid models employed in the calculations.

Once the efficacy of LES has been established. The prediction of far-field noise which uses the near-field unsteady flow calculated via LES can be applied to flows of engineering interest. LES calculations of an experimental flow for which detailed flow and noise data exist may be the next logical step in this direction.

Project 2 References

1. Crighton, D.G., Goals for computational aeroacoustics, in Computational Acoustics: Algorithms and Applications, Elsevier Science Publishers B. V., IMACS, 1988

2. Lighthill, M. J., On sound generated aerodynamically {I}. General Theory, Pro. Roy. Soc. (A), 1952, 211, 564-587

3. Ghosal, S. and Lund, T. S. and Moin P., A dynamic localization model for large eddy simulation of turbulent flows, Center for Turbulence Research Manuscript, 1992, 139

4. Moser, R. D. and Rogers, M. M., Direct simulation of a self-similar plane wake, NASA TM 108815, 1994

5. Moser, R. D. and Rogers, M. M., Direct simulation of a self-similar turbulent mixing layer, Phys. Fluids, 1994, 6, 903-923

6. Spalart, P. R., Numerical study of sink flow boundary layers, J. Fluid Mech, 1986, 172, 306-328

7. Spalart, P. R., Direct simulation of a turbulent boundary layer up to Rq=1410, J. Fluid Mech, 1988, 187, 61-98

8. Spalart, P. R. and Moser, R. D., and Rogers, M. M., 1991, Spectral methods for the Navier Stokes equations with one infinite and two periodic directions, J. Comput. Phys, 96, 297-324

9. Timson, S. F. and Lele, S. K. and Moser, R. D., A method for obtaining a statistically stationary turbulent free shear flow, Center for Turbulence Research Annual Research Briefs, 1994, 365-371

Project 3 - Research on a lifting wing-flap combination with application to airframe generated noise

Research Participants: Prof. B. Cantwell, Dr. Z. Celik, graduate student

Ames Technical Contact: Dr. L. Olsen, Dr. J. Ross

3.1 Introduction

Experiments by Kendall and his co-workers (Refs 1, 3, and 4) and Grosche et al. (Ref. 2), using a directional microphone to measure noise distributions on a wind tunnel model, found that the noise generation was highly localized. The most active locations were the flaps, the wing tip and leading edge, the trailing corner and trailing edge of the flap and the gap separating adjacent flap elements. Areas where the attached turbulent flow existed were found to be non-active for noise generation. In experiments with individual flaps, vortex roll-up was considered to be a major reason for noise generation. When the vortex strength was reduced, the noise intensity decreased. Kendall observed that the major part of the noise generation was caused by the gap between two flaps which were deflected differentially. When the flaps were retracted, the noise intensity distribution was high along the gap between the flap and the wing. Depending on the flap configuration, the noise intensity distribution showed point-like or line-like sources. Kendall also argued that the trailing edge noise did not play an important role.

Recent experiments (Refs. 6 and 7) on a wing-half span flap configuration by NASA investigators in the Ames 7 x 10 included a collaborative program of noise measurements by Boeing researchers. The flap edge was clearly identified as the major source of noise and various edge treatments were tested in an attempt to reduce noise. Large differences in the effectiveness of noise reduction were observed depending on the particular choice of flap edge treatment.

Fluid dynamic sources for noise generation from a lifting surface include confluent turbulent boundary layers and vortex sheets rolling over edges producing large surface pressure fluctuations. The flow involves a wide range of length scales, high local shearing stress and intense turbulence activity over the lifting surface. A better understanding of these fluid dynamic processes is needed for the development of effective methods for airframe noise reduction.

3.2 Objective

The objective of this research is to understand the flow mechanisms responsible for noise generation by a wing and trailing edge flap combination. An NACA 63-215 Mod B airfoil section has been selected for flow measurements at the conjunction of the main element, cove section and

Fowler flap. For comparison purposes, the same model geometry used in the 7 x 10 experiments has been selected for computations by Stanford and Ames investigators and for small scale experiments at Stanford. A 1/3 scale version of the wing flap model is presently being built for flow visualization studies at Stanford. The Stanford experiments will include fluorescent dye visualization in a low speed water channel and smoke visualization and unsteady pressure studies in a subsonic wind tunnel. Various flap edge treatments will be used to see how they modify the flow field about the flap. The visualization studies coupled with the computations will provide a useful tool for investigating the flap aerodynamics. Finally, in a separate study, hot wire measurements will be carried out in the 7x10 in the near wake of the wing flap model in order to determine local turbulence intensities and scales at the higher Reynolds number available in this facility. The facilities at Stanford University and NASA-Ames will enable us to investigate the flap edge flow field over a range of Reynolds numbers from 50,000 to 2.5×10^6.

3.3 Progress of the Research:

In the past year, our efforts were concentrated in the following areas.

(1) The identical model geometry was selected for experiments both at Stanford and NASA-Ames and also for the computational work. Differences between old geometrical data and the newly designed portions of the wing, especially, of the flap and the CFD were eliminated after the surface geometry was remeasured on the existing model at NASA-Ames.

(2) The design and construction of the wing model, the main wing and the split flap portion, and the model support system is nearly completed. Because of the small test section size of the wind tunnel and the water channel at Stanford compared to Ames experiments, the wing model had to be scaled accordingly. The same aspect ratio was retained between the two models. Our efforts concentrated on a model design which could be used in both facilities with minimum modification. The size of the model also restricted the available choices for selecting the number of pressure tappings and pressure sensors. The wing and flap are mounted separately allowing maximum flexibility in choosing the flap gap and angle.

As shown in Figure 1, the model is designed with two interchangeable middle parts which were instrumented accordingly for the wind tunnel and the water channel environments. The material for the model is aluminum to minimize the risk of corrosion in the water channel. For the wind tunnel experiments, the middle section is instrumented with pressure tappings and with pressure sensors mounted under the surface. For the water channel experiments, the middle section is replaced with a geometrically identical section containing dye ports for flow visualization. Instrumentation for the wind tunnel experiments includes 35 pressure tappings on the main section and three pressure sensors on the side of the split flap (Figure 2a). Of the pressure tappings on the main section, any tap can be instrumented with a pressure sensor if desired. For the flow visualization experiments, the main wing has 12 dye ports; four on the top surface, seven on the bottom surface and one at the edge of the cove section (Figure 2b). This gives us the ability to inject dye at different locations of interest. Of three dye holes on the flap, one is located on the top surface while the other two will be used to inject dye on the lower surface.

A support system for the main wing and the flap was designed and manufactured for the water channel. With this system, the angle of attack of the model, flap deflection, flap gap and overlap ratio can be adjusted independently. In addition, end-plates are used to minimize the interaction of the model with the wall boundary layer. The main element of the wing is supported at both ends. The flap is attached at one end only to eliminate the interference which would otherwise be caused by mounting struts. The model support system is shown in Figure 3.

The water channel facility is of moderate size, with a 12 ft long test section designed in a single section for uninterrupted viewing of the flow. The channel cross-section is 20 in.x 28 in. A test section velocity of 1ft/sec is achievable with the pumps operating at maximum capacity and the channel at maximum depth. Given this velocity and the constraint on model size, the maximum chord Reynolds number is approximately 70,000. Optical access to the model is through the glass sidewalls and bottom of the test section as well as through a downstream window. At the present time the facility is in final assembly with testing and characterization of the flow quality scheduled for late July and initial visualization studies of the wing flap model scheduled to begin in August. A schematic of the water channel is shown in Figure 4.

3.4 Proposed Work:

The following experiments are planned.

(1) Wind Tunnel Experiments:

Surface mean and unsteady pressure measurements and wake mean velocity measurements will be carried out at Reynolds Numbers ranging from 400000 to 800000 on a 9-inch chord model in the low-subsonic wind tunnel at Stanford using the settings of the test matrix used in the experiments at Ames [Refs 8,9]. In addition to these settings, a wide range of angle of attack and flap settings will also be explored to investigate the flow structure around the wind tunnel model. The model is designed so that the flap angle and gap can be varied easily without any direct connection to the main wing. Smoke flow and surface oil visualizations will be conducted at the conditions given in the test matrix of Ames experiments. Cove tabs and flap edge treatments devised at Ames will also be studied in the Ames experiments. These experiments will provide a visual understanding of the unsteady flow field and they will provide data at several Reynolds numbers for comparison with the computations carried out on the same wing-flap geometry.

(2) Water Channel Experiments:

Concurrent with the construction of the facility, research has been done to evaluate a variety of flow diagnostic techniques. For the initial flow visualization dye injection will be used to characterize the flow-field and the model has been designed with a variety of dye injection ports to tag interesting flow regions [Figure 2b]. The Reynolds number for these experiments will be approximately 50-70000 based on the chord. Various color dyes will be used on the main airfoil and the flap to visualize the flow interaction between the airfoil elements. For surface measurements, it has been established that the surface oil-flow technique can be successfully applied underwater at the velocities the model will be tested.

(3) CFD Computations and the Correlation Between Flow Properties and Noise Measurements:

Under a separate grant, research is in progress to study ways to correlate the flow properties and the noise intensity distribution around the airfoil. For this purpose, CFD computations of the flow over the split-flap configuration are under way in an effort to determine which flow properties, i.e. Reynolds stresses, vorticity distribution and unsteady pressure distribution, will correlate best with the recent noise measurements done at NASA Ames Research Center [Ref. 8]. Preliminary results indicate that the unsteady surface pressure measurements would correlate better with the noise measurements. Our efforts next year will concentrate in that direction.

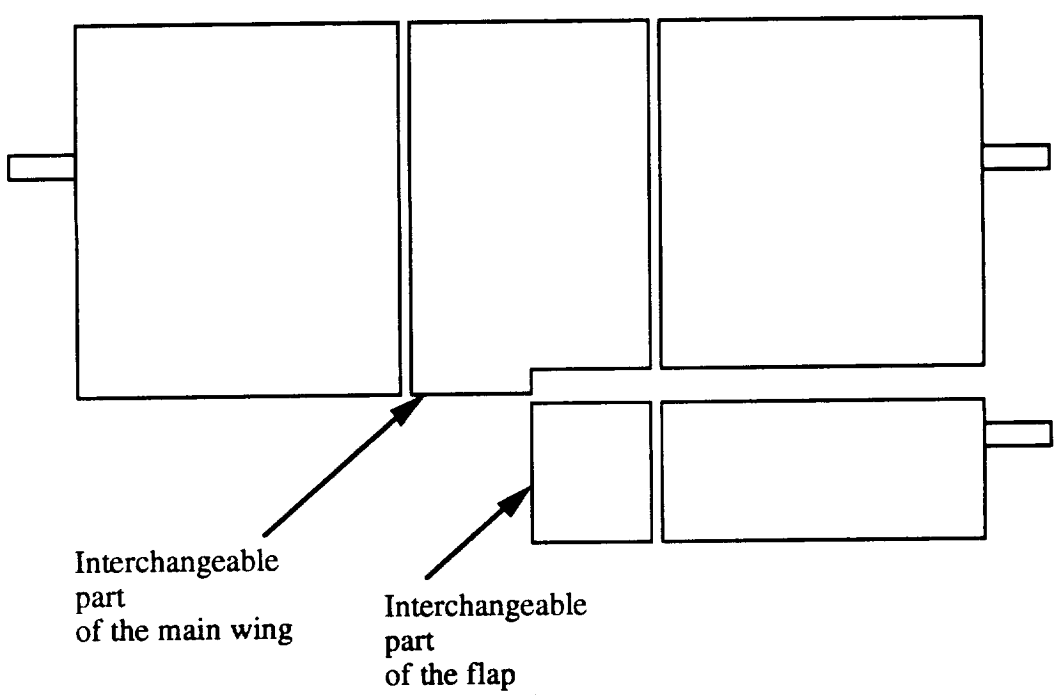
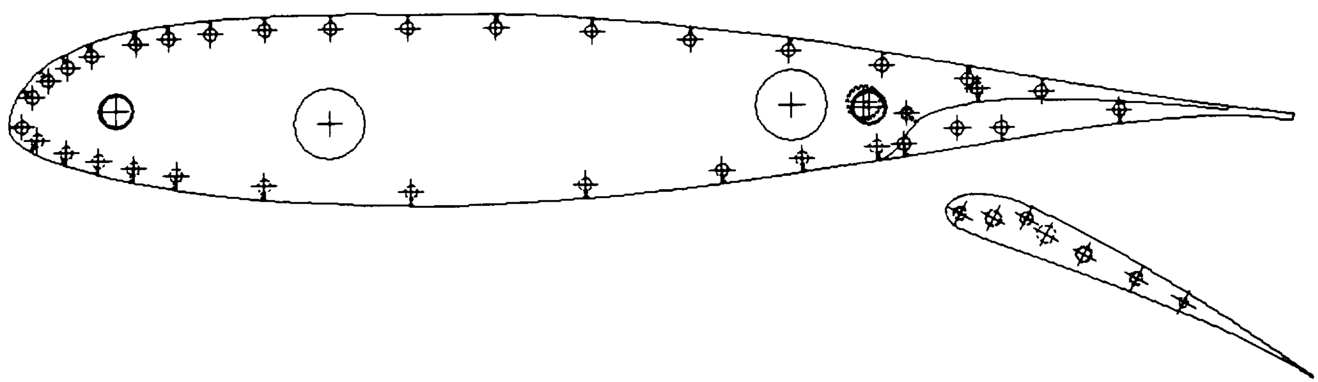
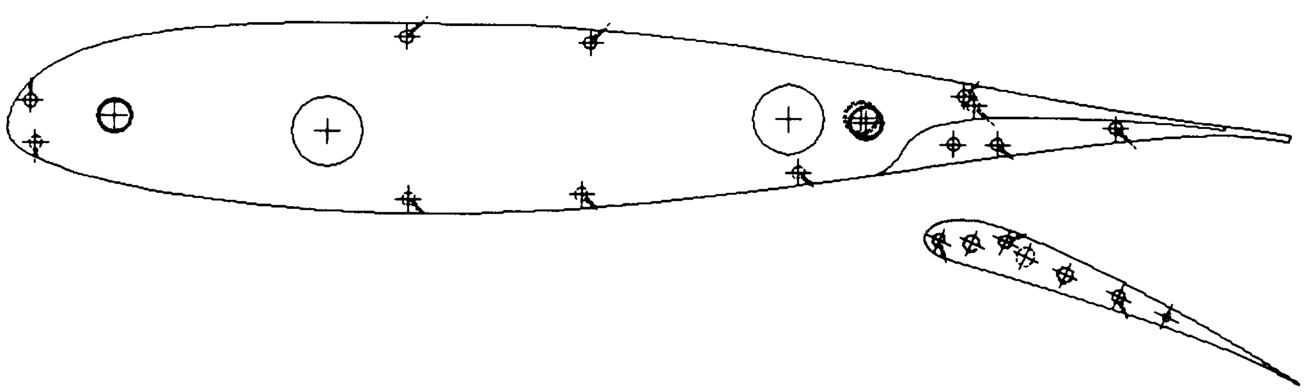


Figure 1: NACA 63-215 Mod B Airfoil with Fowler Flap and regions of interest.



(a)



(b)

Figure 2: (a) Pressure tappings and sensor locations of the model for wind tunnel experiments and (b) Dye port locations of the model for water channel experiments.

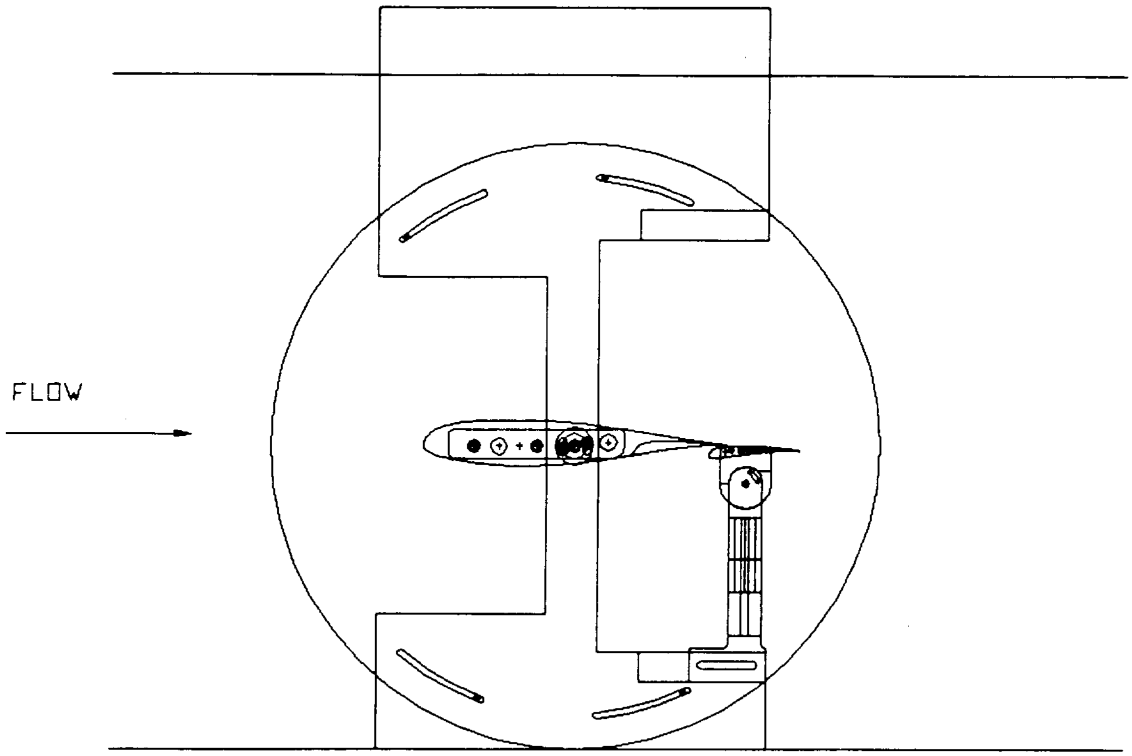


Figure 3: A schematic of the model support system for water channel experiments.

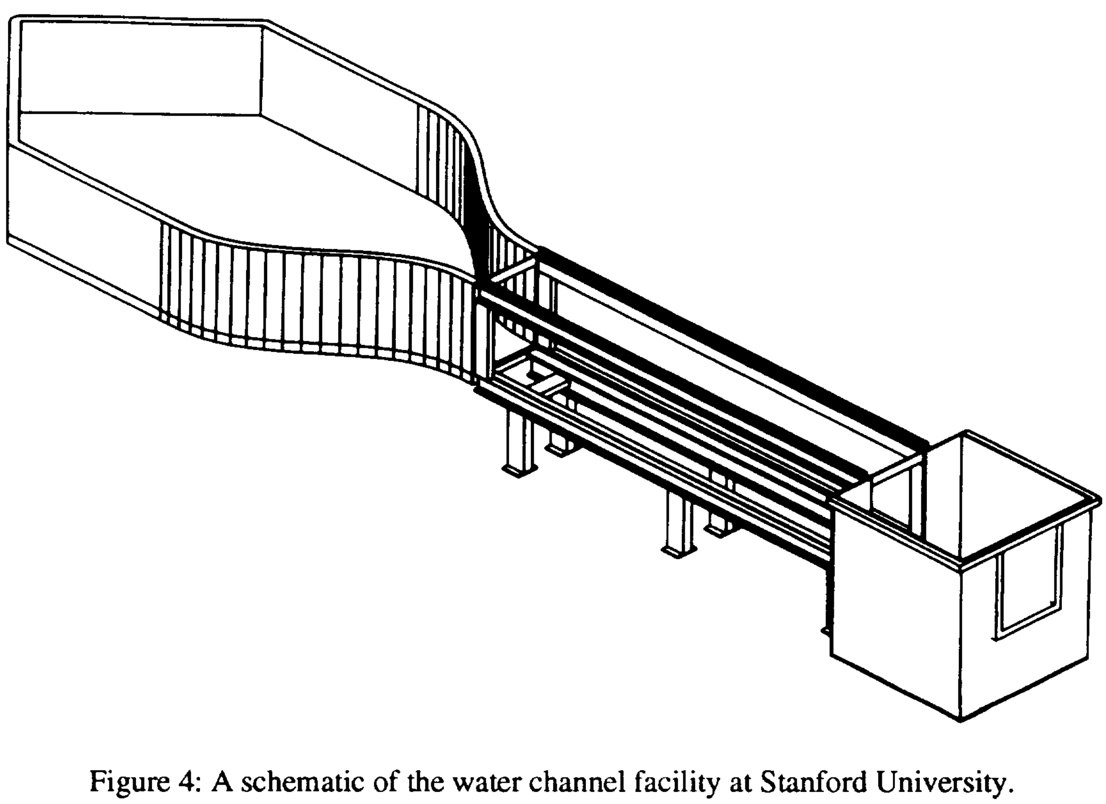


Figure 4: A schematic of the water channel facility at Stanford University.

3.5 Research Activities Planned Beyond 1995-1996

Pressure measurements initiated in 95-96 will be completed. At this stage of the research, flow velocity measurements will be obtained using Particle Image Velocimetry in the water channel and a multihole probe in the wind tunnel. Mean velocity measurements will be carried out using an available 5-hole probe. Mean measurements are required for comparison with the CFD results, for assessment of the turbulence model(s) and for examining the effects of wall interference. To enable us to measure the unsteady velocity field near the flap edge, we have begun to put together the hardware needed for a a low resolution Digital Particle Image Velocimetry system. After gaining experience with this basic system we plan to switch to a higher resolution camera to obtain measurements at an accuracy suitable for comparison with wind tunnel or numerical results. The velocity measurements will provide information needed to evaluate source strengths and to correlate the flow visualization, surface pressure measurements and acoustic surveys.

Project 3 References

[1] Kendall, J.M., "Airframe Noise Measurements by Acoustic Imaging," AIAA 77-0055, January 1977.

[2] Grosche, F.R., Stiewitt, H and Binder, B., "Acoustic Wind Tunnel Measurements with a Highly Directional Microphone," AIAA Journal, Vol. 15, No. 11, pp.1590-1596, November 1977.

[3] Kendall, J.M., "Measurements of Noise Produced by Flow Past Lifting Surfaces," AIAA 78-0239, January 1978.

[4] Kendall, J.M. and Ahtye, W.H., "Noise Generation by a Lifting Wing/Flap Combination at Reynolds Numbers to 2.8 x 106," AIAA 80-0035, January 1980.

[5] Celik, Z. Z., "Flow Measurements of an Airfoil with a Slotted Flap," AIAA 89-0533, January 1989.

[6] Storm, B. and Ross, J.,"Flap Edge Post-Test Debrief," NASA TR-762, 1995.

[7] Takahashi, T.,"Wake Survey Experiments of NACA 63-215 Mod B Airfoil," 1995.

Project 4 - Application of Luminescent Paint Sensors to Fluid Physics Problems

Research Participants: Dr. R. Mehta, Graduate Student

Ames Technical Contact: Dr. S. Davis

4.1 Introduction

Pressure sensitive luminescent paints have some important advantages over pressure taps which are typically used for surface pressure measurements on wind tunnel models. For example, the paint provides a measurement over the entire model surface and measurements with pressure sensitive paint hold the potential of being much less expensive (over time) than those with pressure taps.

Pressure sensitive paints are now used routinely for measuring surface pressures on wind tunnel models at transonic and supersonic Mach numbers (Refs. 1-4). The method utilizes a surface coating containing fluorescent or phosphorescent materials, the brightness of which varies with the local air pressure on the surface. In current practice, a wind tunnel model is coated with the luminescent material, which is then illuminated with light of an appropriate wavelength to excite the material. The illuminated model is imaged with a digital CCD camera during the wind tunnel test. The images are then computer-processed in order to obtain a map of the surface pressure distribution. The relationship between surface brightness and pressure is generally determined by calibrating the paint (in situ) using a few pressure taps on the model.

The pressure sensitive paint technology is by no means fully mature. There are a number of areas where the technique is currently undergoing improvements, such as sensitivity to temperature and motion of the model, for example. Some attempts to obtain surface pressure measurements in low-speed ($M < 0.1$-0.2) flows are also now being reported. Further work is needed in order to improve the accuracy of the technique and to make it more versatile.

4.2 Research Objectives

The overall program objective is to apply the luminescent paint technolgy to the study of basic fluid physics problems, especially at subsonic speeds.

The successful use of pressure sensitive luminescent paints at transonic and supersonic speeds has been well demonstrated in tests conducted over the last five years. In present day aeronautics, low-speed ($M < 0.1$-0.2) testing is becoming increasingly relevant. For example, complex multi-element systems are being designed for subsonic and supersonic transports for the take-off and landing phases which must be extensively studied and tested in wind tunnel experiments. The pressure sensitive paint has the potential of providing surface pressures over the entire model in a relatively fast and cheap manner. However, the one limitation of the pressure sensitive paint is that the brightness of the paint is inversely proportional to the pressure. Therefore, detection of differences in brightness, which relate to differences in pressure on the model, become increasingly difficult as the flow speed is reduced. Our objective is to first bench test various paint compositions supplied mainly by our collaborators at Purdue University. Once the promising paint compositions for low-speed testing have been identified, they will be used to measure pressures on aerodynamic surfaces in the Fluid Mechanics Laboratory (FML) subsonic wind tunnels. Another possible application will be in the model testing of the National Wind Tunnel Complex components, scheduled to take place in the FML in the coming year.

Plans are also under way to use the pressure sensitive paint on a high aspect ratio wing in the High Reynolds Facility and to use pressure and temperature sensitive paints (for transition detection) in the new Supersonic Quiet Wind Tunnel.

4.3 Progress and Activities

Currently, an imaging system consisting of a Digital Cooled CCD Camera linked to a 486 Personal Computer (PC) is being used in a "bench-top" experiment. The bench-top test consists of an inclined turbulent jet (shop air) impinging on a flat plate. The surface pressures on the plate are varied by controlling the flow velocity and also the angle and height of the jet above the plate. Suitable paints and the optimum application techniques are also being identified in these tests. So far the results indicate that certain paint compositions which were initially thought te be very responsive to low pressures were, in fact, responding to some contamination in the shop air. The symptoms suggest that the (dried) shop air may contain excessive oxygen since the measurement technique relies on oxygen quenching. Subsequent tests have been performed with a nitrogen jet so that at least the qualitative behavior of the paints may be assessed. This last observation has inspired an experiment, currently being performed by a graduate student at Stanford University, where the surface interaction of two inclined jets will be studied by supplying one jet with nitrogen and the other with oxygen. The effects of nozzle geometry on the interaction will also be investigated.

In an experimental project, designed as part of the Experimental Techniques Class in the Aero/Astro Department at Stanford, the effects of velocity ratio on mixing layer three-dimensionality are being investigated. This work is being performed in the Mixing Layer Wind Tunnel in the FML by a graduate student. These types of student projects will continue to be offered in the coming year. One project under consideration is that on baseball aerodynamics with the possible application of the pressure sensitive paint.

Project 4 References

(1) Kavandi, J., Callis, J., Gouterman, M., Khalil, G., Wright, D., Green, E., Burns, D., McLachlan, B., 1990, "Luminescent Barometry in Wind Tunnels", Rev. Sci. Instruments, Vol. 61, No. 5, November, pp. 3340-3347.

(2) McLachlan, B.G., Kavandi, J.L., Callis, J.B., Gouterman, M., Green, D., Khalil, G., D. Burns, 1992, "Surface Pressure Field Mapping Using Luminescent Coatings", 1993, Expt. Fluids, Vol. 14, pp. 33-41.

(4) McLachlan, B., Bell, J., Kennelly, R., Schreiner, J., Strong, J., 1992, "Pressure Sensitive Paint Use in the Supersonic High-Sweep Oblique Wing (SHOW) Test", AIAA-92-2868, 10th AIAA Applied Aerodynamics Conference, Palo Alto, California, Jun 22-24, 1992.

(5) Bell, J.H., McLachlan, B.G. "Image Registration for Luminescent Paint Sensors", AIAA-93-0178, 31st Aerospace Sciences Meeting, Reno, Nevada, Jan 11-14, 1993.

Project 5 - Prediction of Wing Maximum Lift for Preliminary Design Methods

Research Participants: Prof. I. Kroo, Graduate Student

Ames Technical Contact: Dr. L. Olsen, Dr. S. Smith

5.1 Summary

The high lift characteristics of wings have important effects on aircraft noise, cost and performance. The proposed research is aimed at improved understanding of the high lift flow regime for high aspect ratio wings in the context of preliminary analysis and design. Computational models will be developed and used to examine the important inviscid and viscous phenomena that effect wing maximum lift, as well as the importance of three-dimensionality on the flowfield. The ultimate goal is to develop an analysis routine that accurately predicts wing maximum lift with the speed, accuracy and sensitivity necessary for use in multidisciplinary optimization design codes.

5.2 Introduction

Recent advances in the field of multidisciplinary optimization make this approach promising for use in the conceptual design of new aircraft. Traditional aircraft selection methods, such as parametric studies and summary charts, allow the designer to pick the "best" design based on variation of a limited number of parameters. This best design is, however, still sensitive to many other variables that were not examined in the original parametric studies. The true optimum design can only be found when all of the available design parameters are varied simultaneously in order to find the best combination that meets all of the given constraints. The major benefit of computerized optimization for aircraft design is the ability to perform these trade studies with many more parameters and much greater speed than can be accomplished through traditional methods.

While numerical optimization provides the benefit of more thorough exploration of the design space, it also presents new challenges for the modeling of the aircraft throughout that design space. In order to best meet their objectives, optimizers tend to push designs to their limits. This sometimes results in exploitation of weaknesses in the aircraft model, producing designs that are not really feasible even though they look very good to the optimizer. For this reason, one of the most important issues in using optimization for aircraft design is accurately modeling the various effects that drive the design. These models must also be simple enough to run quickly since an optimizer may require thousands of function evaluations to explore the design space. There is a fundamental tradeoff between accuracy and speed that must be properly made in order to formulate analysis techniques that make numerical optimization a practical tool for use in conceptual and preliminary design.

Wing maximum lift is one of the areas that has been most difficult to model accurately for optimization. It has also been shown to be a very important parameter for choosing optimum wing planforms (Ref. 1), due to significant effects on aircraft noise, cost and performance. The tradeoff between high sweep for low drag at high Mach numbers and low sweep for good low speed performance and handling qualities is of fundamental importance. However, current methods used to evaluate wing maximum lift in conceptual design phases may not be sufficient to accurately model the effects of sweep.

The ideal model to determine wing maximum lift would be a 3-D Navier Stokes code with a grid density fine enough to capture the details of the stalling phenomenon on the wing. The computational power and time required for such a calculation, however, are not compatible with the simplicity requirements of an optimization model (in fact, they are not compatible with any of the computers available at the current time). Recalculating an entire 3-D Navier Stokes solution for each combination of the many design parameters chosen by the optimizer would be impractical if not impossible.

The starting point for the research described here is a wing model developed by Sean Wakayama at Stanford University (Ref. 1). This method uses a critical section analysis which compares local section lift coefficients, calculated from a Weissinger vortex lattice method, with estimates of 2-D maximum section lift coefficients based on empirical data. Flaps are simulated by increasing wing incidences in the Weissinger model, applying an increment in clmax due to flap deflection on the flapped portion of the wing, and increasing the assumed C_{lmax} due to induced camber on the sections near the flap edge. Finally, the maximum 2-D section lift coefficients are reduced by a factor of cosL as an empirical correction for the effect of sweep on the pressure distribution. The wing is then assumed to be at its maximum usable lift when any section C_l reaches some fraction of its local C_{lmax}. The assumption of a $\cos\Lambda$ variation in C_{lmax}, as opposed to the simple sweep theory (Ref. 3) assumption that C_{lmax} decreases with $\cos 2\Lambda$, is based on experimental observations (Ref 4). It has been observed that proper placement of fences and vortex generators on swept wings can yield C_{lmax} values that closely approach the 2-D unswept values (Refs. 5 and 6).

The critical section method may be justified for unswept wings, but its validity is suspect for wings with significant sweep for several reasons. The method assumes acquisition of 2-D C_{lmax} by placement of boundary layer control devices without actually specifying the location or geometry of such devices. Also, wing sweep changes the shape of the pressure distribution at fixed total lift, increasing the magnitude of the leading edge pressure peak. Most importantly, the existence of transverse pressure gradients along a swept wing induces boundary layer flow in the spanwise direction. This spanwise flow increases the length over which the boundary layer develops, resulting in a weaker boundary layer toward the wing tip. In certain codes used for high lift design at Boeing (Ref. 7) 3-D panel codes are coupled with 2-D boundary layer codes, thus capturing the correct 3-D pressure distribution, but still neglecting 3-D boundary layer effects. The effects of sweep on the transverse boundary layer development and the pressure peak are the primary areas of interest in the proposed research.

5.3 Research Objectives

Some optimization results using the current high lift analysis have indicated the possibility of problems due to inadequately modeling the impact of the various effects of sweep on C_{lmax}. The optimizer tends to give wings a few more degrees of overall sweep than existing designs for the same missions. It has also favored highly swept wing tips. Information on the variation of C_{lmax} with wing sweep is vital to performing planform optimization. Usable maximum lift is a major constraint that limits the sweep of a wing. With optimization results favoring larger sweeps than would be expected, a better assessment of the penalties sweep will impose on maximum lift capabilities is needed. To improve our knowledge of this constraint, we propose to continue a program of research on the effects of wing sweep on maximum usable section lift.

5.4 Research Program

Computational models are being developed and used to examine various inviscid and viscous phenomena that effect the maximum usable lift of wing designs, especially with regard to changes in wing sweep. The results of this study will be used to formulate an improved algorithm for optimization of wing planforms in preliminary design methods.

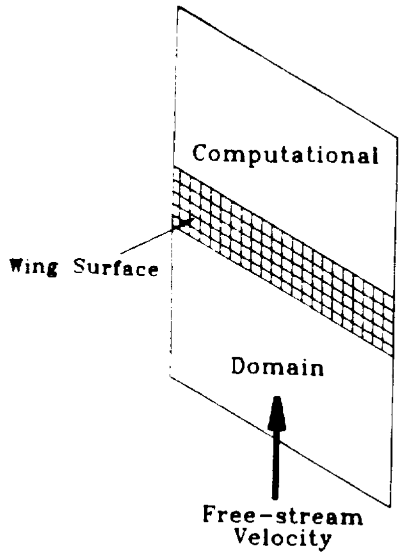
5.5 Research Activities Completed

During the previous year, studies have been initiated in two main areas. The first study focused on transverse boundary layer development, and the second on the effect of sweep on the inviscid pressure distribution for a wing of fixed total lift.

The transverse boundary layer development is studied using a 3-D, incompressible Navier Stokes code (INS-3D) to compute the flow properties along a segment of a swept wing. The wing segment runs all the way to the boundaries of the computational grid, eliminating the need for wing tips or roots. The idea behind modeling only a section of the wing is to use the available computational resources as efficiently as possible. By eliminating the wing tip and root, more grid points can be concentrated along the section of the wing where the boundary layer development is examined. Additionally, the grid is relatively easy to generate, allowing fast and efficient parametric studies varying sweep and airfoil sections. The key element of the model is then applying appropriate boundary conditions to simulate the actual flow over a section of a swept wing with a boundary layer control device.

The computational grid used for the boundary layer study is illustrated in figures 1 and 2. Standard boundary conditions are applied at the wake (surface 2) and at the [illegible]

and 5. This is much like the familiar infinite swept wing except that the remaining conditions are modified in this model so that flow properties do not remain constant in the spanwise direction. The boundary conditions at the wing surface and the inboard and outboard edges of the computational domain simulate the desired flow conditions. The goal of the model is to examine the spanwise development of the boundary layer along a swept wing outboard of some boundary layer control device; the boundary conditions at the inboard wing edge and the wing surface must be set to simulate this 'fresh' spanwise boundary layer condition. The boundary conditions at the outboard wing edge are then used to simulate a continuation of the wing surface. In the current model, the conditions at boundaries 6 and 7 are obtained by a zero-order extrapolation to the nearest computational plane. This is the same condition used for the infinite swept wing model. The wing surface, however, is used to provide the distinction. The first few rows of grid points out from the inboard edge of the wing segment are modeled as a slip surface, so the flow over this section is essentially an inviscid solution with no boundary layer. A no-slip boundary condition is [illegible]



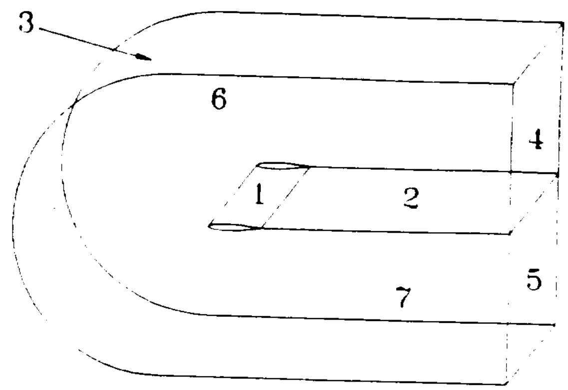


Figure 1 - Plan View of the Swept Wing Segment Model

Figure 2 - Computational Domain of the Swept Wing Segment Model

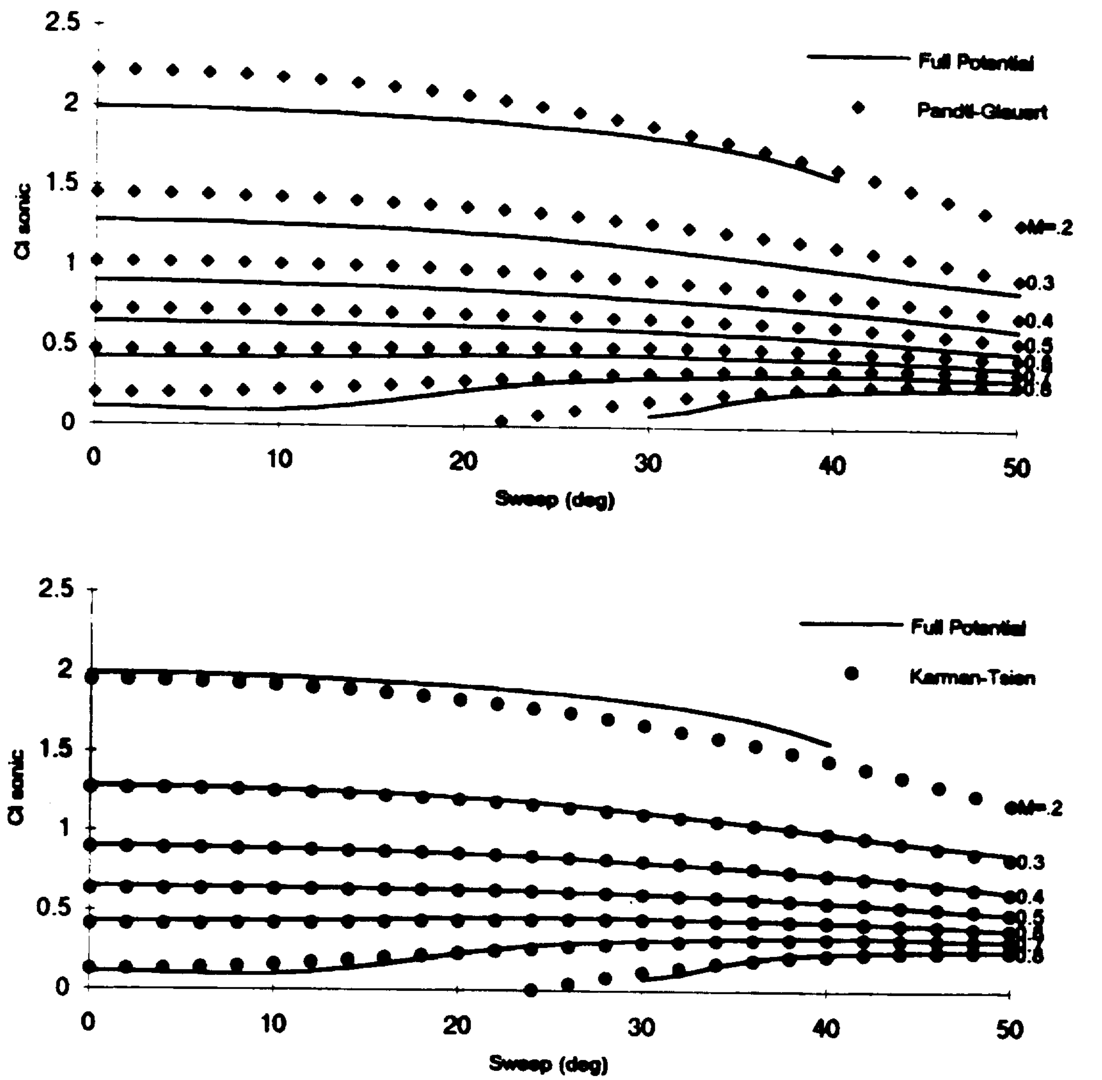


Figure 3 - Section Lift Coefficient for Sonic Conditions on a NACA0012 Airfoil
Comparsion of Full Potential Code and the Closed Form Solution Described in the Proposal

The pressure coefficient corresponding to sonic conditions is calculated as an isentropic function of the normal component of the freestream Mach number. This pressure coefficient is then converted to an 'incompressible' velocity increment using the Prandtl-Glauert or Karman-Tsien compressibility correction. The incompressible velocity distribution over the normal airfoil section is computed at two different angles of attack. Due to the linearity of the solution, the velocity distribution is then known for any angle of attack. This yields a closed form solution (quadratic in alpha) for the lowest angle of attack at which any point on the airfoil reaches sonic conditions, thus defining the maximum allowable angle of attack for the airfoil section. Then the maximum section lift limited by sonic conditions is calculated using this angle of attack and converted back to a compressible lift coefficient, again using the compressibility correction. The maximum lift based on freestream conditions is the computed maximum section lift times the square of the cosine of the sweep angle (which assumes that Jones' simple sweep theory applies for this inviscid model.)

The advantage of this model is that it provides a very simple and fast calculation of a constraint that is widely recognized as one of the primary factors limiting maximum usable lift. The disadvantage is that it relies on a panel code which requires use of a compressibility correction that is not necessarily appropriate with the large velocity increments. The model has been tested, however, against the results of a full potential code for a NACA0012 airfoil section over a wide range of sweep angles and freestream Mach numbers. The results, shown in figure 3, are quite encouraging. The errors are largest at low Mach numbers where the allowable angle of attack is quite large so that the velocity perturbation is much too high for the compressibility correction to be valid. Still, over most of the range of sweep and Mach number the method does a very good job of predicting sonic velocity. The algorithm developed here is substantially faster than the full potential solution for a couple of reasons. First, the panel code solution is faster than the full potential code for any given flight condition. Also, the full potential solution requires iteration to find the angle of attack for sonic conditions, while the new algorithm solves for the critical angle with a simple algebraic equation.

5.6 Research Activities Proposed for 1995-1996

During the coming year the models described above will be developed further to provide a base for the proposed wing design routine. Several issues concerning the current models remain open such as appropriate boundary conditions for the viscous case, and the effect of three-dimensionality on the inviscid pressure distribution.

The issue of appropriate boundary conditions will be the main focus of the work on the 3-D viscous wing section model, and there are several ideas on how to best model the flow conditions of interest. The key is to simulate the flow conditions on a finite swept wing with boundary layer control devices as closely as possible using a computational grid that contains only a piece of the wing. One idea is to model the entire wing section as a no-slip surface and use suction to remove the boundary layer at some spanwise location close to the inboard edge of the wing surface. Another is to somehow model an actual vortex generator or fence on the wing. This would also allow a detailed investigation of how these devices actually delay separation, and could lead to new insight on the best way to design and place such devices. A basic understanding of the physics of fences and vortex generators could be a very valuable tool in the future of wing design. Improving the model by finding better and more physically realistic boundary conditions will be the focus of this project in the coming months.

Further validation of the sonic velocity prediction routine will be carried out using 3-D codes with coupled boundary layer solutions to determine where the method is or is not useful. The routine will also be tested on a wide range of airfoil sections to determine the sensitivity of the model to different types of pressure distributions.

5.7 Research Activities Planned Beyond 1995-1996

The overall goal of this research is to design better wings faster. This could be accomplished in several ways depending on the results obtained in the coming months. A comparison of boundary layer properties on swept and unswept wings at similar conditions could be made to assess the applicability of using a 2-D boundary layer analysis on swept wings. This could be used to determine whether or to what extent the independence principle applies to the flow at conditions near maximum lift. Direct comparisons of velocity profiles, displacement thicknesses, momentum thicknesses and other boundary layer properties could be made between the 2-D wing, and along several cuts of a swept wing (such as normal to the sweep axis, aligned with the freestream velocity or aligned with the inviscid streamlines). These comparisons could be used to assess the validity of using 2-D maximum lift data, and could lead to appropriate corrections to 2-D data based on properties such as inviscid pressure distributions or sweep and other geometric parameters.

An attempt to build a Stratford like criteria for separation on a 3-D lifting surface could be made using data obtained from the research described above. In such a criteria, separation would be correlated with various properties of the flow such as pressure distribution, Reynolds number based on the distance the boundary layer has had to develop and the pressure gradient in the direction of local velocity. If such a criteria could be developed, it presents the best possibility for a fast but general method for preliminary assessment of maximum lift.

Another key factor in modern wing design that would benefit from this improved high lift modeling is the prediction and alleviation of flutter. As the preliminary design code is refined to better assess wing high lift characteristics, a flutter analysis could be included. This would make the wing design algorithm substantially more valuable in the early stages of aircraft synthesis and optimization.

Finally, further exploration of the 3-D flow over various wing configurations could allow examination of some unconventional wing designs. A code that is developed for 3-D analysis of a wide range of flight conditions and geometric parameters could be useful for exploring new ideas such as multi-element cruise airfoils or wing-body configurations. The lack of an analysis tool at the preliminary design stage that accounts for the complexity of 3-D flows makes these unconventional designs very difficult to assess with the current methods.

Project 5 References

[1] Wakayama, S., Lifting Surface Design Using Multidisciplinary Optimization, Ph.D. Dissertation, Stanford University Department of Aeronautics and Astronautics, Stanford, 1994.

[2] Smith, A.M.O., "High Lift Aerodynamics," AIAA Paper 74-939, Aug. 1974.

[3] Jones, R.T., Wing Theory, Princeton University Press, Princeton, NJ, 1990.

[4] Torenbeek, E., Synthesis of Subsonic Airplane Design, Delft University Press, Delft, Holland, 1982.

[5] Furlong, G. Chester, and McHugh, James G., "A Summary and Analysis of Low Speed Longitudinal Characteristics of Swept Wings at High Reynolds Number", NACA Report 1339, 1957.

[6] Letko, William, and Goodman, Alex, "Preliminary Wind Tunnel Investigation at Low Speed of Stability and Control Characteristics of Swept Back Wings", NACA TN 1046,1946.

[7] Brune, G.W., and McMasters, J.H., "Computational Aerodynamics Applied to High-Lift Systems", Applied Computational Aerodynamics, Vol. 125 of Progress in Astronautics and Aeronautics, AIAA, Washington, D.C., 1990.

III. PERSONNEL

The program, as described in the section, Detailed Program Description, will involve faculty, staff and graduate students as follows:

Faculty:

Prof. Brian Cantwell (PI)	15 % academic year, 30% summer
Prof. Steve Rock	5 % academic year, 10% summer
Prof. Sanjiva Lele	5 % academic year, 10% summer
Prof. Ilan Kroo	5 % academic year, 10% summer

Staff:

Senior Research Associate: Dr. Rabi Mehta	100 % calendar year
Research Associate: Dr. Zeki Celik	100 % calendar year
Research Assistants: 4 Ph.D. students	50 % AY/100 % summer

Additionally, the following support staff will be involved:

Sci-Eng. Associate	10 % calendar year

Throughout the conduct of this research and training activity close coordination will take place between the research personnel at Stanford and the research personnel and technical management staff at Ames.

IV. FUNDING

The funding requested for the one-year period, October 1995 - September 1996 is given in the attached Estimated Cost Breakdown. As the University's contribution to the administration of the Institute, indirect costs on Professor Cantwell's administrative salary charges and administrative and secretarial support are waived.

ESTIMATED COST BREAKDOWN - JIAA 95-96 ADMINISTRATIVE BUDGET

GRANT NUMBER NCC 2-55
PROPOSAL NUMBER - AERO 42-95
DURATION - 12 MONTHS BEGINNING OCTOBER 1, 1995 TO SEPTEMBER 30, 1996

	ACADEMIC	SUMMER	CALENDAR	COST
A. SENIOR PERSONNEL				
B. Cantwell, Prof.	10.00%	20.00%		13,711
B. OTHER STAFF				
C. Edwards, Secy.			20.00%	6,634
TOTAL SALARIES and WAGES (A+B)				20,345
C. FRINGE BENEFITS (applied to TOTAL SALARIES AND WAGES)				
Faculty - 26.2% through 8/31/96, 26.2% through 8/31/97				3,592
Staff - 26.2% through 8/31/96, 26.2% through 8/31/97				1,738
TOTAL SALARIES, WAGES and FRINGE BENEFITS (A+B+C)				25,675
D. SUB-TOTAL DIRECT COSTS (A+B+C)				25,675
E. MODIFIED TOTAL DIRECT COSTS (D)				25,675
F. UNIVERSITY INDIRECT COSTS (Waived)				
0.00% through 8/31/95, 0.00% through 8/31/96				
G. ANNUAL AMOUNT REQUESTED (D+F)				25,675
TOTAL PROJECT COST				25,675

TOTAL ESTIMATED JIAA COST FOR 95-96	
Admin Budget	25,675
On-campus Budget	394,405
Off-campus Budget	141,144
ESTIMATED TOTAL JIAA 95-96 BUDGET	561,224

ESTIMATED COST BREAKDOWN - JIAA 95-96 ON-CAMPUS BUDGET

GRANT NUMBER NCC 2-55
PROPOSAL NUMBER - AERO 42-95
DURATION - 12 MONTHS BEGINNING OCTOBER 1, 1995 TO SEPTEMBER 30, 1996

	ACADEMIC	SUMMER	CALENDAR	COST
A. SENIOR PERSONNEL				
B. Cantwell, Prof.	5.00%	10.00%		6,855
S. Rock, Assoc. Prof.	5.00%	10.00%		6,413
I. Kroo, Assoc. Prof.	5.00%	10.00%		6,457
S. Lele, Assist. Prof.	5.00%	10.00%		5,772
B. STUDENTS				
STUDENT RESEARCH ASSISTANT	50.00%	50.00%		17,400
STUDENT RESEARCH ASSISTANT	50.00%	50.00%		17,400
STUDENT RESEARCH ASSISTANT	50.00%	50.00%		17,400
STUDENT RESEARCH ASSISTANT	50.00%	50.00%		17,400
C. OTHER STAFF				
Z. Celik, Res. Assoc.			100.00%	60,501
V. Matte, Sci.-Eng. Assoc.			10.00%	6,272
TOTAL SALARIES and WAGES (A+B+C)				161,870
D. FRINGE BENEFITS (applied to TOTAL SALARIES AND WAGES)				
Faculty - 26.04%through 8/31/96, 26.04% through 8/31/97				6,680
Staff - 26.04%through 8/31/96, 26.04% through 8/31/97				17,495
Graduate - 26.04%through 8/31/96, 26.04% through 8/31/97				18,235
TOTAL SALARIES, WAGES and FRINGE BENEFITS (A+B+C+D)				204,280
E. OTHER COSTS				18,000
Univ. services, communications, xerox, travel,pub.				
F. COSTS NOT SUBJECT TO INDIRECT COSTS				32,000
Capital Equipment				
Project 1 - control system modification - $8,500				
Project 2 - simulation display system-$5,000				
Project 3 - velocity measuring system fabrication - $16,000				
Project 3 - pressure measuring system - $2,500				
G. SUB-TOTAL DIRECT COSTS (A+B+C+D+E+F)				254,280
H. MODIFIED TOTAL DIRECT COSTS (G-F)				222,280
I. UNIVERSITY INDIRECT COSTS ON MTDC (H)				140,125
63.04% through 8/31/96, 63.04% through 8/31/97				
J. ANNUAL AMOUNT REQUESTED (G+I)				394,405
TOTAL PROJECT COST				394,405

ESTIMATED COST BREAKDOWN - JIAA 95-96 OFF-CAMPUS BUDGET

GRANT NUMBER NCC 2-55
PROPOSAL NUMBER - AERO 42-95
DURATION - 12 MONTHS BEGINNING OCTOBER 1, 1995 TO SEPTEMBER 30, 1996

	ACADEMIC	SUMMER	CALENDAR	COST
A. SENIOR PERSONNEL				
B. Cantwell, Prof.				
B. OTHER STAFF				
R. Mehta, Senior Res. Assoc.			100.00%	81,363
TOTAL SALARIES and WAGES (A+B)				81,363
C. FRINGE BENEFITS (applied to TOTAL SALARIES AND WAGES)				21,317
Staff - 26.2% through 8/31/96, 26.2% through 8/31/97				
TOTAL SALARIES, WAGES and FRINGE BENEFITS (A+B+C)				102,680
D. SUB-TOTAL DIRECT COSTS (A+B+C)				102,680
E. MODIFIED TOTAL DIRECT COSTS (D)				102,680
F. UNIVERSITY INDIRECT COSTS ON MTDC (H)				38,464
37.46% through 8/31/96, 37.46% through 8/31/97				
G. ANNUAL AMOUNT REQUESTED (F)				141,144
TOTAL PROJECT COST				141,144