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AERONAUTICAL ENGINEERING

A CONTINUING BIBLIOGRAPHY WITH INDEXES

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Two indexes—subject and author are included.

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Typical Report Citation and Abstract

ACCESSION NUMBER ↓

DOCUMENT ID NUMBER → **19960021053;** 96N24540 NASA Langley Research Center, Hampton, VA USA. ← CORPORATE SOURCE

TITLE → **An Extended Compact Tension Specimen for Fatigue Crack Propagation and Fracture**

AUTHORS → Piascik, R. S., NASA Langley Research Center, USA; Newman, J. C., Jr., NASA Langley Research Center, USA; ← AUTHORS' AFFILIATION

PUBLICATION DATE → Mar. 1996, pp. 16; In English

CONTRACTS/GRANTS → Contract(s)/Grant(s): RTOP 538–02–10–01

REPORT NO.(S) → Report No.(s): NASA–TM–110243; NAS 1.15:110243; No Copyright; Avail: CASI A03, Hardcopy; A01, Microfiche ← AVAILABILITY AND PRICE CODE

ABSTRACT → An extended compact tension specimen, EC(T) has been developed for fatigue and fracture testing. Documented herein are stress-intensity factor and compliance expressions for the EC(T) specimen.

ABSTRACT AUTHOR → Author

SUBJECT TERMS → *Crack Propagation; Stress Intensity Factors; Fatigue (Materials)*

AERONAUTICAL ENGINEERING

A Continuing Bibliography (Suppl. 333)

AUGUST 1996

01
AERONAUTICS

19960017087; 96N22683 Johns Hopkins Univ., Baltimore, MD USA

Electromagnetic stressing/optical detection technique for detection of disbonds in aging aircraft

Mayton, Donna Jean, Johns Hopkins Univ., USA; 1994, pp. 84; Also announced as 19960017087; No Copyright; Avail: Univ. Microfilms Order No. DA9429546, Hardcopy, Microfiche

A novel electromagnetic stressing/optical detection method has been developed in response to the need for better nondestructive evaluation techniques for the detection of disbonds in aging aircraft lap joints. This technique uses magnetic pressure to pull the top surface of a thin conductive bonded sheet and senses the out-of-plane displacement of the surface with an optical lever fiber bundle detector. This method of inspection has the advantages of being noncontacting, relatively cheap, and because it pulls on the top surface, is a promising candidate for the detection of 'kissing' disbonds -- a condition in which there is no material missing from the joint, but the adhesive has failed. A series of three models were developed and implemented to fully simulate the technique from the driving circuitry to the measured response of the sample. The driving circuit was computer modeled and the current through the electromagnetic coil as a function of time was predicted. An analytical pressure model was developed to predict pressure on the sample as a function of time for a given current input. The predicted pressure was used as the driving function in a finite element structural model which predicted displacement of the sample surface. Laboratory experiments were conducted on simple bonded and unbonded samples, and the two cases exhibited large differences in amplitude, resonant frequency, and damping. Tests were also performed on a simulated aircraft lap joint and the mechanical response of bonded and unbonded regions were distinguishable in the frequency domain. The series of models was used successfully to predict the displacement of the bonded and unbonded samples and compared to experimental results using a prototype system. The close correspondence between predicted and measured results indicates that the models are a useful tool in the design and implementation of this technique. Although the development of this technique was prompted by the failure of aging aircraft lap joints, it may be applied to any thin, conductive bonded structure.

Dissert. Abstr.

Adhesive Bonding; Bonded Joints; Lap Joints; Nondestructive Tests; Optical Measurement; Magnetic Effects; Debonding (Materials)

19960017278; 96N22869 Aircraft Research Association Ltd., Bedford, UK

Collaboration in Aerospace: One European's Perspective

Green, J. E., Aircraft Research Association Ltd., UK; May 1995, pp. 26; In English; PICAST2-AAC6, 20-23 Mar. 1995, Melbourne, Australia; Also announced as 19960017278

Report No.(s): ARA-TM-414; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This lecture reviews European experience of collaboration in aerospace over the past forty years with particular reference to collaboration in aircraft development and production. It ranges from national and international collaboration in the provision of major wind tunnels, through intergovernmental collaboration on military aircraft and inter-company collaboration on civil aircraft to collaboration between aeronautical research organizations and also between learned societies. Finally, the general lessons to be drawn from this experience are outlined and their relevance to the developing aerospace industries of the Pacific Region is discussed.

Author

Aerospace Industry; Aircraft Design; International Cooperation; Europe; Research

19960017307; 96N22898 Defence Science and Technology Organisation, Canberra, Australia

F-111 Aircraft Fatigue Data Analysis System (AFDAS) in Service Development *Progress Report, No. 1*

Walker, K., Defence Science and Technology Organisation, Australia; Jul. 01, 1995, pp. 84; Also announced as 19960017307

Report No.(s): AD-A301201; DSTO-TR-05; DODA-AR-008-992; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

The Aircraft Fatigue Data Analysis System (AFDAS) is a twelve channel, strain based fatigue data collection and analysis system. The RAAF have recognized that AFDAS offers significant potential improvements over fatigue meters and parametric based systems for the purpose of fatigue monitoring and structural integrity management. AFDAS has therefore been implemented on a number of aircraft types, including the F-111. The system does, however, require further development and refinement. AMRL was requested by the RAAF to provide assistance and advice on the F-111 AFDAS installation. This report details the progress made so far on the current AMRL F-111 AFDAS support activity. The activity has included aspects which are unique to the F-111 and also some aspects which have a general applicability to the AFDAS installation on other aircraft types. Significant progress has been made in the following areas: (1) establishing why the strain sensor locations were chosen and how they relate to other locations of importance for fatigue or structural life monitoring reasons; (2) eliminating operational errors and difficulties which decrease the integrity of the data; and (3) developing new data screening procedures which check the data for inconsistencies and invalid results.
DTIC
F-111 Aircraft; Structural Failure; Fatigue (Materials); Strain Measurement; Strain Distribution; Computer Programs; Data Reduction

19960017732; 96N23263 NASA Lewis Research Center, Cleveland, OH USA
Research and Technology, 1992
1992, pp. 170; In English; Also announced as 19960017732 Report No.(s): NASA-TM-105924; E-7425; NAS 1.15: 105924; No Copyright; Avail: CASI; A08, Hardcopy; A02, Microfiche

This report selectively summarizes the NASA Lewis Research Center's research and technology accomplishments for the fiscal year 1992. It comprises approximately 200 short articles submitted by the staff members of the technical directorates. The report is organized into six major sections: Aeronautics, Aerospace Technology, Space Flight Systems, Space Station Freedom, Engineering and Computational Support, and Lewis Research Academy. A table of contents by subject has been developed to assist the reader in finding articles of special interest. This report is not intended to be a comprehensive summary of all the research and technology work done over the past fiscal year. Most of the work is reported in Lewis-published technical reports, journal articles, and presentations prepared by Lewis staff or by contractors. In addition, university grants have enabled faculty members and graduate students to engage in sponsored research that is reported at technical meetings or in journal articles. For each article in this report a Lewis contact person has been identified, and where possible, a reference document is listed so that additional information can be easily obtained. The diversity of topics attests to the breadth of research and technology being pursued and to the skill mix of the staff that makes it possible.
Author
Research and Development; Aeronautical Engineering; Aerospace Engineering; Space Station Freedom; Spacecraft Instruments; Flight Instruments; NASA Programs

02
AERODYNAMICS

Includes aerodynamics of bodies, combinations, wings, rotors, and control surfaces; and internal flow in ducts and turbomachinery.

19960015583; 96N22391 Old Dominion Univ., Norfolk, VA USA
Simulation of active control of asymmetric flows around slender pointed forebodies
SharafEl-Din, Hazem Hafez, Old Dominion Univ., USA; Jan. 01, 1994, pp. 236; In English; Also announced as 19960015583; No Copyright; Avail: Univ. Microfilms Order No. DA9514746, Hardcopy, Microfiche

At high angles of attack, the flowfield over slender forebodies becomes asymmetric with substantial side force, which may exceed the available control capability. The unsteady compressible Navier-Stokes equations are used to investigate the effectiveness of different active control methods to alleviate and possibly eliminate the flow asymmetry and the subsequent side force. Although the research work focuses on active control methods, a passive control method has been investigated. The implicit, Roe flux-difference splitting, finite volume scheme is used for the numerical computations. Both locally-conical and three-dimensional solutions of the Navier-Stokes equations are obtained. The asymmetric flow over five-degree semi-apex angle cone is used as a reference case to which the different control methods are applied and compared. For the passive control method, the side-strakes control is investigated. The parametric study includes the control effectiveness of the strake span length. For the active control methods, flow injection in the normal and tangential directions to the body surface has been investigated. Both uniform and pressure-sensitive mass flow injection are applied, and the effects of mass flow rate, injection angle and injection length have also been studied. Injection, with a parabolic profile, is applied from the cone sides tangent to its surface. Surface-heating, where temperature of the cone surface is increased, is also investigated. The effectiveness of a hybrid method of flow control, which combines injection with surface heating, has been studied. The cone spinning and rotary oscillation around its axis are applied as an active control method. The computational applications include the effects of uniform spinning rates and periodic rotary oscillations at different amplitudes and frequencies on the flow asymmetry.
Dissert. Abstr.

Flow Distribution; Active Control; Forebodies; Compressible Flow; Unsteady Flow; Navier-Stokes Equation; Flux Difference Splitting; Finite Volume Method; Computational Fluid Dynamics

19960015598; 96N22406 Toronto Univ., Ontario Canada
A study of the fortified Navier-Stokes approach for viscous airfoil computations
Bergeron, Maurice Denis, Toronto Univ., Canada; Jan. 01, 1994, pp. 153; In English; Also announced as 19960015598; ISBN 0-315-92601-5; No Copyright; Avail: Univ. Microfilms Order No. DANN92601, Hardcopy, Microfiche

In general, the application of current computational methods to attached viscous airfoil flow results in lift predictions to within 3 percent and drag predictions to within 5 percent of experimental values. For separated flows, the errors can be considerably larger. These errors are a combination of physical-model errors and numerical errors. Very fine grids are required to reduce numerical errors in drag to acceptable levels which drastically increases the computer resources required, especially in three-dimensional computations. Therefore, there is considerable motivation to examine numerical methods which might reduce the number of grid points required to achieve a given level of accuracy in drag. One such method is the fortified Navier-Stokes (FNS) approach. A global Navier-Stokes solution computed on a coarse grid is 'fortified' with a boundary layer solution which is obtained efficiently on a fine grid superimposed near the body surface. Other authors have demonstrated that FNS increases the efficiency and accuracy of a Navier-Stokes solver when it is applied to flat plate flows. In this work, FNS is extended to viscous airfoil computations. Factors affecting the fortification process are investigated, suitable blending functions are developed and an overall fortification strategy is implemented with the diagonal form of an implicit approximate-factorization algorithm. Computational results are presented for five subsonic and transonic viscous airfoil cases which display a variety of attached and separated flow conditions. In all cases, the FNS approach is successful in reducing numerical errors in drag, particularly for cases dominated by friction drag. However, due to constraints on the amount of forcing which can be applied, the FNS approach is only partially effective at improving the accuracy of the pressure field relative to the coarse-mesh Navier-Stokes solution. Therefore, the approach is most appropriate for attached and mildly separated flows.
Dissert. Abstr.
Navier-Stokes Equation; Viscous Flow; Computational Grids; Airfoils; Friction Drag

19960015653; 96N22461 Notre Dame Univ., IN USA
Numerical simulations of steady and unsteady oblique detonation phenomena with application to propulsion
Matthew, John, Notre Dame Univ., USA; Jan. 01, 1994, pp. 249; In English; Also announced as 19960015653; No Copyright; Avail: Univ. Microfilms Order No. DA9516632, Hardcopy, Microfiche

Oblique detonations and their propulsive applications were studied by numerically integrating the two-dimensional, reactive Euler equations in a generalized, curvilinear coordinate system. The integration was accomplished using the Roe scheme combined with fractional stepping: nonlinear flux limiting was used to prevent unphysical solution oscillations near discontinuities. The method was verified on one- and two-dimensional flows with exact solutions, and its ability to correctly predict one-dimensional detonation instability was demonstrated. Unsteady phenomena were considered in a study of straight oblique detonations attached to curved walls. Using the exact, steady oblique detonation solution as an initial condition, the numerical simulation predicted both steady and unsteady oblique detonation solutions when a detonation parameter known as the normal overdrive was varied. The normal overdrive value at which the oblique detonation transitioned from steady to unsteady behavior was slightly higher than the corresponding value for a one-dimensional detonation. An oblique detonation application was considered to determine the steady propagation speed of an idealized ram accelerator. Propagation speeds were found which gave rise to shocks of such strength as to induce a reaction zone to be in a region which allowed the combustion induced thrust to balance the wave drag. For fixed heat release greater than a critical value, two steady propagation speeds were predicted. The solution at the higher Mach number was stable to static perturbations while the solution at the lower Mach number was unstable. The Chapman-Jouguet velocity in the direction of projectile propagation was found to be relevant only as an overly conservative lower bound for possible flight speeds. In the far-field the detonation wave angle was found to be that of a Chapman-Jouguet oblique detonation.
Dissert. Abstr.
Detonation Waves; Perturbation; Differential Equations; Far Fields; Heat Transfer; Stability; Thrust Control

19960015728; 96N21512 Pennsylvania State Univ., University Park, PA USA
Numerical Study of Steady and Oscillatory Flow Structures in an Axisymmetric Supersonic Inlet
Oh, Jong Yun, Pennsylvania State Univ., USA; Jan. 01, 1994, pp. 183; In English; Also announced as 19960015728; No Copyright; Avail: Univ. Microfilms Order No. DA9504264, Hardcopy, Microfiche

Steady and oscillatory flowfields of an axisymmetric mixed-compression supersonic inlet have been numerically studied by solving the compressible Euler and Navier-Stokes equations. Turbulence in the boundary layers is modeled using the algebraic model of Baldwin and Lomax for steady-state flow calculations and the two-layer model of Rodi for unsteady flow calculations. The governing equations are dis-

cretized with the finite-volume method, and the discretized equations are solved by the LU implicit scheme for the steady-state flows and by the four-stage Runge-Kutta explicit scheme for the unsteady flows. The upwind TVD scheme of Harten and Yee is adopted to capture shocks without numerical oscillations. Steady-state axisymmetric flowfields are investigated under various engine-operating conditions. The inviscid-flow results indicate that a vortex sheet, extending downstream from the shock intersection point, creates non-uniform flowfields in the radial direction at downstream of the terminal shock because of different compression histories of the flow just upstream of the shock. The viscous-flow results show that the boundary layers are thickened and separation bubbles are formed across the terminal shock due to shock/boundary-layer interactions. Unsteady axisymmetric flowfields are also analyzed to study the shock/acoustic-wave interactions. As the normal shock oscillates because of the fluctuating pressure imposed at the exit, the shock intersection point also fluctuates which can add one more mode of oscillation to the flowfields, especially to the velocity field. Finally, steady-state three-dimensional flowfields with finite angles of attack are investigated. When the angle of attack is not zero, the terminal shock is skewed due to different compression histories and the degree of skewness becomes greater with the increased angle of attack.
Dissert. Abstr.
Steady Flow; Oscillating Flow; Supersonic Inlets; Axisymmetric Bodies; Inlet Flow; Inviscid Flow; Mathematical Models; Viscous Flow; Unsteady Flow; Sound Waves; Compressible Flow; Navier-Stokes Equation; Euler Equations of Motion

19960015740; 96N21524 Pennsylvania State Univ., University Park, PA USA
The Interaction Between Crossing-Shock Waves and a Turbulent Boundary Layer
Garrison, Timothy Jon, Pennsylvania State Univ., USA; 1994, pp. 297; In English; Also announced as 19960015740; No Copyright; Avail: Univ. Microfilms Order No. DA9504213, Hardcopy, Microfiche

An experimental study has been performed to investigate the flowfield structure created by two crossing oblique shock waves interacting with a turbulent boundary layer. Such an interaction is of practical importance in the design of high-speed inlets. The interaction is created by a test model, consisting of two sharp fins mounted at symmetric angles-of-attack to a flat plate, placed in a supersonic freestream flow. The interaction structure is examined for a range of symmetric fin angles from 7 to 15 degrees at freestream Mach numbers of 3 and 3.85. This parametric study produces a broad range of interaction strengths, resulting in barely separated to strongly separated flows. A variety of experimental techniques are used to study the crossing-shock interaction. Planar laser scattering visualization and kerosene lampblack surface flow visualization are used to reveal the flowfield structure of these interactions. The visualizations show that the flowfield is comprised of a complex, irregular crossing of the two individual shock/boundary layer interactions created by each fin. Quantitative surface data are obtained through wall static pressure and skin friction coefficient measurements. The skin friction measurements, made using a laser interferometer skin friction meter, reveal regions with zero shear as well as regions of extremely high shear. Quantitative flowfield data, obtained through five-hole probe surveys, show the presence of a large, low-Mach-number, low-total-pressure region comprised of the separated incoming boundary layer fluid. The combined experimental dataset is used to formulate a detailed flowfield model of the crossing-shock interaction. The structure of these interactions is found to be similar over a broad range of interaction strengths and is dominated by a large, separated, viscous flow region which forms on the interaction plane of symmetry. Additionally, the experimental results are compared with two available computational solutions to assess the capabilities of current CFD programs to model such complex interactions.
Dissert. Abstr.
Turbulent Boundary Layer; Shock Wave Interaction; Flow Distribution; Oblique Shock Waves; Supersonic Flow; Free Flow; Separated Flow; Viscous Flow; Angle of Attack; Fins; Flow Visualization

19960015741; 96N21525 Pennsylvania State Univ., University Park, PA USA
An Experimental Study of Fluctuating Pressure Loads Beneath Swept Shock Wave/Boundary Layer Interactions
Garg, Sanjay, Pennsylvania State Univ., USA; 1994, pp. 156; In English; Also announced as 19960015741; No Copyright; Avail: Univ. Microfilms Order No. DA9504210, Hardcopy, Microfiche

An experimental research program providing basic knowledge and establishing a database on the fluctuating pressure loads produced on aerodynamic surfaces beneath three-dimensional shock wave/boundary layer interactions is described. Such loads constitute a fundamental problem of critical concern to future supersonic and hypersonic flight vehicles. A turbulent boundary layer on a flat plate is subjected to interactions with swept planar shock waves generated by sharp fins. Fin angles from 10 deg to 20 deg at freestream Mach numbers of 3 and 4 produce a variety of interaction strengths from weak to very strong. Miniature pressure transducers flush-mounted in the flat plate have been used to measure interaction-induced wall pressure fluctuations. The distributions of properties of the pressure fluctuations, such as their rms level, amplitude distribution and power spectra, are also determined. Measurements have been made for the first time in the aft regions of these interactions, revealing fluctuating pressure levels as high as 155 dB, which places them in the category of significant aeroacoustic load

generators. The fluctuations near the foot of the fin are dominated by low frequency (0-5 kHz) components, and are caused by a previously unrecognized random motion of the primary attachment line. This phenomenon is probably intimately linked to the unsteadiness of the separation shock at the start of the interaction. The characteristics of the pressure fluctuations are explained in light of the features of the interaction flowfield. In particular, physical mechanisms responsible for the generation of high levels of surface pressure fluctuations are proposed based on the results of the study. The unsteadiness of the flowfield of the surface is also examined via a novel, non-intrusive optical technique. Results show that the entire shock structure generated by the interaction undergoes relatively low-frequency oscillations.
Dissert. Abstr.
Pressure; Pressure Distribution; Shock Wave Interaction; Three Dimensional Boundary Layer; Turbulent Boundary Layer; Control Surfaces; Interactional Aerodynamics

19960015774; 96N21558 California Univ., Los Angeles, CA USA
The control of 2-D wakes/mixing layers via suction at the trailing edge of a splitter plate
Leu, Tzong-Shyng, California Univ., USA; Jan. 01, 1994, pp. 166; In English; Also announced as 19960015774; No Copyright; Avail: Univ. Microfilms Order No. DA9502028, Hardcopy, Microfiche

Wake and mixing layer flows controlled by suction at the trailing edge of a splitter are investigated both experimentally and theoretically. The experiments are performed in a shear layer water channel. In the wake flow controlled by a suction, a threshold suction speed is found to be equal to one half of freestream velocity. When the suction speed is below the threshold, the wake flow is unstable. When the suction speed is above the threshold, the wake becomes stable and no vortex shedding exists. The stability analysis is performed in order to understand the physical mechanism in this wake control problem. The results show the vortex shedding frequency can be well predicted by the velocity profile immediately behind the solid boundary. Furthermore, the threshold suction speed can be reasonably predicted by the global linear stability analysis in a weakly non-parallel flow. The observed changes of the global flow field are related to an absolute instability region near the trailing edge. The non-parallel flow effects are responsible for this global instability breakdown.
Dissert. Abstr.
Mixing Layers (Fluids); Flow Distribution; Velocity Distribution; Vortex Shedding; Trailing Edges; Suction; Wakes; Shear Layers

19960015784; 96N21568 Rensselaer Polytechnic Inst., Troy, NY USA
Lifting surface design using trailing edge devices, marginal aeroelastic stability and feedback control
Tseng, Stephen Pai-Yu, Rensselaer Polytechnic Inst., USA; Jan. 01, 1994, pp. 167; In English; Also announced as 19960015784; No Copyright; Avail: Univ. Microfilms Order No. DA9434604, Hardcopy, Microfiche

Considerable promise is seen in the application of strain actuator technology as Smart Materials' for fulfilling control functions of various kinds in aircraft. However, the strain and force capability of these actuators are known to impose limitations. to overcome these limitations, the aeromechanical design of fixed and rotary wings from a control viewpoint have been investigated in this study. Three design analyses have been conducted using a simple model with a trailing-edge hinged flap or tab representing the strain actuators, including a fixed wing-flap-tab with constant forward speed, a wing-flap with pulsating velocity superimposed on constant forward speed, and a helicopter rotor blade in hovering flight. A generalized two-dimensional, time-domain, finite-state, Theodorsen-Greenberg unsteady aerodynamic model of the wing-flap in a pulsating velocity field has been extended and applied in these analyses to calculate the aerodynamic loads on the airfoils with a trailing-edge hinged control surface. State space representations of those lifting surface aerodynamics have been derived in time-varying format. Single- and multi-objective design optimizations have then been conducted to assess the capabilities of possible designs which maximize the amplification of the rotations and minimize force required of the control surfaces. As these optimal designs optimizations have ten been conducted to assess the capabilities of possible designs which maximize the amplification of the rotations and minimize the force required of the control surfaces. As these optimal designs tend to approach aeroelastic instability, Linear Quadratic Gaussian (LQG) feedback controller design has been applied to assure the stability of resulting motions and improve the output performance. A discrete periodic time-varying LQG controller has also been developed to stabilize the systems in the presence of pulsating velocity effects. A dynamic inflow model has been linearized and included in the design analyses of the helicopter rotor blade to represent the three-dimensional transient behavior of the complete rotor. The results of this research have shown the feasibility of applying properly designed trailing-edge control surfaces which make use of strain actuation concepts to both fixed and rotary wings. Further analyses and experiments are expected to be required to develop this concept of Smart Structure' control surfaces to the point where they are ready for practical aircraft applications.
Dissert. Abstr.
Aircraft Design; Lifting Bodies; Tabs (Control Surfaces); Feedback Control; Aerodynamic Stability; Aeroelasticity; Linear Quadratic Gaussian Control; Controllers; Design Analysis; Trailing Edge Flaps

19960015881; 96N21659 Instituto Nacional de Pesaciais Espaciais, Sao Jose dos Campos Brazil

Determination of Coefficeint of Drag of the Sphere-Cone Configurations by Method of Belotserkovskii ***Determinacao do Coefficiente de Arrasto de Configuracoes Esfera-Cone Pelo Metodo de Belotserkovskii***

daSilva, Fernando Cachucho Canesin, Instituto Nacional de Pesaciais Espaciais, Brazil; Feb. 1995, pp. 123; In Portugese; Also announced as 19960015881

Report No.(s): INPE-5679-TDI/566; No Copyright; Avail: CASI; A06, Hardcopy; A02, Microfiche

A computational code based on the first aproximation of the method of Belotserkovskii has been developed to calculate the coefficient of drag of the sphere-cone configurations. In this method, the region of integration is subdivided in several strips using curves which shapes are determined by the profile of the region. The basic objective is to integrate, over this strips, the governing partial differential equations with respect to one of the independent variables. Following the integrands are approximated to m order polynomials in this variable. After integrating these polynomials an approximated system of ordinary differential equations is obtained as a function of the remaining independent variable. The results obtained for the pressure and drag coefficients are compared with the experimental data obtained in wind tunnels.

Author

Drag Coefficients; Partial Differential Equations; Cones; Aerodynamic Drag

19960015882; 96N21660 High Technology Corp., Hampton, VA USA

Large-Eddy Simulation of Turbulent Wall-Pressure Fluctuations

Singer, Bart A., High Technology Corp., USA; Feb. 1996, pp. 38; In English; Also announced as 19960015882

Contract(s)/Grant(s): NAS1-20059; RTOP 505-59-50-02

Report No.(s): NASA-CR-198276; NAS 1.26:198276; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Large-eddy simulations of a turbulent boundary layer with Reynolds number based on displacement thickness equal to 3500 were performed with two grid resolutions. The computations were continued for sufficient time to obtain frequency spectra with resolved frequencies that correspond to the most important structural frequencies on an aircraft fuselage. The turbulent stresses were adequately resolved with both resolutions. Detailed quantitative analysis of a variety of statistical quantities associated with the wall-pressure fluctuations revealed similar behavior for both simulations. The primary differences were associated with the lack of resolution of the high-frequency data in the coarse-grid calculation and the increased jitter (due to the lack of multiple realizations for averaging purposes) in the fine-grid calculation. A new curve fit was introduced to represent the spanwise coherence of the cross-spectral density.

Author

Turbulent Boundary Layer; Wall Pressure; Pressure Oscillations; Aircraft Noise; Noise Reduction; Structural Vibration; Computational Grids; Quantitative Analysis; Runge-Kutta Method; Crank-Nicholson Method

19960015887; 96N21664 NASA Ames Research Center, Moffett Field, CA USA

A Computational and Experimental Study of Nonlinear Aspects of Induced Drag

Smith, Stephen C., NASA Ames Research Center, USA; Feb. 1996, pp. 140; In English; Also announced as 19960015887

Contract(s)/Grant(s): RTOP 505-59-10

Report No.(s): NASA-TP-3598; A-960804; NAS 1.60:3598; No Copyright; Avail: CASI; A07, Hardcopy; A02, Microfiche

Despite the 80-year history of classical wing theory, considerable research has recently been directed toward planform and wake effects on induced drag. Nonlinear interactions between the trailing wake and the wing offer the possibility of reducing drag. The nonlinear effect of compressibility on induced drag characteristics may also influence wing design. This thesis deals with the prediction of these nonlinear aspects of induced drag and ways to exploit them. A potential benefit of only a few percent of the drag represents a large fuel savings for the world's commercial transport fleet. Computational methods must be applied carefully to obtain accurate induced drag predictions. Trefftz-plane drag integration is far more reliable than surface pressure integration, but is very sensitive to the accuracy of the force-free wake model. The practical use of Trefftz plane drag integration was extended to transonic flow with the Tranair full-potential code. The induced drag characteristics of a typical transport wing were studied with Tranair, a full-potential method, and A502, a high-order linear panel method to investigate changes in lift distribution and span efficiency due to compressibility. Modeling the force-free wake is a nonlinear problem, even when the flow governing equation is linear. A novel method was developed for computing the force-free wake shape. This hybrid wake-relaxation scheme couples the well-behaved nature of the discrete vortex wake with viscous-core modeling and the high-accuracy velocity prediction of the high-order panel method. The hybrid scheme produced converged wake shapes that allowed accurate Trefftz-plane integration. An unusual split-tip wing concept was studied for exploiting nonlinear wake interaction to reduced induced drag. This design exhibits significant nonlinear interactions between the wing and wake that produced a 12% reduction in induced drag compared to an equivalent elliptical wing at a lift coefficient of 0.7. The performance of the split-tip wing was also investigated by wing tunnel experiments. Induced drag was determined from force measurements by subtracting the estimated

viscous drag, and from an analytical drag-decomposition method using a wake survey. The experimental results confirm the computational prediction.
Author
Induced Drag; Nonlinearity; Wakes; Wing Tips; Interactional Aerodynamics

19960015890; 96N21666 NASA Langley Research Center, Hampton, VA USA
Effects of Winglets on the Drag of a Low-Aspect-Ratio Configuration
Smith, Leigh Ann, NASA Langley Research Center, USA; Campbell, Richard L., NASA Langley Research Center, USA; Feb. 1996, pp. 66; In English; Also announced as 19960015890
Contract(s)/Grant(s): RTOP 505-59-10-30
Report No.(s): NASA-TP-3563; L-17456l; NAS 1.60:3563; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

A wind-tunnel investigation has been performed to determine the effect of winglets on the induced drag of a low-aspect-ratio wing configuration at Mach numbers between 0.30 and 0.85 and a nominal angle-of-attack range from -2 deg to 20 deg. Results of the tests at the cruise lift coefficient showed significant increases in lift-drag ratio for the winglet configuration relative to a wing-alone configuration designed for the same lift coefficient and Mach number. Further, even larger increases in lift-drag ratio were observed at lift coefficients above the design value at all Mach numbers tested. The addition of these winglets had a negligible effect on the static lateral-directional stability characteristics of the configuration. No tests were made to determine the effect of these winglets at supersonic Mach numbers, where increases in drag caused by winglets might be more significant. Computational analyses were also performed for the two configurations studied. Linear and small-disturbance formulations were used. The codes were found to give reasonable performance estimates sufficient for predicting changes of this magnitude.
Author
Aircraft Design; Drag Reduction; Winglets; Aerodynamic Configurations; Low Aspect Ratio Wings; Induced Drag; Angle of Attack; Wind Tunnel Tests; Mach Number; Aerodynamic Coefficients; Lift; Prediction Analysis Techniques

19960015932; 96N22123 Old Dominion Univ., Norfolk, VA USA
An Experimental Investigation of Wall-Cooling Effects on Hypersonic Boundary-Layer Stability in a Quiet Wind Tunnel *Final Report*
Blanchard, Alan E., Old Dominion Univ., USA; Selby, Gregory V., Old Dominion Univ., USA; Feb. 1996, pp. 120; In English; Also announced as 19960015932
Contract(s)/Grant(s): NCC1-180; RTOP 505-59-50-02
Report No.(s): NASA-CR-198287; NAS 1.26:198287; No Copyright; Avail: CASI; A06, Hardcopy; A02, Microfiche

One of the primary reasons for developing quiet tunnels is for the investigation of high-speed boundary-layer stability and transition phenomena without the transition-promoting effects of acoustic radiation from tunnel walls. In this experiment, a flared-cone model under adiabatic- and cooled-wall conditions was placed in a calibrated, 'quiet' Mach 6 flow and the stability of the boundary layer was investigated using a prototype constant-voltage anemometer. The results were compared with linear-stability theory predictions and good agreement was found in the prediction of second-mode frequencies and growth. In addition, the same 'N=10' criterion used to predict boundary-layer transition in subsonic, transonic, and supersonic flows was found to be applicable for the hypersonic flow regime as well. Under cooled-wall conditions, a unique set of continuous spectra data was acquired that documents the linear, nonlinear, and breakdown regions associated with the transition of hypersonic flow under low-noise conditions.
Author
Boundary Layer Stability; Boundary Layer Transition; Hypersonic Boundary Layer; Wind Tunnel Walls; Hypersonic Wind Tunnels; Wind Tunnel Tests; Surface Cooling

19960015957; 96N21692 Minnesota Univ., Duluth, MN USA
On the Dispersion of Particles in the Near Wake of a Blunt Obstruction
Gomes, Marcos Sebastiao dePaula, Minnesota Univ., USA; Jan. 01, 1994, pp. 216; In English; Also announced as 19960015957; No Copyright; Avail: Univ. Microfilms Order No. DA9501098, Hardcopy, Microfiche

This work examines the dispersion of particles in the flow around a blunt obstruction, more specifically the residence time for particles in the near wake region after a source for the aerosols has been removed. It involved the development of a numerical model and several experiments for the validation of the model and for extending our knowledge on the subject. The numerical procedure was based on the kappa-epsilon model of turbulence, using the SIMPLER algorithm (Patankar, 1980), developed for inertialess particles. Preliminary experiments in which the concentration was measured through a sampling line proved inadequate. The dynamic response was slow due to the convective and diffusive dispersion of particles inside the tubing. We investigated the problem and developed a procedure that may be applied to other aerosol measurements. After building novel instrumentation, new experiments were started. The dimensionless residence time for particles in the near wake region H = tau U/D (suggested by Humphries and Vincent, 1976; r is the time constant for the concentration decay, U is the free-stream velocity and D is the plate width) was measured in the range of Reynolds numbers 1,000 less than or equal to Re less than or equal to 10,000. H was found to be approximately constant at about 9

+/- 1.5 for Re greater than or equal to 4,000. These results were in agreement with the numerical procedure. We then investigated the effect of free-stream turbulence for a range of dimensionless turbulence parameters A(sub f) = k(sub f)(exp l/2) l(sub f)UD (k(sub f) is the turbulence kinetic energy and l(sub f) is the integral length scale of the freestream turbulence) from 0.0015 to 0.071, for 1,500 less than or equal to Re less than or equal to 11,000. For Re above 4,000 H was 7.2 +/- 1.5 for 0.0015 less than or equal to A(sub f) less than or equal to 0.013, progressively increasing up to 10 for A(sub f) = 0.071. This was explained by the interaction of the free-stream turbulence with the vortex shedding mechanism. Finally, we performed experiments to investigate inertial effects. The aerosol was in the aerodynamic diameter range from 14 to 47 micro m, corresponding to Stokes numbers (the ratio of particle response time to a time scale for changes in the Row) from 0.02 to 1.8. Visualization of the particles' trajectories identified organized structures as a consequence of centrifugal inertial effects, referred to as 'de-mixing' by Crowe (1992).
Dissert. Abstr.
Reynolds Number; Vortex Shedding; Turbulence; Near Wakes; K-Epsilon Turbulence Model; Dynamic Response

19960015961; 96N21696 Iowa State Univ. of Science and Technology, Ames, IA USA
Numerical Evaluation of Corona Discharge as a Means of Boundary Layer Control and Drag Reduction
El-Khabiry, Samir Hamza, Iowa State Univ. of Science and Technology, USA; 1994, pp. 192; In English; Also announced as 19960015961; No Copyright; Avail: Univ. Microfilms Order No. DA9503549, Hardcopy, Microfiche

Problems of viscous drag reduction and boundary layer control have been and continue to be objectives of research for their economic impact and the enhancement of the flight characteristics of flying vehicles. Corona discharge is a new technique in this regard. The study of this technique requires consideration of the electrostatics and fluid mechanics. In order to effectively evaluate the technique with minimum complications, the geometry chosen was a dc positive corona discharge on a flat plate of zero thickness at zero angle of attack with the fluid flow as steady state, two-dimensional, incompressible viscous one. Five coupled partial differential equations govern this model requiring the simultaneous solution of these equations. A finite difference method has been employed to approximate these equations through an appropriate scheme for each equation. A clustered grid is used in the vertical direction to handle the high velocity gradient inside the boundary layer. The insufficient boundary conditions necessary for the numerical solution of Poisson's equation is compensated by making the numerical model find the appropriate computational domain which leads to a unique solution. Stability conditions of the five-point scheme approximating Poisson's equation has been determined computationally. Results obtained using the numerical model are presented. As a result of this research the corona discharge near a surface of finite conductivity is now better understood as an electrostatic phenomena. The corona discharge between wire-to-wire electrodes occurs if the electric potential difference between the electrodes is raised to a value higher than the corona onset voltage. The corona current is proportional to the potential difference between the electrodes and inversely proportional to the corona wire diameter. At the same time it does not significantly respond to the gap length between the electrodes until the corona wire diameter becomes large, then it varies inversely as the gap length. The corona discharge can be applied to reduce drag on bodies when the Reynolds number is relatively small. The drag reduction achieved by corona discharge inside a boundary layer is a function of many parameters. The drag reduction is proportional to both the electric potential difference and the gap length between the two electrodes and inversely proportional to the free stream velocity. It is also proportional to the location of the corona electrodes as measured from the plate leading edge and inversely proportional to the corona wire diameter. The increased effect of corona discharge at low flow speeds confirms its ability to significantly enhance the cooling rate of a hot body by boosting the convection of the flow around that body. The quantitative analysis of electrostatic cooling is the natural extension of this study.
Dissert. Abstr.
Boundary Layer Control; Boundary Conditions; Electric Corona; Drag Reduction; Viscous Drag; Finite Difference Theory; Partial Differential Equations; Electrostatics

19960016110; 96N22148 NASA Ames Research Center, Moffett Field, CA USA
A parametric study of supersonic laminar flow for swept wings using linear stability analysis
Cummings, Russell M., California Polytechnic State Univ., USA; Garcia, Joseph A., NASA Ames Research Center, USA; Tu, Eugene L., NASA Ames Research Center, USA; 1995, pp. 13; In English; 26th AIAA Fluid Dynamics Conference, 19-22 Jun. 1995, San Diego, CA, USA; Also announced as 19960016110
Report No.(s): NASA-TM-111258; NAS 1.15:111258; AIAA Paper 95-2277; Copyright Waived (NASA); Avail: CASI; A03, Hardcopy; A01, Microfiche

A parametric study to predict the extent of laminar flow on the upper surface of a generic swept-back wing (NACA 64A010 airfoil section) at supersonic speeds was conducted. The results were obtained by using surface pressure predictions from an Euler/Navier-Stokes computational fluid dynamics code coupled with a boundary layer code, which predicts detailed boundary layer profiles, and finally with a linear stability code to determine the extent of laminar flow. The parameters addressed are Reynolds number, angle of attack, and leading-edge wing sweep. The results of this study

show that an increase in angle of attack, for specific Reynolds numbers, can actually delay transition. Therefore, higher lift capability, caused by the increased angle of attack, as well as a reduction in viscous drag due to the delay in transition is possible for certain flight conditions.
Author
Swept Wings; Laminar Flow; Supersonic Flow; Supersonic Speed; Flow Stability; Prediction Analysis Techniques; Pressure; Applications Programs (Computers); Linear Equations; Upper Surface Blowing; Viscous Drag

19960016119; 96N22157 California Univ., Davis, CA USA
Analysis of in-flight boundary-layer state measurements on a subsonic transport wing in high-lift configuration
vanDam, C. P., California Univ., USA; Los, S. M., California Univ., USA; Miley, S. J., Old Dominion Univ., USA; Yip, L. P., NASA Langley Research Center, USA; Banks, D. W., NASA Langley Research Center, USA; Roback, V. E., NASA Langley Research Center, USA; Bertelrud, A., Analytical Services and Materials, Inc., USA; 1995, pp. 18; In English; 1st AIAA Aircraft Engineering, Technology, and Operations Congress, 19-21 Sep. 1995, Los Angeles, CA, USA; Also announced as 19960016119
Contract(s)/Grant(s): NCC1-163; NCC1-207; NAS1-19000; NAS1-19864
Report No.(s): NASA-CR-200146; NAS 1.26:200146; AIAA Paper 95-3911; Copyright Waived (NASA); Avail: CASI; A03, Hardcopy; A01, Microfiche

Flight experiments on NASA Langley's B737-100 (TSRV) airplane have been conducted to document flow characteristics in order to further the understanding of high-lift flow physics, and to correlate and validate computational predictions and wind-tunnel measurements. The project is a cooperative effort involving NASA, industry, and universities. In addition to focusing on in-flight measurements, the project includes extensive application of various computational techniques, and correlation of flight data with computational results and wind-tunnel measurements. Results obtained in the most recent phase of flight experiments are analyzed and presented in this paper. In-flight measurements include surface pressure distributions, measured using flush pressure taps and pressure belts on the slats, main element, and flap elements; surface shear stresses, measured using Preston tubes; off-surface velocity distributions, measured using shear-layer rakes; aeroelastic deformations of the flap elements, measured using an optical positioning system; and boundary-layer transition phenomena, measured using hot-film anemometers and an infrared imaging system. The analysis in this paper primarily focuses on changes in the boundary-layer state that occurred on the slats, main element, and fore flap as a result of changes in flap setting and/or flight condition. Following a detailed description of the experiment, the boundary-layer state phenomenon will be discussed based on data measured during these recent flight experiments.
Author
In-Flight Monitoring; Boundary Layer Transition; Subsonic Speed; Transport Aircraft; Aerodynamic Configurations; Lift; Flow Characteristics; Shear Stress; Wind Tunnel Tests; Pressure; Flapping

19960016290; 96N22176 NASA Lewis Research Center, Cleveland, OH USA
Effect of a delta tab on fine scale mixing in a turbulent two-stream shear layer
Foss, J. K., NASA Lewis Research Center, USA; Zaman, K. B. M. Q., NASA Lewis Research Center, USA; Jan. 1996, pp. 13; In English; 34th Aerospace Sciences Meeting and Exhibit, 15-18 Jan. 1996, Reno, NV, USA; Sponsored by American Inst. of Aeronautics and Astronautics, USA; Also announced as 19960016290
Contract(s)/Grant(s): RTOP 537-02-22
Report No.(s): NASA-TM-107131; E-10057; AIAA Paper 96-0546; NAS 1.15:107131; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The fine scale mixing produced by a delta tab in a shear layer has been studied experimentally. The tab was placed at the trailing edge of a splitter plate which produced a turbulent two-stream mixing layer. The tab apex tilted downstream and into the high speed stream. Hot-wire measurements in the 3-D space behind the tab detailed the three velocity components as well as the small scale population distributions. These small scale eddies, which represent the peak in the dissipation spectrum, were identified and counted using the Peak-Valley-Counting technique. It was found that the small scale populations were greater in the shear region behind the tab, with the greatest increase occurring where the shear layer underwent a sharp turn. This location was near, but not coincident, with the core of the streamwise vortex, and away from the region exhibiting maximum turbulence intensity. Moreover, the tab increased the most probably frequency and strain rate of the small scales. It made the small scales smaller and more energetic.
Author
Tabs (Control Surfaces); Turbulent Mixing; Mixing Layers (Fluids); Shear Layers

19960016595; 96N22239 Army Aviation Research and Development Command, Aeroflightdynamics Directorate., Moffett Field, CA USA
Suppression of Dynamic Stall by Steady and Pulsed Upper-Surface Blowing
Weaver, D., California Polytechnic State Univ., USA; McAlister, K. W., Army Aviation Research and Development Command, USA; Tso, J., California Polytechnic State Univ., USA;

Feb. 01, 1996, pp. 106; In English; Also announced as 19960016595
Contract(s)/Grant(s): RTOP 505-59-87
Report No.(s): NASA-TP-3600; NAS 1.60:3600; USAAT-COM-95-A-005; A-95103; No Copyright; Avail: CASI; A06, Hardcopy; A02, Microfiche

The Boeing-Vertol VR-7 airfoil was experimentally studied with steady and pulsed upper-surface blowing for sinusoidal pitching oscillations described by alpha = alpha(sub m) + 10 deg sin(omega t). The tests were conducted in the U.S. Army Aeroflightdynamics Directorate's Water Tunnel at NASA Ames Research Center. The experiment was performed at a Reynolds number of 100,000. Pitch oscillations with alpha(sub m) = 10 deg and 15 deg and with reduced frequencies ranging from k = 0.005 to 0.15 were examined. Blowing conditions ranged from C(sub mu) = 0.03 to 0.66 and F(+) = 0 to 3. Unsteady lift, drag, and pitching-moment loads were measured, and fluorescent-dye flow visualizations were obtained. Steady, upper-surface blowing was found to be capable of trapping a separation bubble near the leading edge during a portion of the airfoil's upward rotation. When this occurred, the lift was increased significantly and stall was averted. In all cases, steady blowing reduced the hysteresis amplitudes present in the loads and produced a large thrust force. The benefits of steady blowing diminished as the reduced frequency and mean angle of oscillation increased. Pulsed blowing showed only marginal benefits for the conditions tested. The greatest gains from pulsed blowing were achieved at F(+) = 0.9.
Author
Aerodynamic Stalling; Airfoils; Pitching Moments; Upper Surface Blowing

19960016631; 96N22264 NASA Langley Research Center, Hampton, VA USA
Computational considerations for the simulation of shock-induced sound
Casper, Jay, Old Dominion Univ., USA; Carpenter, Mark H., NASA Langley Research Center, USA; Dec. 1995, pp. 26; In English; Also announced as 19960016631
Contract(s)/Grant(s): RTOP 505-59-53-01
Report No.(s): NASA-TM-110222; NAS 1.15:110222; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The numerical study of aeroacoustic problems places stringent demands on the choice of a computational algorithm, because it requires the ability to propagate disturbances of small amplitude and short wavelength. The demands are particularly high when shock waves are involved, because the chosen algorithm must also resolve discontinuities in the solution. The extent to which a high-order-accurate shock-capturing method can be relied upon for aeroacoustics applications that involve the interaction of shocks with other waves has not been previously quantified. Such a study is initiated in this work. A fourth-order-accurate essentially nonoscillatory (ENO) method is used to investigate the solutions of inviscid, compressible flows with shocks in a quasi-one-dimensional nozzle flow. The design order of accuracy is achieved in the smooth regions of a steady-state test case. However, in an unsteady test case, only first-order results are obtained downstream of a sound-shock interaction. The difficulty in obtaining a globally high-order-accurate solution in such a case with a shock-capturing method is demonstrated through the study of a simplified, linear model problem. Some of the difficult issues and ramifications for aeroacoustics simulations of flows with shocks that are raised by these results are discussed.
Author
Aeroacoustics; Shock Wave Interaction; Shock Waves; Acoustic Simulation

19960016641; 96N22270 NASA Lewis Research Center, Cleveland, OH USA
Calculation of Turbulent Subsonic Diffuser Flows Using the NPARC Navier-Stokes Code
Dudek, J. C., NASA Lewis Research Center, USA; Georgiadis, N. J., NASA Lewis Research Center, USA; Yoder, D. A., Cincinnati Univ., USA; Feb. 1996, pp. 16; In English; 34th Aerospace Sciences Meeting and Exhibit, 15-18 Jan. 1996, Reno, NV, USA; Sponsored by American Inst. of Aeronautics and Astronautics, USA; Also announced as 19960016641
Contract(s)/Grant(s): RTOP 505-62-52
Report No.(s): NASA-TM-107177; E-10139; NAS 1.15:107177; AIAA Paper 96-0497; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Axisymmetric subsonic diffuser flows were calculated with the NPARC Navier-Stokes code in order to determine the effects various code features have on the flow solutions. The code features examined in this work were turbulence models and boundary conditions. Four turbulence models available in NPARC were used: the Baldwin-Lomax algebraic model, the Baldwin-Barth one-equation model, and the Chien kappa-epsilon and Wilcox kappa-omega two-equation models. The three boundary conditions examined were the free boundary, the mass flux boundary and the subsonic outflow with variable static pressure. In addition to boundary condition type, the geometry downstream of the diffuser was varied to see if upstream influences were present. The NPARC results are compared with experimental data and recommendations are given for using NPARC to compute similar flows.
Author
Axisymmetric Flow; Navier-Stokes Equation; Subsonic Flow; Turbulence Models; Turbulent Flow; Diffusers; Incompressible Flow; Computational Fluid Dynamics

19960016681; 96N22293 Purdue Univ., School of Aeronautics and Astronautics., Indianapolis, IN USA
High lift wake investigation *Progress Report*
Sullivan, J. P., Purdue Univ., USA; Schneider, S. P., Purdue

Univ., USA; Hoffenberg, R., Purdue Univ., USA; Feb. 02, 1996, pp. 15; In English; 13th AIAA Applied Aerodynamics Conference, Jun. 1995, USA; Also announced as 19960016681
Contract(s)/Grant(s): NAG2-854
Report No.(s): NASA-CR-200256; AIAA Paper 95-1912; NAS 1.26:200256; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The behavior of wakes in adverse pressure gradients is critical to the performance of high-lift systems for transport aircraft. Wake deceleration is known to lead to sudden thickening and the onset of reversed flow; this 'wake bursting' phenomenon can occur while surface flows remain attached. Although known to be important for high-lift systems, few studies of such decelerated wakes exist. In this study, the wake of a flat plate has been subjected to an adverse pressure gradient in a two-dimensional diffuser, whose panels were forced to remain attached by use of slot blowing. Pitot probe surveys, L.D.V. measurements, and flow visualization have been used to investigate the physics of this decelerated wake, through the onset of reversed flow.
Author
Wakes; Reversed Flow; Lift Devices; Pressure Gradients; Flow Measurement; Pressure Effects; Wind Tunnel Tests; Flow Distribution; Pressure Measurement; Turbulent Flow

19960016722; 96N22308 Research Inst. for Advanced Computer Science, Moffett Field, CA USA
Efficient Helicopter Aerodynamic and Aeroacoustic Predictions on Parallel Computers
Wissink, Andrew M., Minnesota Univ., USA; Lyrintzis, Anastasios S., Purdue Univ., USA; Strawn, Roger C., NASA Ames Research Center, USA; Oliker, Leonid, NASA Ames Research Center, USA; Biswas, Rupak, NASA Ames Research Center, USA; Jan. 1996, pp. 14; In English; AIAA 34th Aerospace Sciences Meeting and Exhibit, 15-18 Jan. 1996, Reno, NV, USA; Also announced as 19960016722
Contract(s)/Grant(s): NAS2-13721
Report No.(s): NASA-CR-199469; NAS 1.26:199469; RIACS-TR-96-04; AIAA Paper 96-0153; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This paper presents parallel implementations of two codes used in a combined CFD/Kirchhoff methodology to predict the aerodynamics and aeroacoustics properties of helicopters. The rotorcraft Navier-Stokes code, TURNS, computes the aerodynamic flowfield near the helicopter blades and the Kirchhoff acoustics code computes the noise in the far field, using the TURNS solution as input. The overall parallel strategy adds MPI message passing calls to the existing serial codes to allow for communication between processors. As a result, the total code modifications required for parallel execution are relatively small. The biggest bottleneck in running the TURNS code in parallel comes from the LU-SGS algorithm that solves the implicit system of equations. We use a new hybrid domain decomposition implementation of LU-SGS to obtain good parallel performance on the SP-2. TURNS demonstrates excellent parallel speedups for quasi-steady and unsteady three-dimensional calculations of a helicopter blade in forward flight. The execution rate attained by the code on 114 processors is six times faster than the same cases run on one processor of the Cray C-90. The parallel Kirchhoff code also shows excellent parallel speedups and fast execution rates. As a performance demonstration, unsteady acoustic pressures are computed at 1886 far-field observer locations for a sample acoustics problem. The calculation requires over two hundred hours of CPU time on one C-90 processor but takes only a few hours on 80 processors of the SP2. The resultant far-field acoustic field is analyzed with state of-the-art audio and video rendering of the propagating acoustic signals.
Author
Helicopters; Aeroacoustics; Navier-Stokes Equation; Parallel Computers; Rotary Wing Aircraft; Sound Fields; Computational Fluid Dynamics

19960016866; 96N22495 Michigan Univ., Ann Arbor, MI USA
The structure of sphere wakes at intermediate Reynolds numbers in still and turbulent environments
Wu, Jong-Shinn, Michigan Univ., USA; 1994, pp. 158; In English; Also announced as 19960016866; No Copyright; Avail: Univ. Microfilms Order No. DA9501066, Hardcopy, Microfiche

The structure of sphere wakes was studied for sphere Reynolds numbers in the range of 30-4000 in still environments and for sphere Reynolds numbers in the range of 130-1600 in turbulent environments. Measurements included dye traces illuminated by a laser light sheet for flow visualization and laser velocimetry for streamwise and cross-stream velocities. Classical similarity results for turbulent and laminar wakes were used to interpret and correlate the measurements. There are three wake regions identified in still environments: a fast-decay wake region that was observed only when vortex shedding was present (Re280), followed in succession by turbulent and laminar wake regions. Mean velocities within the turbulent and laminar wake regions scaled according to classical similarity theories, with transition between these regions at conditions where their estimates of mean streamwise velocities along the axis were the same: this occurred at local wake Reynolds numbers, Re(sub w)10, which is defined in terms of centerline velocity and wake width. Within the turbulent wake region, turbulence intensities along the axis were roughly 85% for Re(sub w)70; however, as the onset of the laminar wake region was approached, turbulence intensities along the axis became proportional to Re(sub w)(exp 7/4), which is consistent with scaling proposed some years ago by Batchelor and Townsend, and Philips, for the final decay period of axisymmetric turbulent wakes. The

structure of sphere wakes were measured in the turbulent environments with streamwise integral scales much larger than the sphere diameter and Kolmogorov length scales less than the sphere diameter. At these conditions, some phenomena observed for sphere wakes at comparable Reynolds numbers in nonturbulent environments were suppressed; instead, while the wakes were turbulent, their mean streamwise velocities scaled like self-preserving laminar wakes but with enhanced viscosities due to turbulence. Effective turbulent viscosities were relatively independent of position and the ratios of integral length scales and Kolmogorov length scales to sphere diameters; however, they progressively increased with sphere Reynolds numbers. Low and high Reynolds number regimes were observed, separated by a transition regime involving sphere Reynolds numbers in the range of 300-600 which was associated with the onset of vortex shedding in the near wake region. Wake mixing properties within the laminar-like turbulent wake region were affected by both the sphere Reynolds number and the ambient turbulence intensity in the low Reynolds number regime; in contrast, wake mixing properties mainly varied with the sphere Reynolds number in the high Reynolds number regime where wake turbulence was dominated by wake generated turbulence. Observed Strouhal numbers associated with eddy shedding were generally lower than the values observed earlier in turbulence-free environments.
Dissert. Abstr.
Laminar Wakes; Reynolds Number; Spheres; Turbulent Wakes; Flow Distribution; Turbulence

19960017161; 96N22753 NASA Lewis Research Center, Cleveland, OH USA
Skin friction reduction by micro-blowing technique
Hwang, Danny P., Inventor, NASA Lewis Research Center, USA; 1 DEC. 1995, pp. 29; In English; Also announced as 19960017161
Patent Info.: NASA-Case-LEW-15920-1; US-PATENT-APPL-SN-566211; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A system and method for reducing skin friction of object in relative motion to a fluid is discussed. The prior art includes air and water craft where fluids were pumped out of the craft to lower friction or provide other desirable qualities. The invention includes a skin forming a boundary between the object and the fluid, the skin having holes through which air is blowable and a transmitting apparatus for transmitting air through the skin. The skin has an inner layer and an outer layer the inner layer being a low permeable porous sheet, the outer layer being a plate having high porosity holes. The system may further include a suction apparatus for suctioning air from the outer layer into the inner layer. The method includes the steps of transmitting air through the inner layer and passing the air transmitted through the inner layer through the outer layer. The method may further include the step of bleeding air off of the outer layer using the suction apparatus.
NASA
Blowing; Friction Reduction; Skin Friction; Suction; Fluid Flow; Boundary Layer Control

19960017257; 96N22849 MCAT Inst., San Jose, CA USA
Numerical Simulation of the Flow About the F-18 HARV at High Angle of Attack *Final Report*
Murman, Scott M., MCAT Inst., USA; Feb. 1995, pp. 38; In English; Also announced as 19960017257; Original contains 2 color illustrations
Contract(s)/Grant(s): NCC2-729
Report No.(s): NASA-CR-200742; NAS 1.26:200742; MCAT-95-13; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The flowfield about aircraft maneuvering at high angles of attack is characterized by large regions of separated flow, the formation of strong vortical structures on the leeward side of the aircraft, and a close coupling between the flows about the separate components of the aircraft. As part of NASA's High Alpha Technology Program, research has been aimed at developing and extending numerical methods to accurately predict the high Reynolds number flow about the NASA High Alpha Research Vehicle (HAR17) at large angles of attack. The HARV is a highly-instrumented F-18 aircraft which has been used for flight testing because of its exceptional high-angle-of-attack capability. The resulting codes have been validated by comparison of the numerical results with in-flight aerodynamic measurements and flow visualization obtained on the HARV. These flight-validated numerical methods have been used to perform numerical analyses and optimization of new control concepts for high-alpha maneuverability. This can lead to safer and more efficient aircraft operating at high angles of attack. This report summarizes research done over the pist two years as part of NASA Grant NCC 2-729. This research has been aimed at validating numerical methods for computing the flow about the complete F-18 HARV at alpha = 30 deg and alpha = 45 deg. At 30 deg angle of attack, the flow about the F-18 is dominated by the formation, and subsequent breakdown, of strong vortices over the wing leading-edge extensions (LEX). As the angle of attack is increased to alpha = 45 deg, the fuselage forebody of the F-18 contains significant laminar and transitional regions which are not present at alpha = 30 deg. Further, the flow over the LEX at alpha = 45 deg is dominated by an unsteady shedding in time, rather than strong coherent vortices. This complex physics, combined with the complex geometry of a full-aircraft configuration, provides a challenge for current computational fluid dynamics (CFD) techniques. The following sections present a numerical method and grid generation scheme that was used, a review of prior research done to numerically model the F-18 HARV, and a discussion of the current research. The current research is broken into three main topics; the effect of

engine-inlet mass-flow rate on the F-18 vortex breakdown position, the results using a refined F-18 computational model to compute the flow at alpha = 30 deg and alpha = 45 deg, and research done using the simplified geometry of an ogive-cylinder configuration to investigate the physics of unsteady shear-layer shedding. The last section briefly summarizes the discussion.
Author
Angle of Attack; Computational Fluid Dynamics; F-18 Aircraft; Flow Distribution; Grid Generation (Mathematics); Vortices; High Reynolds Number; Inlet Flow; Engine Inlets

19960017462; 96N23052 Virginia Polytechnic Inst. and State Univ., Blacksburg, VA USA
Crossflow stability and transition experiments in a swept-wing flow
Dagenhart, John Ray, Virginia Polytechnic Inst. and State Univ., USA; 1993, pp. 291; In English; Also announced as 19960017462; No Copyright; Avail: Univ. Microfilms Order No. DA9318183, Hardcopy, Microfiche

An experimental examination of crossflow instability and transition on a 45 swept wing is conducted in the Arizona State University Unsteady Wind Tunnel. The stationary-vortex pattern and transition location are visualized using both sublimating-chemical and liquid-crystal coatings. Extensive hot-wire measurements are conducted at several measurement stations across a single vortex track. The mean and travelling-wave disturbances are measured simultaneously. Stationary crossflow disturbance profiles are determined by subtracting either a reference or a span-averaged velocity profile from the mean-velocity data. Mean, stationary-crossflow, and travelling-wave velocity data are presented as local boundary-layer profiles and as contour plots across a single stationary-crossflow vortex track. Disturbance-mode profiles and growth rates are determined. The experimental data are compared to predictions from linear stability theory. The mean-velocity profiles vary slightly across the stationary crossflow vortex at the first measurement station. The variation across the vortex increases with downstream distance until nearly all of the profiles become highly-distorted S-shaped curves. Local stationary crossflow disturbance profiles having either purely excess or deficit values develop at the upstream measurement stations. Further downstream the profiles take on crossover shapes not anticipated by the linear theory. The maximum streamwise stationary-crossflow velocity disturbances reach +/-20% of the edge velocity just before transition. The travelling-wave disturbances have single lobes at the upstream measurement stations as expected, but further downstream double lobed travelling-wave profiles develop. The travelling-wave intensity is always more than an order of magnitude lower than the stationary crossflow-vortex strength. The mean streamwise-velocity contours are nearly flat and parallel to the model surface at the first measurement station. Further downstream, the contours rise up and begin to roll over like a wave breaking on the beach. The stationary-crossflow contours show that a plume of low-velocity fluid rises near the center of the wavelength while high velocity regions develop near the surface at each end of the wavelength. There is no distinct pattern to the low-intensity travelling-wave contours until a short distance upstream of the transition location where the travelling-wave intensity suddenly peaks near the center of the vortex and then falls abruptly.
Dissert. Abstr.
Boundary Layer Transition; Cross Flow; Swept Wings; Traveling Waves; Velocity Distribution; Vortices; Wind Tunnel Tests; Boundary Layer Stability

19960017490; 96N23080 Georgia Inst. of Tech., Atlanta, GA USA
An experimental study of the interaction between a rotor wake and an airframe with and without flow separation
Kim, Jaimoo, Georgia Inst. of Tech., USA; 1993, pp. 212; In English; Also announced as 19960017490; No Copyright; Avail: Univ. Microfilms Order No. DA9400426, Hardcopy, Microfiche

The present study investigated the viscous interaction between a rotor wake and an airframe. Two different classes of viscous interaction were considered, interaction without and with pre-existing separated flow on the airframe surface. Two different simple configurations were used to simulate these situations: a rotor wake interaction with the boundary layer on a hemispheric cylinder airframe and a rotor wake interaction with a free shear layer downstream of a backward-facing step on the surface of the airframe. A tip vortex and a counter-rotating vortex were observed from a study of the isolated rotor wake, and their effects on the airframe pressure were identified. The tip vortex suction peak was observed all around the airframe and the flow visualization confirmed survival of the vortex filament under the airframe. A secondary vortex was visualized just upstream of the tip vortex when the tip vortex interacted with the airframe surface. Its effect was noted as a secondary suction peak on the airframe surface next to the tip vortex suction peak. Axial motion in the tip vortex was visualized and its effect on the airframe pressure was seen as a positive pressure peak. When the tip vortex interacted with the shear layer generated by an axisymmetric backward-facing step on the surface of the circular airframe, increased unsteadiness of the vortex location was observed with flow visualization and velocity histograms. Periodic destruction and reconstruction of the shear layer were observed both in the flow visualization and vorticity contours calculated from velocity measurement data. A velocity profile of the shear layer, measured with a fine vertical spacing, indicated that the flow reversal under the shear layer occurred only under the influence of the tip vortex. Periodic flapping of the shear layer due to the tip vortex effect was also observed in the velocity profile. Measurements of the periodic surface pressure dis-

tribution showed the tip vortex effect to be present all around the airframe. Surface pressure spectra measured just downstream of the backward-facing step indicated multiple harmonics of blade passage frequency and the undisturbed backstep flow frequency.
Dissert. Abstr.
Airframes; Boundary Layer Separation; Flapping; Flow Visualization; Separated Flow; Vortices; Fluid-Solid Interactions; Configuration Interaction; Aircraft Wakes

19960017520; 96N23110 Georgia Inst. of Tech., Atlanta, GA USA
A hybrid inverse optimization method for aerodynamic design of lifting surfaces
Santos, Luis Carlos de Castro, Georgia Inst. of Tech., USA; 1993, pp. 141; In English; Also announced as 19960017520; No Copyright; Avail: Univ. Microfilms Order No. DA9400437, Hardcopy, Microfiche

The present state of the world economy, challenges the aerospace industry to dramatically reduce the development cost of new designs. Adding that to the increasing demands in the performance of aerospace vehicles, more efficient ways of analysis and design of the aerodynamic configurations are necessary. Computational fluid dynamics presents itself as an alternative to costly wind tunnel studies, and can reduce development costs, during the preliminary design phase. A hybrid inverse optimization design method was developed to reduce the computational cost of the design procedure and also incorporate the enforcement of constraints. An existing solver independent inverse design technique was modified to include an optimization routine. The kernel of the inverse design routine, which is the solution of the linear system, resulting from finite difference approximation of an auxiliary equation, was replaced by an optimization technique. Using the conjugate direction method, the linear system is solved, in the least squares sense. The constraints are simultaneously enforced, keeping the computational cost very close to the original inverse design method, and adding the versatility of an optimization method. The method is general. Several applications of this approach to airfoil and wing design, using full-potential and Navier-Stokes flow solvers, in the transonic flight regime are presented. The low computational effort and generality of the technique represent an improvement to the state-of-the-art in aerodynamic design and allow its application in a Multidisciplinary Design optimization structure.
Dissert. Abstr.
Computational Fluid Dynamics; Cost Reduction; Finite Difference Theory; Multidisciplinary Design Optimization

19960017524; 96N23114 Iowa State Univ. of Science and Technology, Ames, IA USA
A numerical investigation of laminar airfoil stall
Black, Daniel Wayne, Iowa State Univ. of Science and Technology, USA; 1993, pp. 183; In English; Also announced as 19960017524; No Copyright; Avail: Univ. Microfilms Order No. DA9334962, Hardcopy, Microfiche

The details of an interacting boundary layer algorithm capable of calculating large scale laminar separation past airfoils at low speeds is given. Rationale behind various convergence acceleration methods is given. It is shown that linear based acceleration methods are limited to 50 percent savings in convergence rate. A nonlinear extrapolation method is proposed and tested on two simple model problems. Savings exceed the 50 percent limitation of the previous methods. Boundary layer results for laminar flow past symmetric airfoils at zero incidence are presented as a test of the methods. Leading edge marginal separation results at finite Reynolds numbers are presented. Richardson extrapolation of successive calculations is used to improve accuracy. Results for a zero thickness uncambered plate at angle of attack are presented.
Dissert. Abstr.
Aerodynamic Stalling; Laminar Flow; Boundary Layers; Reynolds Number; Leading Edges; Boundary Layer Separation; Uncambered Wings; Angle of Attack; Computational Fluid Dynamics; Grid Generation (Mathematics)

19960017525; 96N23115 Old Dominion Univ., Hampton, VA USA
Boundary layer influences on the subsonic near-wake of a family of three-dimensional bluff bodies
Alcorn, Charles William, Old Dominion Univ., USA; 1993, pp. 192; In English; Also announced as 19960017525; No Copyright; Avail: Univ. Microfilms Order No. DA9333593, Hardcopy, Microfiche

A study is reported on subsonic bluff body near-wake flows. It has been determined that one family of bluff bodies, namely slanted-base ogive cylinders, can experience either a closed recirculating near-wake or a longitudinal vortex near-wake depending on the base slant-angle and the Reynolds number. This suggests a dependence of near-wake parameters on the state of the boundary layer ahead of separation. This report addresses the influence of the boundary layer on the near wake of slanted-base bluff bodies. Experiments were conducted in two facilities, the 6-inch Magnetic Suspension and Balance System (MSBS) at NASA Langley Research Center and the Old Dominion University low-speed wind tunnel. Interference-free drag measurements in the 6-inch MSBS validated previous drag results. Measurements in the ODU facility were made to determine base pressures, wake stagnation point locations, and boundary layer velocity profiles. Furthermore, spectral and cross-spectral analyses of the fluctuating streamwise velocity in the near-wake were performed to determine frequencies and coherence of large-scale structures. It was determined that despite variations in the boundary layer state, base pressures and wake stagnation point locations correlate with the Reynolds number based on the boundary layer momentum thickness as the independent

variable. Variations in the frequency and coherence of large-scale structures were shown to exist with fixed boundary layer transition. A two-dimensional representation of a slanted-base configuration was studied analytically using classical theories and computationally using an existing finite element package. This study confirmed that the sudden changeover in wake structure is a result of flow reattachment onto the slanted-base.
Dissert. Abstr.
Bluff Bodies; Boundary Layers; Near Wakes; Reynolds Number; Three Dimensional Bodies; Vortices; Wind Tunnel Tests; Subsonic Flow; Recirculative Fluid Flow

19960017551; 96N23132 National Aerospace Lab., Tokyo, Japan
Effect of Blade Tip Planform on Shock Wave of Advancing Helicopter Blade
Aoyama, Takashi, National Aerospace Lab., Japan; Saito, Shigeru, National Aerospace Lab., Japan; Jul. 1995; ISSN 0389-4010, pp. 24; In Japanese; Also announced as 19960017551
Report No.(s): NAL-TR-1275; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Shock waves are generated on the advancing side of the blade tip of a helicopter rotor in the high-speed forward flight condition. They cause an increase in drag, vibration, and noise. In this study, the effect of blade tip planform on behavior of shock wave on the advancing rotor blade was investigated in detail using a calculation method which solves three-dimensional Euler equations by an implicit finite-difference method. The Newton iterative method was added to obtain the unsteady solution in forward flight. The calculations were performed for blades having the NACA0012 airfoil section along the entire blade span to investigate the planform effect alone. The parametric study clarified the effect of the sweep and the taper on shock wave behavior. In addition, a guideline of the blade planform design for the advancing blade was suggested, and a newly-devised tip planform which prevents shock wave generation was proposed. The effectiveness of the BERP and ONERA PF 2 planform which are examples of advanced tip planforms were also presented.
Author
Planforms; Blade Tips; Rotary Wings; Horizontal Flight; Shock Waves; Finite Difference Theory; Euler Equations of Motion; Computational Fluid Dynamics

19960017672; 96N23205 Lockheed Martin Tactical Aircraft Systems, Fort Worth, TX USA
Euler Technology Assessment for Preliminary Aircraft Design: Compressibility Predictions by Employing the Cartesian Unstructured Grid SPLITFLOW Code *Progress Report, 1 Feb. - 30 Sep. 1995*
Finley, Dennis B., Lockheed Martin Tactical Aircraft Systems, USA; Karman, Steve L., Jr., Lockheed Martin Tactical Aircraft Systems, USA; Mar. 1996, pp. 102; In English; Also announced as 19960017672
Contract(s)/Grant(s): NAS1-19000; RTOP 505-68-30-03
Report No.(s): NASA-CR-4710; NAS 1.26:4710; No Copyright; Avail: CASI; A06, Hardcopy; A02, Microfiche

The objective of the second phase of the Euler Technology Assessment program was to evaluate the ability of Euler computational fluid dynamics codes to predict compressible flow effects over a generic fighter wind tunnel model. This portion of the study was conducted by Lockheed Martin Tactical Aircraft Systems, using an in-house Cartesian-grid code called SPLITFLOW. The Cartesian grid technique offers several advantages, including ease of volume grid generation and reduced number of cells compared to other grid schemes. SPLITFLOW also includes grid adaption of the volume grid during the solution to resolve high-gradient regions. The SPLITFLOW code predictions of configuration forces and moments are shown to be adequate for preliminary design, including predictions of sideslip effects and the effects of geometry variations at low and high angles-of-attack. The transonic pressure prediction capabilities of SPLITFLOW are shown to be improved over subsonic comparisons. The time required to generate the results from initial surface data is on the order of several hours, including grid generation, which is compatible with the needs of the design environment.
Author (revised)
Compressibility Effects; Computational Fluid Dynamics; Euler Equations of Motion; Unstructured Grids (Mathematics); Cartesian Coordinates; Transonic Flow; Vortices

19960017689; 96N23221 Ecole Polytechnique, Montreal, Quebec Canada
A Multi-Block Euler Method for the Complete Challenger Configuration
Kafyeke, Fassi, Ecole Polytechnique, Canada; 1993, pp. 287; In English; Also announced as 19960017689; ISBN-0-315-90020-2; No Copyright; Avail: Univ. Microfilms Order No. DANN90020, Hardcopy, Microfiche

The present work proposes a methodology for solving the Euler equations governing the flow of inviscid compressible fluids. This methodology was developed with the objective of predicting such flows around complex aircraft configurations. The main application sought is the modelling of Canadair's Challenger CL-601 aircraft featuring turbofan nacelle closely coupled with the wing. The particular Euler method proposed in this work was implemented in the MBTEC program (MultiBlock Transonic Euler Code). In the MBTEC model, the Euler equations are cast in their unsteady conservative form, indicating balances of mass, momentum and energy. The equations can be solved as an initial value problem. Starting with a uniform flow approximation, the final steady flow is obtained by integrating the equations in time until convergence is reached. The numerical solution is

obtained by discretizing the equations using a finite volume explicit method. The discretization is made using a structured body-fitted grid defined around the aircraft. The spatial derivatives are centrally differenced. The scheme is stabilized by the addition of artificial viscosity, leading to the definition of shock waves without oscillations. The integration of the equations in time is made using a five-stage Runge-Kutta algorithm with excellent stability properties. to further improve the convergence of the computations, local time-stepping is used, advancing everywhere the solution at the largest time-step allowed by the Courant, Friedriechs and Lewy local condition. The application to complex three-dimensional configurations is by way of a particular multi-block technique which is original in the method that we have developed. The space around the aircraft is divided into subdomains or blocks, each simply connected and topologically equivalent to a cube, with six faces and eight corners. The Euler equations are solved in each block in sequence. The numerical information is transferred from block to block using an effective block interfacing scheme. The program uses one type of boundary condition per block face but any one of the following boundary conditions can be specified for any face of any block in the field: solid surface condition, symmetry condition, inflow and outflow conditions and, in particular, engine inlet and exhaust conditions. A progressive testing method was used in order to validate the multiblock code. Starting with simple isolated wing cases, we have adjusted the block interface boundary conditions, the convective flux accumulation, the addition of artificial viscosity in a multi-block grid and other numerical details to guarantee satisfactory convergence of the computations even in the most complex cases. The program was then used to model the isolated nacelle of the General Electric CF-34 engine powering the CL-601 Challenger. Calculations were made for the complete, powered CL-601. This model, made of 600 blocks and 1,571,580 nodes yielded excellent comparisons with experimental data, demonstrating once more the excellent accuracy of the code in the most complex cases.

Dissert. Abstr.

Aircraft Configurations; Euler Equations of Motion; Runge-Kutta Method; Finite Volume Method; Engine Inlets; Inviscid Flow; Multiblock Grids; Turbofans; Wings

19960017751; 96N23282 Maryland Univ., College Park, MD USA

The effects of surface catalysis on the hypersonic shock wave/boundary layer interaction

Grumet, Adam Abraham, Maryland Univ., USA; 1994, pp. 233; In English; Also announced as 19960017751; No Copyright; Avail: Univ. Microfilms Order No. DA9507959, Hardcopy, Microfiche

To accurately determine the local heat loads that could occur on a hypersonic vehicle, this dissertation presents a realistic numerical calculation of a hypersonic nonequilibrium shock wave/boundary layer interaction (SWBLI) that includes the effects of surface catalysis. The catalysis model from Jumper, Seward, and Newman was used to calculate the atomic oxygen and nitrogen recombination on a reaction-cured glass (RCG) surface. Also, catalysis rates for recombination on copper oxide were used as well. Furthermore, the surface temperature was determined by coupling the thermal conduction inside the surface with the convective heat flow due to the external fluid dynamics, for a reasonable heat pipe configuration. Results demonstrate that RCG, though relatively benign with regard to oxygen, does catalyze the atomic nitrogen at the surface, enough to make an appreciable difference in the surface heating. Results have also shown that for highly catalytic surfaces, such as copper, nitric oxide formation is considerable, even though its production is neglected from the catalytic mechanisms. With regard to the coupling between catalytic heating and the separation region fo the SWBLI, a comparison of results obtained from both mildly and highly separated flow indicate that surface catalysis significantly increases the heat transfer downstream of the separation region. However, because the majority of the chemical recombination is contained inside the separation zone and away from the region of peak heating, the effects of surface catalysis on the peak heat transfer is not critically affected by the separation zone size.

Dissert. Abstr.

Shock Wave Interaction; Hypersonic Shock; Boundary Layers; Catalysis; Separated Flow; Conductive Heat Transfer; Convective Heat Transfer; Mathematical Models; Hypersonic Flow; Nonequilibrium Flow; Recombination Reactions

19960017810; 96N23337 National Aerospace Lab., Tokyo, Japan

Flat Spin of Axisymmetric Bodies Near the Critical Reynolds Number Region

Tate, Atsushi, National Aerospace Lab., Japan; Iwasaki, Akihito, National Aerospace Lab., Japan; Fujita, Toshimi, National Aerospace Lab., Japan; Yoshinaga, Takashi, National Aerospace Lab., Japan; Jun. 1995; ISSN 0389-4010, pp. 24; In Japanese; Also announced as 19960017810

Report No.(s): NAL-TR-1271; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A series of flat spin experiments for recovering re-entering bodies have been conducted in a wind tunnel with cylinder, cone-cylinder, and bicone-cylinder models in the critical and supercritical Reynolds number regions. Using the free rotation method, the steady state spin rate was measured as a function of the freestream Reynolds number. Two modes of flat spin, a high spin mode, and a low spin mode were found to exist. The high spin mode was initiated in a fairly narrow freestream velocity region near the critical Reynolds number and continued in the supercritical Reynolds number region. During the high spin mode, the circumferential velocity of cylindrical models reached the magnitude of the freestream

velocity, In the case of the bicone-cylinder, the spin rate was shown to be erratic, appearing sensitive to small irregularities in the nose shape. Slim triangular prisms on these cylinder surfaces, aligned with the body axis, are effective to reduce the steady state spin rate.
Author
Spin; Conical Bodies; Axisymmetric Bodies; Angle of Attack; Reynolds Number; Spin Reduction; Wind Tunnel Stability Tests

19960017814; 96N23339 National Aerospace Lab., Tokyo, Japan
The Effect of Time Lag on the Stability of HOPE Model in a Dynamic Wind Tunnel Test
Motoda, Toshikazu, National Aerospace Lab., Japan; Sasa, Shuichi, National Aerospace Lab., Japan; Yanagihara, Masaaki, National Aerospace Lab., Japan; Tukamoto, Taro, National Aerospace Lab., Japan; Apr. 1995; ISSN 0389-4010, pp. 20; In Japanese; Also announced as 19960017814
Report No.(s): NAL-TR-1265; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The first dynamic wind tunnel experiment of the HOPE 16% scale model was conducted in 1992. In one experiment, the motion of the model plane became unstable when the elevon deflected suddenly and enormously. The mode change from altitude control to attitude control caused serious elevon deflection, but the attitude control system was theoretically stable and still worked at that time. The problem is that the unstable motion happened in spite of the theoretical expectation of stable model motion. The effect of time lag of the control system on the stability of the HOPE model was investigated using a Nyquist diagram and computer simulation. The aerodynamic data of the static wind tunnel test was used for these analyses. The time lag required to make the motion unstable was derived from the Nyquist diagram, and the results were confirmed by numerical simulation based on a linear motion equation. Also, non-linear simulation showed that the stability of the system is affected by non-linear characteristics of the model plane.
Author
Hope Aerospace Plane; Altitude Control; Attitude Control; Elevons; Nyquist Diagram; Time Lag; Wind Tunnel Tests; Deflection

03
AIR TRANSPORTATION AND SAFETY

Includes passenger and cargo air transport operations; and aircraft accidents.

19960015899; 96N21672 NASA Lewis Research Center, Cleveland, OH USA
Wind Tunnel Measured Effects on a Twin-Engine Short-Haul Transport Caused by Simulated Ice Accretions
Reehorst, Andrew, NASA Lewis Research Center, USA; Potapczuk, Mark, NASA Lewis Research Center, USA; Ratvasky, Thomas, NASA Lewis Research Center, USA; Laflin, Brenda Gile, NASA Langley Research Center, USA; Jan. 1996, pp. 28; In English; 34th Aerospace Sciences Meeting and Exhibit, 15-18 Jan. 1996, Reno, NV, USA; Sponsored by American Inst. of Aeronautics and Astronautics, USA; Also announced as 19960015899
Contract(s)/Grant(s): RTOP 505-68-10
Report No.(s): NASA-TM-107143; NAS 1.15:107143; E-10073; AIAA Paper 96-0871; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A series of wind tunnel tests were conducted to assess the effects of leading edge ice contamination upon the performance of a short-haul transport. The wind tunnel test was conducted in the NASA Langley 14 by 22 foot facility. The test article was a 1/8 scale twin-engine short-haul jet transport model. Two separate leading edge ice contamination configurations were tested in addition to the uncontaminated baseline configuration. Several aircraft configurations were examined including various flap and slat deflections, with and without landing gear. Data gathered included force measurements via an internal six-component force balance, pressure measurements through 700 electronically scanned wing pressure ports, and wing surface flow visualization measurements. The artificial ice contamination caused significant performance degradation and caused visible changes demonstrated by the flow visualization. The data presented here is just a portion of the data gathered. A more complete data report is planned for publication as a NASA Technical Memorandum and data supplement.
Author
Aircraft Icing; Wind Tunnel Tests; Aircraft Performance; Flow Visualization; Aircraft Configurations

19960016586; 96N22230 Johns Hopkins Univ., Center for Injury Research and Policy., Baltimore, MD USA
Crashes of Instructional Flights: Analysis of Cases and Remedial Approaches *Final Report*
Baker, Susan P., Johns Hopkins Univ., USA; Lamb, Margaret W., Johns Hopkins Univ., USA; Li, Guohua, Johns Hopkins Univ., USA; Dodd, Robert S., Johns Hopkins Univ., USA; Feb. 01, 1996, pp. 46; In English; Also announced as 19960016586
Contract(s)/Grant(s): FAA-93-G-045
Report No.(s): DOT/FAA/AM-96/3; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Instructional flights experience more than 300 crashes annually and are involved in more than one-third of all midair collisions. Research was undertaken to identify the circumstances of instructional crashes and describe factors related to pilots, aircraft, and the environment. NTSB data tapes were analyzed for crashes during 1989-1992 that involved a student pilot or a flight for instructional purposes. From this study it

was found that a greater emphasis during flight training needs to be placed on avoiding stalls and midair collisions, managing crosswinds, and understanding the elements of takeoff and landing flight dynamics prior to solo touch-and-go practice. Instructors contribute to crashes both directly during dual instruction and, less directly, through failure to successfully monitor their students and convey the elements of safe and proficient flight. Problems identified in this study should be communicated to flight instructors in their initial training as instructors, during preparation for their biennial re-licensure, and through dissemination of materials to all instructors.
Author
Flight Training; Crash Landing; Midair Collisions; Flight Safety; Students; Instructors; Assessments; Aircraft Control

19960016966; 96N22572 Wichita State Univ., National Inst. for Aviation Research., Wichita, KS USA
FY95 annual report and strategic plan: FY96-FY2000
Oct. 1995, pp. 97; In English; Also announced as 19960016966; No Copyright; Avail: CASI; A05, Hardcopy; A02, Microfiche

The National Institute for Aviation Research (NIAR) presents an assessment of the work accomplished in FY 1995. This assessment includes progress made, goals met, milestones marked, budgets, and expenditures. The major areas concentrated on in FY95 included crash dynamics, composites, aerodynamics, and CAD/CAM. Also reported is the NIAR Faculty Fellows program, other cooperative programs, and technology applications. The Institute's proposed plan of action and goals for FY96 through FY2000 are summarized.
CASI
Technology Utilization; Management Planning; Budgets; Research Management; Project Management; Aeronautics

19960017145; 96N22737 General Accounting Office, Washington, DC USA
Briefing Report to the Ranking Minority Member, Subcommittee on Military Procurement, Committee on National Security, House of Representatives. Military Aircraft Safety: Significant Improvements Since 1975
Cekala, Sharon A., General Accounting Office, USA; Beusse, William E., General Accounting Office, USA; Brady, Hugh, General Accounting Office, USA; Reid, Sharon, General Accounting Office, USA; Taylor, Harry E., Jr., General Accounting Office, USA; Feb. 1996, pp. 32; In English; Also announced as 19960017145
Report No.(s): GAO/NSIAD-96-69BR; B-270647; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche; GAO, PO Box 6015, Gaithersburg, MD 20877 HC; GAO, PO Box 6015, Gaithersburg, MD 20877 HC

DOD aviation safety has improved significantly over the last 21 years. Between fiscal years 1975 and 1995 for example, the annual number of Class A mishaps decreased from 309 to 76, while the number of fatalities decreased from 285 to 85. During this period, Class A mishaps per 100,000 flying hours, referred to as the mishap rate, also decreased from about 4.3 to 1.5. The value of Clas A losses remained fairly constant over the last 6 years, ranging from a high of about 1.6 billion dollars in fiscal year 1993 to a low of 1.2 billion in fiscal year 1994.
Derived from text
Aircraft Safety; Flight Safety

19960017458; 96N23048 Massachusetts Inst. of Tech., Cambridge, MA USA
Multidisciplinary design of an integrated microburst alerting system
Wanke, Craig R., Massachusetts Inst. of Tech., USA; 1993; In English; Also announced as 19960017458; No Copyright; Avail: Issuing Activity (MIT Libraries, Rm. 14-0551, Cambridge, MA), Hardcopy, Microfiche

A user-centered approach was applied to identify and analyze critical system design issues in a series of multidisciplinary studies. This approach is designed to optimize the decision-making performance of the end user, and thereby optimize the operational efficiency of the microburst alerting system. The primary end users of microburst alerts are flight crews, so the initial task was to determine which information is required by the crew for effective microburst alerting and the most effective presentation techniques for this information. This was done with a pilot survey and with two piloted part-task simulator experiments. Iconic graphical alerts were found to be significantly more useful than verbal or text communications, and several format and procedural issues for implementation of graphical alerts were examined. These studies included development of a part-task flight simulator, based on an advanced workstation with powerful graphics and rapid prototyping capabilities, which was found to be an exceptionally useful tool for preliminary evaluation of advanced cockpit information systems. Next, the issue of producing the required alert information -- the alert generation task -- was addressed. Possible hazard metrics for microburst intensity were compared, and an effective one selected based on a batch flight simulation technique. The energy-based 'F-factor' criterion averaged over 3000 ft of aircraft flight path was found to be a much better indicator of aircraft hazard than the 'delta-V' (total headwind-to-tailwind change) criterion currently used by the Terminal Doppler Weather Radar (TDWR) microburst detection system and the Low Level Windshear Alert System (LLWAS). A method was then developed for establishing a multilevel microburst alerting structure, using threshold values of a hazard metric based on measurable airmass parameters. The method was applied to determine three-level alerting thresholds for jet transports on approach to landing. Alert generation requires estimates of microburst characteristics; a model-based data assimilation technique for estimating microburst characteristics from multisensor measurements was therefore developed. The

extended Kalman filter-based technique incorporates elementary fluid mechanics and statistical characteristics of microbursts to improve estimation accuracy, and to allow estimation of quantities which cannot be directly measured. Dissert. Abstr.

Flight Crews; Graphical User Interface; Warning Systems; Microbursts (Meteorology); Aviation Meteorology; Human Factors Engineering; Flight Hazards

04
AIRCRAFT COMMUNICATIONS AND NAVIGATION

Includes digital and voice communication with aircraft; air navigation systems (satellite and ground based); and air traffic control.

19960016111; 96N22149 NASA Ames Research Center, Moffett Field, CA USA

Effects of ATC automation on precision approaches to closely space parallel runways

Slattery, R., NASA Ames Research Center, USA; Lee, K., Sterling Software, Inc., USA; Sanford, B., Sterling Software, Inc., USA; 1995, pp. 9; In English; AIAA Guidance, Navigation, and Control Conference, 7-10 Aug. 1995, Baltimore, MD, USA; Also announced as 19960016111

Report No.(s): NASA-TM-111255; NAS 1.15:111255; AIAA Paper 95-3367; Copyright Waived (NASA); Avail: CASI; A02, Hardcopy; A01, Microfiche

Improved navigational technology (such as the Microwave Landing System and the Global Positioning System) installed in modern aircraft will enable air traffic controllers to better utilize available airspace. Consequently, arrival traffic can fly approaches to parallel runways separated by smaller distances than are currently allowed. Previous simulation studies of advanced navigation approaches have found that controller workload is increased when there is a combination of aircraft that are capable of following advanced navigation routes and aircraft that are not. Research into Air Traffic Control automation at Ames Research Center has led to the development of the Center-TRACON Automation System (CTAS). The Final Approach Spacing Tool (FAST) is the component of the CTAS used in the TRACON area. The work in this paper examines, via simulation, the effects of FAST used for aircraft landing on closely spaced parallel runways. The simulation contained various combinations of aircraft, equipped and unequipped with advanced navigation systems. A set of simulations was run both manually and with an augmented set of FAST advisories to sequence aircraft, assign runways, and avoid conflicts. The results of the simulations are analyzed, measuring the airport throughput, aircraft delay, loss of separation, and controller workload.

Author

Air Traffic Control; Automatic Control; Aircraft Landing; Airborne Radar Approach; Flight Simulation; Distance Measuring Equipment; Aircraft Approach Spacing

19960016183; 96N22162 NASA Ames Research Center, Moffett Field, CA USA

Knowledge-based scheduling of arrival aircraft

Krzeczowski, K., NASA Ames Research Center, USA; Davis, T., NASA Ames Research Center, USA; Erzberger, H., NASA Ames Research Center, USA; Lev-Ram, I., Sterling Federal Systems, Inc., USA; Bergh, C., Massachusetts Inst. of Tech., USA; Aug. 10, 1995, pp. 12; In English; AIAA Guidance, Navigation, and Control Conference, 7-10 Aug. 1995, Baltimore, MD, USA; Also announced as 19960016183

Report No.(s): NASA-TM-111254; NAS 1.15:111254; AIAA Paper 95-3366; Copyright Waived (NASA); Avail: CASI; A03, Hardcopy; A01, Microfiche

A knowledge-based method for scheduling arrival aircraft in the terminal area has been implemented and tested in real-time simulation. The scheduling system automatically sequences, assigns landing times, and assigns runways to arrival aircraft by utilizing continuous updates of aircraft radar data and controller inputs. The scheduling algorithms is driven by a knowledge base which was obtained in over two thousand hours of controller-in-the-loop real-time simulation. The knowledge base contains a series of hierarchical 'rules' and decision logic that examines both performance criteria, such as delay reduction, as well as workload reduction criteria, such as conflict avoidance. The objective of the algorithms is to devise an efficient plan to land the aircraft in a manner acceptable to the air traffic controllers. This paper will describe the scheduling algorithms, give examples of their use, and present data regarding their potential benefits to the air traffic system.

Author

Air Traffic Control; Knowledge Based Systems; Knowledge Bases (Artificial Intelligence); Scheduling; Artificial Intelligence; Decision Making; Control Systems Design; Command and Control; Self Adaptive Control Systems; Automatic Landing Control; Controllers

19960016283; 96N22174 Galaxy Scientific Corp., Atlanta, GA USA

Design of a cooperative problem-solving system for en-route flight planning: An empirical evaluation

Layton, Charles, Galaxy Scientific Corp., USA; Smith, Philip J., Ohio State Univ., USA; McCoy, C. Elaine, Ohio Univ., USA; Jan. 01, 1994, pp. 26; In English; Also announced as 19960016283

Contract(s)/Grant(s): NCC2-615

Report No.(s): NASA-CR-200217; NAS 1.26:200217; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Both optimization techniques and expert systems technologies are popular approaches for developing tools to

assist in complex problem-solving tasks. Because of the underlying complexity of many such tasks, however, the models of the world implicitly or explicitly embedded in such tools are often incomplete and the problem-solving methods fallible. The result can be 'brittleness' in situations that were not anticipated by the system designers. to deal with this weakness, it has been suggested that 'cooperative' rather than 'automated' problem-solving systems be designed. Such cooperative systems are proposed to explicitly enhance the collaboration of the person (or a group of people) and the computer system. This study evaluates the impact of alternative design concepts on the performance of 30 airline pilots interacting with such a cooperative system designed to support enroute flight planning. The results clearly demonstrate that different system design concepts can strongly influence the cognitive processes and resultant performances of users. Based on think-aloud protocols, cognitive models are proposed to account for how features of the computer system interacted with specific types of scenarios to influence exploration and decision making by the pilots. The results are then used to develop recommendations for guiding the design of cooperative systems.
Author
Problem Solving; Flight Plans; Expert Systems; Decision Making; Planning

19960016580; 96N22224 Colorado Univ., Boulder, CO USA
A study of GPS measurement errors due to noise and multipath interference for CGADS ***Final Report***
Axelrad, Penina, Colorado Univ., USA; MacDoran, Peter F., Colorado Univ., USA; Comp, Christopher J., Colorado Univ., USA; Jan. 31, 1996, pp. 21; In English; Also announced as 19960016580
Contract(s)/Grant(s): NCC5-67
Report No.(s): NASA-CR-200249; NAS 1.26:200249; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This report describes a study performed by the Colorado Center for Astrodynamics Research (CCAR) on GPS measurement errors in the Codeless GPS Attitude Determination System (CGADS) due to noise and multipath interference. Preliminary simulation models fo the CGADS receiver and orbital multipath are described. The standard FFT algorithms for processing the codeless data is described and two alternative algorithms - an auto-regressive/least squares (AR-LS) method, and a combined adaptive notch filter/least squares (ANF-ALS) method, are also presented. Effects of system noise, quantization, baseband frequency selection, and Doppler rates on the accuracy of phase estimates with each of the processing methods are shown. Typical electrical phase errors for the AR-LS method are 0.2 degrees, compared to 0.3 and 0.5 degrees for the FFT and ANF-ALS algorithms, respectively. Doppler rate was found to have the largest effect on the performance.
Author
Global Positioning System; Error Analysis; Multipath Transmission; Least Squares Method; Algorithms; Noise Propagation; Attitude Control; Fast Fourier Transformations; Regression Analysis

19960016862; 96N22491 New Brunswick Univ., Fredericton, New Brunswick Canada
Composite GPS receiver modelling, simulations and applications
Zhuang, Weihua, New Brunswick Univ., Canada; 1993, pp. 262; In English; Also announced as 19960016862; ISBN-0-315-89235-8; No Copyright; Avail: Univ. Microfilms Order No. DANN89235, Hardcopy, Microfiche

A global positioning system (GPS) receiver has been modeled, analyzed, and implemented in software. The receiver has the L1-C/A, L1-P, and L2-P channels, with the output of GPS observables including the pseudorange time delay, carrier beat phase, and Doppler frequency shift. Based on the fundamentals of the GPS system and the signal structure, various relationships among the input signal parameters, internal variables of the receiver, and the GPS observables have been modeled mathematically. A digital energy detector is designed for the initial pseudorandom noise (PRN) code coarse synchronization on the time and frequency uncertainty region with double-dwell, straight line, and expanding-window serial search strategies. A digital delay lock loop (DDLL) with full-time early-late high speed correlators is modeled to despread the input signal and to extract the input code phase delay. From the power spectral density analysis of both input signal and noise, the closed form expressions of the detection and false-alarm probabilities of the code phase acquisition process and the variance of the code phase tracking error are derived. The linear and nonlinear performance of the DDLL is analyzed for optimizing the DDLL parameters with the trade-off between the tracking errors due to receiver dynamics and input noise. A digital phase-locked loop (DPLL) is used to extract the input carrier phase from the code-free signals after the DDLL. Upon analysis of the characteristics of the input phase noise, stable second and third order DPLL's are designed and analyzed with emphasis on the performance of the noise suppression and dynamic tracking capability. The Doppler shift is estimated based on the DPLL residual phase error. The output data volume of the observables is compressed by polynomial fit. The receiver functions are implemented with source code written mostly in C language. Simulations with the software package are performed respectively under the UNIX and TSO programming environments, which verify the theoretical performance analysis of the modeled receiver. The software receiver has been applied to analyze the multipath interference effects on the GPS observables

which is in accordance with theoretical analysis, and to study the effects of GPS antenna characteristics, including the antenna residual phase and antenna center (best-fit sphere) movement, on the GPS observables. The software receiver provides an additional approach to GPS research in positioning accuracy and is an important tool in designing the GPS receivers and in analyzing the effects of signal disturbances on GPS observables.
Dissert. Abstr.
Global Positioning System; Mathematical Models; Channel Noise; Radio Receivers; Signal Processing; Radio Signals; Navigation Instruments

19960017677; 96N23209 Advisory Group for Aerospace Research and Development, Mission Systems Panel., Neuilly-Sur-Seine, France
Twenty-five Years of Contributions to Air Traffic Handling (Research, Development, Operations and History): A Bibliography *Vingt-cinq annees de contributions au controle du trafic aerien (recherches, developpement, operations et historique): Une bibliographie*
Benoit, Andre, Editor, European Organization for the Safety of Air Navigation, Belgium; Feb. 1996, pp. 168; In English; Also announced as 19960017677
Report No.(s): AGARD-R-811; ISBN-92-836-1029-6; Copyright Waived; Avail: CASI; A08, Hardcopy; A02, Microfiche

Over 25 years, the Guidance and Control Panel of the Advisory Group for Aerospace Research and Development to the North Atlantic Treaty Organization has devoted part of its activities to the fascinating field known historically as Air Traffic Control, covering also Air Traffic Management, and more generally Air Traffic Handling. This Report provides a list of the summaries of the papers which were presented at the symposia and included in the AGARDographs devoted to this wide and most challenging subject, covering essentially, Air Traffic Control Systems (1972); A Survey of Modern Air Traffic Control, Vols. 1 and 2 (1975); Plans and Developments for Air Traffic Systems (1975); Air Traffic Management (1979); Air Traffic Control in Face of Users' Demand and Economy Constraints (1982); Efficient Conduct of Individual Flights and Air Traffic (1986); Aircraft Trajectories, Vols. 1, 2 and 3 (1990); Machine Intelligence in Air Traffic Management (1993); and On-Line Handling of Air Traffic (1994). The Report is completed by two indexes, an extended subject Index, and an Authors and Contributors Index.
Author
Bibliographies; Air Traffic Control; Air Traffic; Management Methods; Control Systems Design

05
AIRCRAFT DESIGN, TESTING AND PERFORMANCE

Includes aircraft simulation technology.

19960015635; 96N22443 Stanford Univ., CA USA
Lifting surface design using multidisciplinary optimization
Wakayama, Sean Rikio, Stanford Univ., USA; Jan. 01, 1995, pp. 181; In English; Also announced as 19960015635; No Copyright; Avail: Univ. Microfilms Order No. DA9516930, Hardcopy, Microfiche

Lifting surface design is affected by many considerations; drag, weight, and high-lift are particularly important. These effects place different and often opposite requirements on wing shape, complicating the selection of a best configuration. to assist this selection, a preliminary design method using nonlinear optimization has been developed. An isolated lifting surface design problem is formulated from aircraft mission parameters, typically to calculate the platform and twist minimizing cruise drag or maximizing range, subject to constraints such as structural weight and maximum section lift. Solving this with optimization requires very fast analyses that are capable of capturing the effects of detailed changes in wing shape. This motivated significant improvements that were made to structural calculations and maximum-lift prediction methods for preliminary design. A method was developed to evaluate wing weight and stiffness considering bending and buckling strength. A critical section method was modified to enable the prediction of flaps-down maximum lift, correcting for induced camber near the flap edge. The lifting surface optimization method performs platform design while accounting for many effects: static aeroelasticity, weight evaluated from multiple structural design conditions, induced drag, profile drag, compressibility drag, maximum lift, static stability, and control power constraints. The method was used to explore the influence of these effects on optimal wings, demonstrating how strongly lifting surface design is influenced by maximum-lift constraints. The method was also applied to studies of wing tip shape and optimal wing-tail configurations. In many cases, the optimizer exploits physical effects, creating design features that are easy to interpret in hindsight but difficult to predict in advance. In creating these designs, the method has demonstrated that optimization can be a valuable tool for lifting surface design.
Dissert. Abstr.
Lifting Bodies; Surface Properties; Multidisciplinary Design Optimization; Lift; Stiffness; Wings; Structural Design; Aerodynamic Configurations

19960015693; 96N21477 Stanford Univ., CA USA
New approaches to optimization in aerospace conceptual design
Gage, Peter James, Stanford Univ., USA; Jan. 01, 1995, pp.

188; In English; Also announced as 19960015693; No Copyright; Avail: Univ. Microfilms Order No. DA9516823, Hardcopy, Microfiche

Aerospace design can be viewed as an optimization process. A multi-dimensional search is conducted, with the physical description modified to produce the best functional performance. Despite this character and the wide availability of optimization software, conceptual design is rarely performed using formal search algorithms. The failure to exploit the benefits of optimization is examined in this thesis, and three key issues that restrict the success of automatic search are found. New approaches are introduced to address the integration of analyses and optimizers, to avoid the need for accurate gradient information and a smooth search space (required for calculus based optimization), and to remove the restrictions imposed by complexity problem formulations. Conceptual design of aircraft is a difficult endeavour. The disciplines are tightly coupled and performance assessment must be precise, so a design system is a complex combination of large analysis modules. Optimization should be performed in a flexible, extensible environment that conveniently links these modules and automatically co-ordinates their execution. A quasi-procedural architecture provides such an environment, and its efficient control of a large-scale design task is demonstrated. Calculus-based optimizers use gradient information to search for performance improvement. For many design tasks, gradients cannot be accurately computed, because variables have discrete values, analyses are noisy, or the search space has discontinuous topology. A genetic algorithm that provides a search method for such domains is studied. Applications demonstrate the utility of genetic optimization. The relationship between the genetic algorithm use a complete a priori prescription of the search space, which, to a large degree, defines a design concept. The value of each variable is optimized, but the search scheme cannot alter the fixed set of parameters. A variable-complexity genetic algorithm is created to permit more flexible parameterization, so that the level of characteristics are identified in simple designs, and refined by increasing description details. Structural and aerodynamic applications prove that this optimizer can discover novel designs.

Dissert. Abstr.

Aircraft Design; Aerospace Industry; Optimization

19960015849; 96N21633 Maryland Univ., College Park, MD USA

Aeroelastic optimization of advanced geometry and composite helicopter rotors

Ganguli, Ranjan, Maryland Univ., USA; Jan. 01, 1994, pp. 398; In English; Also announced as 19960015849; No Copyright; Avail: Univ. Microfilms Order No. DA9514522, Hardcopy, Microfiche

Using an analytical approach, sensitivity analyses and aeroelastic optimization procedures are developed for advanced geometry and composite rotors. Aeroelastic and sensitivity analyses of the rotor based on a finite element method in space and time are linked with automated optimization algorithms to perform optimization of a four-bladed, soft-inplane hingeless rotor. For the advanced geometry rotor, design variables include nonstructural mass and its placement, blade bending stiffness (flap, lag and torsion) and blade geometry (sweep, droop and planform taper) at five spanwise stations. The objective function constitutes minimization of all six vibratory hub loads subject to constraints on frequency placement, autorotational inertia and aeroelastic stability of the blade in forward flight Optimization results show reduction in the 4/rev hub loads of 25-60 percent. In another part of this study, aeroelastic optimization is performed for a composite rotor with the blade span modeled as a single-cell as well as a two-cell box-beam. The design variables are the ply angles of the cross-section of the blade spar. Optimization is performed for several layups and configurations and it is concluded that elastic stiffnesses reduce the objective function by about 20-40 percent; composite couplings yield a further reduction in the objective of about 10-15 percent. Using vibratory hub loads alone in the objective function can cause an increase in the blade root bending moments leading to higher dynamic stresses. This issue is addressed by performing a multi-objective optimization on the composite rotor by using a combination of vibratory hub loads and vibratory bending moments in the objective function. Optimum designs for the multi- objective optimization show a reduction in the objective function of about 30 percent from the starting design; 20 percent of this reduction is due to elastic stiffness and 10 percent due to composite coupling (flap bending-torsion coupling). As compared to the starting design, the optimum solution results in a 15-60 percent reduction of the 4/rev hub loads as well as a reduction in the peak-to-peak flap and lag bending moments of 11 and 14 percent, respectively, compared to the starting design. Starting from an initially infeasible design with a 3 percent requirement on lag mode damping, the optimum solution with composite chordwise bending-torsion coupling results in an increase in lag mode damping of 130-200 percent, compared to the starting design.

Dissert. Abstr.

Aeroelasticity; Rotary Wings; Design Analysis; Finite Element Method; Airfoil Profiles

19960015856; 96N21635 Arizona State Univ., Dept. of Mechanical and Aerospace Engineering, Tempe, AZ USA

Development of Continuous and Discrete Approaches for Aerodynamic Sensitivity Analysis *Progress Report*

Chattopadhyay, Aditi, Arizona State Univ., USA; Narayan, Johnny R., Arizona State Univ., USA; Xu, Wensheng, Arizona State Univ., USA; Jan. 1995, pp. 19; In English; Also announced as 19960015856

Contract(s)/Grant(s): NCC2-5127

Report No.(s): NASA-CR-200626; NAS 1.26:200626; No

Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A comparative study of discrete and continuous semi-analytical sensitivity analysis techniques for evaluating aerodynamic sensitivities in optimization procedures involving CFD solvers for accurate and detailed flow solutions, was proposed. Although there is extensive literature on the discrete and continuous approaches for calculating aerodynamic sensitivities, a study of the relative efficiencies of the approaches has not been performed. This study is crucial in selecting an appropriate sensitivity analysis technique for large scale, multidisciplinary design optimization problems using comprehensive analysis procedures. The study will compare the accuracy and computational efficiency of the two techniques using representative model optimization problems. The nature of the optimum solutions and the computational time (CPU hours) required for arriving at optimum designs are also of interest. The purpose of the present research is to extend the ongoing effort on semi-analytical sensitivity analysis techniques by performing this important comparative study of the two techniques. The proposed study will compare the accuracy and computational efficiency of the two techniques using representative model optimization problems. The nature of the optimum solutions and the CPU time required for convergence will also be of interest. Finally, parallelization of the developed sensitivity analysis procedures is proposed to improve the computational efficiency and the adaptability to a variety of CFD solvers.

Derived from text

Multidisciplinary Design Optimization; Design Analysis; Computational Fluid Dynamics; Flow Distribution; Aerodynamic Coefficients; Aircraft Design

19960015868; 96N21647 Cleveland State Univ., Dept. of Civil Engineering., OH USA

Probabilistic Fiber Composite Micromechanics *Final Report*

Stock, Thomas A., Cleveland State Univ., USA; Jan. 1996, pp. 196; In English; Also announced as 19960015868

Contract(s)/Grant(s): NAG3-550; RTOP 505-63-5B

Report No.(s): NASA-CR-198443; E-10082; NAS 1.26:198443; No Copyright; Avail: CASI; A09, Hardcopy; A03, Microfiche

Probabilistic composite micromechanics methods are developed that simulate expected uncertainties in unidirectional fiber composite properties. These methods are in the form of computational procedures using Monte Carlo simulation. The variables in which uncertainties are accounted for include constituent and void volume ratios, constituent elastic properties and strengths, and fiber misalignment. A graphite/epoxy unidirectional composite (ply) is studied to demonstrate fiber composite material property variations induced by random changes expected at the material micro level. Regression results are presented to show the relative correlation between predictor and response variables in the study. These computational procedures make possible a formal description of anticipated random processes at the intra-ply level, and the related effects of these on composite properties.

Author

Fiber Composites; Graphite-Epoxy Composites; Probability Theory; Micromechanics; Monte Carlo Method; Computerized Simulation; Random Processes; Reinforcing Fibers

19960016113; 96N22151 NASA Langley Research Center, Hampton, VA USA

Airbreathing hypersonic vehicle design and analysis methods

Lockwood, Mary Kae, NASA Langley Research Center, USA; Petley, Dennis H., NASA Langley Research Center, USA; Hunt, James L., NASA Langley Research Center, USA; Martin, John G., Lockheed Martin Engineering and Sciences Co., USA; Jan. 18, 1996, pp. 11; In English; 34th Aerospace Sciences Meeting and Exhibit, 15-18 Jan. 1996, Reno, NV, USA; Also announced as 19960016113; Original contains 1 color illustration

Report No.(s): NASA-TM-111249; NAS 1.15:111249; AIAA Paper 96-0381; Copyright Waived (NASA); Avail: CASI; A03, Hardcopy; A01, Microfiche

The design, analysis, and optimization of airbreathing hypersonic vehicles requires analyses involving many highly coupled disciplines at levels of accuracy exceeding those traditionally considered in a conceptual or preliminary-level design. Discipline analysis methods including propulsion, structures, thermal management, geometry, aerodynamics, performance, synthesis, sizing, closure, and cost are discussed. Also, the on-going integration of these methods into a working environment, known as HOLIST, is described.

Author

Air Breathing Engines; Air Breathing Boosters; Design Analysis; Hypersonic Vehicles; Propulsion System Configurations; Structural Analysis

19960016182; 96N22161 NASA Langley Research Center, Hampton, VA USA

Equivalent plate modeling for conceptual design of aircraft wing structures

Giles, Gary L., NASA Langley Research Center, USA; 1995, pp. 18; In English; 1st AIAA Aircraft Engineering Technology and Operations Congress, 19-21 Sep. 1995, Los Angeles, CA, USA; Also announced as 19960016182

Report No.(s): NASA-TM-111263; NAS 1.15:111263; AIAA Paper 95-3945; Copyright Waived (NASA); Avail: CASI; A03, Hardcopy; A01, Microfiche

This paper describes an analysis method that generates conceptual-level design data for aircraft wing structures. A key requirement is that this data must be produced in a timely manner so that is can be used effectively by multidisciplinary synthesis codes for performing systems studies. Such a capability is being developed by enhancing an equivalent plate

structural analysis computer code to provide a more comprehensive, robust and user-friendly analysis tool. The paper focuses on recent enhancements to the Equivalent Laminated Plate Solution (ELAPS) analysis code that significantly expands the modeling capability and improves the accuracy of results. Modeling additions include use of out-of-plane plate segments for representing winglets and advanced wing concepts such as C-wings along with a new capability for modeling the internal rib and spar structure. The accuracy of calculated results is improved by including transverse shear effects in the formulation and by using multiple sets of assumed displacement functions in the analysis. Typical results are presented to demonstrate these new features. Example configurations include a C-wing transport aircraft, a representative fighter wing and a blended-wing-body transport. These applications are intended to demonstrate and quantify the benefits of using equivalent plate modeling of wing structures during conceptual design.
Author
Plates (Structural Members); Mathematical Models; Body-Wing Configurations; Wing Profiles; Aircraft Design; Aircraft Structures; Winglets; Applications Programs (Computers); Multidisciplinary Design Optimization

19960016674; 96N22287 NASA Langley Research Center, Hampton, VA USA
Multidisciplinary aerospace design optimization: Survey of recent developments
Sobieszczanski-Sobieski, Jaroslaw, NASA Langley Research Center, USA; Haftka, Raphael T., Florida Univ., USA; Jan. 01, 1995, pp. 38; In English; 34th Aerospace Sciences Meeting and Exhibit, 15-18 Jan. 1995, Reno, NV, USA; Also announced as 19960016674
Report No.(s): NASA-TM-111250; AIAA Paper 96-0711; NAS 1.15:111250; Copyright Waived (NASA); Avail: CASI; A03, Hardcopy; A01, Microfiche

The increasing complexity of engineering systems has sparked increasing interest in multidisciplinary optimization (MDO). This paper presents a survey of recent publications in the field of aerospace where interest in MDO has been particularly intense. The two main challenges of MDO are computational expense and organizational complexity. Accordingly the survey is focussed on various ways different researchers use to deal with these challenges. The survey is organized by a breakdown of MDO into its conceptual components. Accordingly, the survey includes sections on Mathematical Modeling, Design-oriented Analysis, Approximation Concepts, Optimization Procedures, System Sensitivity, and Human Interface. With the authors' main expertise being in the structures area, the bulk of the references focus on the interaction of the structures discipline with other disciplines. In particular, two sections at the end focus on two such interactions that have recently been pursued with a particular vigor: Simultaneous Optimization of Structures and Aerodynamics, and Simultaneous Optimization of Structures Combined With Active Control.
Author
Multidisciplinary Design Optimization; Mathematical Models; Active Control

19960017096; 96N22692 Maryland Univ., College Park, MD USA
A design-oriented aeromechanical analysis for hingeless rotor helicopters in turning flight
Spence, Anne Marie Oetting, Maryland Univ., USA; 1994, pp. 343; In English; Also announced as 19960017096; No Copyright; Avail: Univ. Microfilms Order No. DA9514584, Hardcopy, Microfiche

This dissertation describes a mathematical model for a hingeless rotor helicopter capable of flight in both steady, level and turning flight regimes. The rotor blade is modeled as a Bernoulli-Euler beam undergoing moderately large elastic deflections in coupled flap-lag-torsion. In turn, the fuselage is described by nonlinear Euler equations valid for large fuselage attitudes and rates. The combined effects of turn rate, aircraft speed, and flight path angle on the blade response, aeromechanical stability and flight dynamic stability are discussed. The results indicate that steady level turns tend to stabilize the lag mode, while descending turns are destabilizing and may limit the maneuver envelope of the helicopter due to instabilities in the progressive and differential lag modes. These instabilities, however, tend to be weak. Judicious choice of flap and lag fundamental frequencies is necessary to insure stability of the lag mode in descending flight. Minor modifications are made to the original aeromechanical analysis to provide frequency response information for comparison with the new helicopter handling qualities specifications. The effect of selected rotor design parameters on the handling qualities ratings, aeromechanic and night dynamic characteristics of a hingeless rotor helicopter are then analyzed. The results indicate that a decrease in the flapping frequency and a positive elastic axis/aerodynamic center offset have a degrading effect on handling qualities while forward center of gravity offset improves the overall handling qualities. Regressive lag mode stability is improved by a positive elastic axis/aerodynamic center offset and by decreasing the flapping frequency while a forward center of gravity offset causes the regressive lag mode to be destabilized. The addition of roll attitude feedback can also be used to stabilize an unstable regressive lag mode, although gains are limited by the stability of the coupled roll/regressive flap mode. A method is described for the calculation of the sensitivities of blade root loads, hub loads and modal loads to changes of blade design parameters using a chain rule differentiation approach. The algorithm exploits features of the formulation of the blade and fuselage equations of motion, and of the solution technique to calculate the sensitivities at a fraction of the cost of an aero-

elastic analysis. The results indicate that the semi-analytical technique is very accurate and computationally efficient.
Dissert. Abstr.
Aerodynamic Stability; Aeroelasticity; Equations of Motion; Helicopter Control; Helicopters; Mathematical Models; Turning Flight; Rotor Dynamics

19960017104; 96N22700 Minnesota Univ., Duluth, MN USA
Optimal helicopter operation from a clear heliport in the event of one engine failure
Sharma, Vivek, Minnesota Univ., USA; 1994, pp. 200; In English; Also announced as 19960017104; No Copyright; Avail: Univ. Microfilms Order No. DA9514680, Hardcopy, Microfiche

The Federal Aviation Administration (FAA) certifies a transport helicopter as either Category-A or Category-B. The Category-A certification applies to multiengine helicopters with independent engine systems, and requires that the helicopter can either continue flight or land safely in the event of single engine failure. The Category-B certification applies to either single engine or multiengine helicopters, and only requires that a safe landing be possible in the case of an engine failure. For a Category-A helicopter, the decision to land or to 'stay up' depends on the conditions at engine failure. If an engine fails during a normal takeoff before the helicopter reaches the takeoff decision point (TDP), the pilot must reject the takeoff (RTO). If an engine fails after passing the TDP, the pilot must continue the takeoff (CTO). If an engine fails during a normal landing approach before reaching the landing decision point (LDP), the pilot may either continue or balk the landing (BL). The pilot must continue the landing (CL) for an engine failure past the LDP. Optimal clear heliport operation of a Category-A helicopter is studied using nonlinear optimal control theory. The objectives are to maximize the payload capacity and to minimize the runway length requirements, subject to the FAA safety requirements. A point-mass model of the UH-60A helicopter is used for trajectory optimizations. The results reveal essential features of Category-A helicopter flight. In either CTO or BL, the helicopter needs to climb out. The rotor speed stays at its minimum limit for some time so that rotational energy is transformed to potential and kinetic energy. The maximum weight is determined by the OEI power available and the minimum height allowed. In either RTO or CL, the helicopter needs to land. The rotor speed stays at its maximum limit for some time so that the stored rotor energy can be used to cushion the touchdown. The helicopter weight does not affect the landing trajectories significantly. Optimal trajectories provide the minimum runway length requirement, and the maximum payload capacity of the helicopter, with guaranteed safety. The solutions are useful for choosing optimal critical decision points for specified conditions.
Dissert. Abstr.
Control Theory; Engine Failure; Helicopters; Heliports; Multiengine Vehicles; Takeoff; Helicopter Control; Helicopter Engines

19960017241; 96N22833 Naval Air Warfare Center, Aircraft Div., Patuxent River, MD USA
Developing Flight Test Techniques to Ensure Proper Rigging of Highly Augmented Aircraft
Traven, Ricardo, Naval Air Warfare Center, USA; Whitley, Susan E., Naval Air Warfare Center, USA; Sep. 05, 1995, pp. 12; In English; Also announced as 19960017241
Report No.(s): AD-A300965; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Uncommanded rolling moments have been considered a natural part of the aging process of some F/A-18 airplanes. These rolling moments can range from an annoyance requiring constant trimming to a serious degradation in handling qualities which can cause premature departures during maneuvering flight. Uncommanded rolling moments in flight can generally be eliminated with rigging if the surface which is causing the roll is known. Due to the highly augmented flight control architecture of the F/A-18, it is very difficult to isolate the actual control surface that is causing the uncommanded rolling moment since so many surfaces are active. NAWC-AD has developed techniques to identify the flight control surface causing uncommanded rolling moments. These techniques have been used successfully on four F/A-18 airplanes to identify which flight control surface was causing uncommanded roll rates. Additional flight tests are planned to further refine the techniques described in this paper. These techniques have been demonstrated to reduce uncommanded rolling moments in the F/A-18 and can be applied to all aircraft.
DTIC
Flight Tests; Flight Control; Control Systems Design; Rigging; Roll

19960017250; 96N22842 Naval Air Warfare Center, Aircraft Div., Patuxent River, MD USA
US Navy T-45A Departure and Spin Evaluation
Mortensen, Joseph A., Naval Air Warfare Center, USA; Aug. 03, 1995, pp. 19; In English; Also announced as 19960017250
Report No.(s): AD-A300964; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

U.S. Navy T-45A departure and spin evaluation was conducted from 8 July to 30 September 1994. The initial T-45A flight manual departure and spin recovery procedures evaluated proved unsatisfactory in providing consistent recoveries from inverted spin modes. Specifically, the procedures allowed the rudder to blow-out in the pro-spin direction during recovery, resulting in spin mode transitions and delayed recoveries. Furthermore, a third sustained inverted spin mode was discovered during flight tests. The initial flight manual procedures were successfully modified to reduce the occur-

rence of rudder blow-out during recovery attempts and provided more consistent recovery characteristics.
DTIC
Flight Tests; Turboprop Engines; Aircraft Performance; Aircraft Control; Aircraft Maneuvers

19960017296; 96N22887 Defence Science and Technology Organisation, Airframes and Engines Div., Canberra, Australia
Vibration Test on a Nomad N24A Aircraft Fitted with One Modified Aileron
Farrell, P. A., Defence Science and Technology Organisation, Australia; Dunn, S. A., Defence Science and Technology Organisation, Australia; Rider, C. D., Defence Science and Technology Organisation, Australia; Jun. 1995, pp. 41; Also announced as 19960017296
Report No.(s): AD-A301189; DSTO-TN-0010; DODA-AR-009-262; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Following doubt raised about the loads experienced on the flap and aileron of Nomad aircraft in flight, a flight test program was formulated to measure these loads. An aileron fitted with strain gauges and the associated wiring was installed on a Nomad N24A aircraft but this rendered the aircraft non-standard. to verify the aeroelastic stability of the aircraft when fitted with the instrumented aileron, a flight flutter trial was proposed. An abbreviated Ground Vibration Test (GVT) was conducted on the aircraft to support the flutter trial, and this report describes the GVT and presents the results.
DTIC
Flight Tests; Ground Tests; Ailerons; Research Aircraft; Vibration Tests; Flight Stability Tests; Flaps (Control Surfaces)

19960017304; 96N22895 Army Aeromedical Research Lab., Fort Rucker, AL USA
Utilization of Visual Ports in U.S. Army Rotary-Wing Aircraft *Final Report*
Ivey, Rebecca H., Army Aeromedical Research Lab., USA; Rash, Clarence E., Army Aeromedical Research Lab., USA; Levine, Richard R., Army Aeromedical Research Lab., USA; Wentworth, Sean L., Army Aeromedical Research Lab., USA; McGowin, Everette, III, Army Aeromedical Research Lab., USA; Aug. 1995, pp. 192; Also announced as 19960017304
Contract(s)/Grant(s): 3M162787A879
Report No.(s): AD-A301143; No Copyright; Avail: CASI; A09, Hardcopy; A02, Microfiche

This report documents the location and extent of visual ports (windows) in U.S. Army rotary-wing aircraft and assesses utilization and associated problems. The study was conducted in three parts: First, the visual ports of each aircraft type were documented through visual plots and photography. Second, questionnaire data for 333 aviators and crewmen addressing the use of and problems with each visual port were collected and analyzed. Third, a search of the U.S. Army Safety Center accident database was collected to identify the frequency of accidents in which impaired vision was a possible factor.
DTIC
Aircraft Pilots; Position (Location); Rotary Wing Aircraft; Visual Observation; Visual Flight

19960017336; 96N22927 Naval Air Warfare Center, Patuxent River, MD USA
Integrated New Test Aircraft Capability (INTAC)
Carico, Dean, Naval Air Warfare Center, USA; Aug. 11, 1995, pp. 11; ITEA International Symposium, 2-5 Oct. 1995, Huntsville, AL, USA; Also announced as 19960017336
Report No.(s): AD-A300966; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The current and projected emphasis on declining budgets, reduced staffs, and tightened schedules, while trying to maintain or improve productivity, dictates a new approach to T&E that takes advantage of the ongoing technology explosion. Automation and standardization, combined with analysis and high technology simulation options, permit a new look at traditional Developmental Testing (DT). This proposed concept, called Integrated New Test Aircraft Capability (INTAC), involves using a high-fidelity, physical model based, generic structure simulation to certify the aircraft in a virtual environment, and using the aircraft to certify the simulation model. INTAC includes supporting, enhancing, augmenting, and ultimately automating traditional T&E and making this capability available at the pilot/engineer work area. INTAC promises to reinvent T&E by allowing test team members to do testing better, faster, cheaper, and safer. A high fidelity virtual air vehicle life cycle simulation capability, like INTAC, also gives the program manager better information on how to optimize the application of scarce future test dollars.
DTIC
Flight Simulation; Aircraft Performance; Flight Tests; Performance Tests

19960017361; 96N22952 Naval Air Warfare Center, Aircraft Div., Patuxent River, MD USA
E-6A Precipitation Static Assessments
Haines, Joel, Naval Air Warfare Center, USA; Clelland, Mike, Naval Air Warfare Center, USA; Whitaker, Mike, Naval Air Warfare Center, USA; DePasquale, William, Dual, Inc., USA; Lubosch, Bernd, Dual, Inc., USA; Sep. 27, 1995, pp. 33; In English; Also announced as 19960017361
Report No.(s): AD-A300960; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Electromagnetic Environmental Effects (E3) is a serious concern in today's Navy. As part of the E3 life-cycle effort,

P-Static Assessments of TACAMO Fleet aircraft being conducted. These assessments are used to monitor aircraft bonding integrity and maintenance procedures throughout the aircraft life-cycle. to minimize the problems associated with E3, aircraft bonding integrity must be maintained. A P-Static assessment is a quick-look method for finding bonding inconsistencies on the skin of the aircraft. Working hand in hand with the Fleet, problem areas can be identified and maintenance actions initiated to correct them. In the future, the P-Static assessments will be incorporated into the Enhanced Phase Maintenance (EPM) procedures. The P-Static assessments consist of power off testing along with dc bonding measurements of the static discharging system. The Fleet receives a brief report describing problems found during the assessment with recommended fixes. A database will be developed to document all problem areas and note any similarities between aircraft.
DTIC
Electronic Aircraft; Flight Tests; Aircraft Reliability; Life Cycle Costs

19960017585; 96N23163 Defence Science and Technology Organisation, Airframes and Engines Div., Melbourne, Australia
Orion P3C Horizontal Stabiliser Strain Survey and Corrosion Repair Development
Kaye, Robert H., Defence Science and Technology Organisation, Australia; Jones, R., Monash Univ., Australia; Hayes, Peter, Defence Science and Technology Organisation, Australia; Nov. 1995, pp. 50; In English; Also announced as 19960017585
Report No.(s): DSTO-TR-0196; AR-009-445; Copyright; Avail: Issuing Activity (DSTO Aeronautical and Maritime Research Lab., PO Box 4331, Melbourne, Victoria 3001, Australia), Hardcopy, Microfiche

This report presents results of an experimental investigation into the stress/strain field at the lower skin root region of the P3 horizontal stabilizer. Thermo-elastic and strain gauge methods were used with a bare structure P3 tail-plane loaded under laboratory conditions. Applied load was restricted to 60 percent of design limit by the nature of the test article and the loading rig. The results show that the stresses in the region of interest are predominantly spanwise in direction. Stress concentrations consisting typically of 30 percent in-plane shear stress and 70 percent spanwise stress were found to occur near the inboard corners of each of the three lower skin planks. The largest of these was found at the forward corner of the forward plank. The load transfer between the skin panels and the spars was found to take place over the inboard 100 mm of the stabilizer. A bonded repair to corrosion damage has been presented on the basis of the bonded joints in the repair being stronger than the surrounding parent material. This repair design is not in any way dependent on the stress/strain test results and falls within the requirements of the recently produced RAAF Engineering Standard C5033. The region has a complex stress distribution due to the stabilizer skin panels terminating at the fuselage. Laboratory investigation of this stress distribution has been brought about by the availability of a suitable test article and has provided valuable ancillary information.
Author
Stress-Strain Relationships; Skin (Structural Member); Stress Distribution; Bonded Joints; Horizontal Tail Surfaces; Aircraft Structures; Corrosion; P-3 Aircraft; Load Tests

19960017632; 96N23188 Virginia Univ., School of Engineering and Applied Science., Charlottesville, VA USA
NASA-UVA Light Aerospace Alloy and Structures Technology Program (LA2ST) ***Progress Report, 1 Jul. - 31 Dec. 1995***
Scully, John R., Virginia Univ., USA; Shiflet, Gary J., Virginia Univ., USA; Stoner, Glenn E., Virginia Univ., USA; Wert, John A., Virginia Univ., USA; Jan. 1996, pp. 168; In English; Also announced as 19960017632
Contract(s)/Grant(s): NAG1-745
Report No.(s): NASA-CR-200723; NAS 1.26:200723; UVA/528266/MSE96/119; No Copyright; Avail: CASI; A08, Hardcopy; A02, Microfiche

The NASA-UVA Light Aerospace Alloy and Structures Technology (LA2ST) Program was initiated in 1986 and continues with a high level of activity. The objective of the LA2ST Program is to conduct interdisciplinary graduate student research on the performance of next generation, lightweight aerospace alloys, composites and thermal gradient structures in collaboration with NASA-Langley researchers. Specific technical objectives are presented for each research project. We generally aim to produce relevant data and basic understanding of material mechanical response, environmental/corrosion behavior, and microstructure; new monolithic and composite alloys; advanced processing methods; new solid and fluid mechanics analyses; measurement and modeling advances; and a pool of educated graduate students for aerospace technologies. Three research areas are being actively investigated, including: (1) Mechanical and environmental degradation mechanisms in advanced light metals, (2) Aerospace materials science, and (3) Mechanics of materials for light aerospace structures.
Author
Aircraft Construction Materials; Aircraft Structures; Light Alloys; Spacecraft Construction Materials; Corrosion Resistance; Microstructure; Fracture Strength; Crack Propagation

19960017715; 96N23246 National Aerospace Lab., Tokyo, Japan
Simulation Study for a Fire Helicopter
Okuno, Yoshinori, National Aerospace Lab., Japan; Funabiki, Kohei, National Aerospace Lab., Japan; Saito, Shigeru,

National Aerospace Lab., Japan; Harada, Masashi, National Aerospace Lab., Japan; May 1995; ISSN 0389-4010, pp. 30; In Japanese; Also announced as 19960017715
Report No.(s): NAL-TR-1269; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The feasibility of the concept of using a helicopter to suppress fires in high-rise buildings was investigated by means of flight simulations and flight tests. The piloted simulations using a flight simulator were performed to investigate the safety and effectiveness of the fire fighting operation. The results demonstrated that nearly 90% of the discharged water can be hit on the target window at wind speeds of up to 7 m/s when the operator of the water discharge boom cooperates with the helicopter pilot. On the other hand, flight tests were conducted, in which the pilots performed hovering at a distance of 20 m from a real high-rise building and tried to keep the searchlight, which was mounted on the helicopter, fixed on a target window. It was pointed out that the fire helicopter should be equipped with an indicator of the distance to the building wall to improve the safety and efficiency during the fire fighting operation. The results of these experiments validated the feasibility of the fire helicopter and also clarified the problems to be solved to realize this concept.
Author
Helicopters; Fire Fighting; Flight Simulation

19960017815; 96N23340 National Aerospace Lab., Tokyo, Japan
Wing Rock of Delta Wings with an Analysis by the Phase Plane Method
Tate, Atsushi, National Aerospace Lab., Japan; Noda, Junichi, National Aerospace Lab., Japan; Yoshinaga, Takashi, National Aerospace Lab., Japan; Apr. 1995; ISSN 0389-4010, pp. 22; In Japanese; Also announced as 19960017815
Report No.(s): NAL-TR-1266; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Experimental investigation of the rolling motion of thin delta wings with varying sweep angles, using using a free-to-roll method was conducted. The boundaries of dynamically stable and unstable regions of the delta wings were obtained and compared with vortex symmetry and vortex burst curves on a plane of angle of attack of the roll axis vs. delta wing sweep angle. An analytical expression of the rolling moment using a phase plane method was found to be satisfactory for generating the limit cycle similar to the one experimentally obtained. Using this expression, energy transfer in the unsteady motion was calculated.
Author
Delta Wings; Thin Wings; Sweep Angle; Angle of Attack; Rolling Moments; Wing Oscillations

06
AIRCRAFT INSTRUMENTATION

Includes cockpit and cabin display devices; and flight instruments.

19960015616; 96N22424 Michigan Univ., Ann Arbor, MI USA
Computing range maps for low-altitude rotorcraft flight using computer vision and a hierarchical framework
LeBlanc, David J., Michigan Univ., USA; Jan. 01, 1994, pp. 203; In English; Also announced as 19960015616; No Copyright; Avail: Univ. Microfilms Order No. DA9500979, Hardcopy, Microfiche

Computer vision is one of many sensing modalities suited for obstacle detection onboard autonomous vehicles navigating in unknown or only partially known environments. For rotorcraft flying at very low altitudes, the use of image sequences from cameras mounted to the vehicle is desirable because vision is passive and provides rapid sensing of the environment with near infinite range and a wide field of view. Unfortunately, the processing of visual images to generate range maps of the nearby environment is computationally expensive, and currently cannot be performed in real time, even using relatively low-performance cameras. Furthermore, this problem will be exacerbated by the use of new high-performance cameras whose high resolution is necessary to provide the range map denseness required for safe navigation. We propose a hierarchical set of computations that guarantees that range maps based on feature points can be computed in real time. This is achieved by selecting, at regular intervals in time, a subset of features to track, such that the onboard computing resources are not exceeded. This is enabled and enhanced by an efficient multirate, multiresolutional approach to tracking features. A high level computation determines which features to track, then assigns each feature to one of many parallel slave processors, allocating a maximum amount of computing resources for the feature. Features are chosen to improve overall range map accuracy feature density, as required for the navigation objectives. Computations at the feature level select image data from a multirate and multiresolution image data structure to produce the best range estimates while using only the available computing resources. The result is a more efficient and more focused processing of image data, more control over the distribution of features in the range map, and a guarantee of real time computing performance.
Dissert. Abstr.
Computer Vision; Image Processing; Parallel Processing (Computers); Real Time Operation; Rangefinding; Optical Range Finders; Navigation Aids; Nap-Of-The-Earth Navigation

19960016374; 96N22180 NASA Ames Research Center, Moffett Field, CA USA

Human-Centered Aviation Automation: Principles and Guidelines

Billings, Charles E., NASA Ames Research Center, USA; Feb. 1996, pp. 222; In English; Also announced as 19960016374

Contract(s)/Grant(s): NAG2-810

Report No.(s): NASA-TM-110381; A-961056; NAS 1.15:110381; No Copyright; Avail: CASI; A10, Hardcopy; A03, Microfiche

This document presents principles and guidelines for human-centered automation in aircraft and in the aviation system. Drawing upon operational experience with highly automated aircraft, it describes classes of problems that have occurred in these vehicles, the effects of advanced automation on the human operators of the aviation system, and ways in which these problems may be avoided in the design of future aircraft and air traffic management automation. Many incidents and a few serious accidents suggest that these problems are related to automation complexity, autonomy, coupling, and opacity, or inadequate feedback to operators. An automation philosophy that emphasizes improved communication, coordination and cooperation between the human and machine elements of this complex, distributed system is required to improve the safety and efficiency of aviation operations in the future.

Author

Operator Performance; Complex Systems; Human Factors Engineering; Aircraft Accidents; Aircraft Safety; Automation

07
AIRCRAFT PROPULSION AND POWER

Includes prime propulsion systems and systems components, e.g., gas turbine engines and compressors; and onboard auxiliary power plants for aircraft.

19960015715; 96N21499 Washington Univ., Seattle, WA USA

Effects of rotation on impinging jets for turbine cooling

Aronstein, David Claude, Washington Univ., USA; Jan. 01, 1994, pp. 128; In English; Also announced as 19960015715; No Copyright; Avail: Univ. Microfilms Order No. DA9504590, Hardcopy, Microfiche

The effects of background rotation on impinging-jet flows are studied using flow visualization in a rotating water tank. Implications on impingement cooling of rotating turbine blades are inferred based on established properties of jets and the vortical structures they generate. The problem is divided into two parts, depending upon the orientation of the jet relative to the rotation axis. An axial jet is parallel to the rotation axis, while a transverse jet is perpendicular or at a substantial angle to the rotation axis. In the case of axial jets, all of the jet flow may be absorbed by Ekman layers on the boundaries, if the Rossby number is low. This will occur on the impingement plate directly in the path of the jet if the Rossby number is less than the square root of the Ekman number; or in an area surrounding the jet nozzle if the Rossby number is higher, up to approximately one. The first case is trivial. An analytical solution is presented for the second case, including the starting transient, and observations of the flow are made. At higher Rossby numbers, the near-field flow resembles that of an impinging jet in an inertial frame of reference, but a transition to columnar, rotation-dominated flow is observed within a few jet-diameters from the impingement point. Vortical structures are suppressed in this columnar regime so the implication is that the radial extent of effective cooling will be limited. Transverse jets in the same range of Rossby number also exhibit the transition to columnar Row within a few jet-diameters from the impingement point, and the radial flow on the impingement plate breaks up into a pair of planar wall currents proceeding outward in opposite directions. Vortical structures are changed from a radial pattern to a columnar pattern. An interesting asymmetry is observed. One of the wall currents is prone to separation because of a low pressure region close to the wall, while the other is very persistent due to a high pressure region along the wall. Implications on turbine cooling are discussed.

Dissert. Abstr.

Jet Impingement; Jet Flow; Turbine Blades; Wall Flow; Rotating Fluids

19960015803; 96N21587 Stanford Univ., CA USA

An adaptive optimal combustion control strategy

Padmanabhan, Krishnan Turlough, Stanford Univ., USA; 1994, pp. 189; In English; Also announced as 19960015803; No Copyright; Avail: Univ. Microfilms Order No. DA9508425, Hardcopy, Microfiche

A strategy for optimizing the performance of a laboratory-scale combustor with respect to volumetric heat release (increase) and pressure fluctuations (decrease) has been developed. This strategy utilizes (1) actuation techniques that simultaneously control volumetric heat release and pressure fluctuations; (2) sensing techniques that measure combustor performance; and (3) an adaptive optimal control strategy. Actuation techniques were chosen for their ability to alter the unsteady flow associated with the shear layer and recirculating flow regions of the combustor. Periodic spanwise forcing of the inlet boundary layer is used to reduce combustion-induced pressure fluctuations. Crossflow jets upstream of the inlet are used to control volumetric heat release. Sensing techniques were selected to measure or estimate the two performance parameters to be controlled. A fast-response piezoelectric pressure transducer measures the magnitude of the pressure fluctuations. A streamwise array of photodiodes measures light emission from the flame and enables estimation of volumetric heat release. Combustor performance is

explicitly defined in terms of a cost function that is a weighted combination of the rms pressure fluctuation level and the mean volumetric heat release. The control strategy performs an on-line minimization of the cost function by continuously seeking the optimal combination of actuator settings and subsequently maintaining the cost at a minimum when the combustor is subject to unknown inlet condition changes. The strategy has been experimentally tested with unknown inlet condition changes, such as flow disturbances, changes in equivalence ratio, and changes in inlet velocity. Sample optimization results show the control strategy to be capable of indirectly sensing these inlet condition changes and to be effective at finding the new optimal actuator settings that reminimize the cost function within thirty seconds to four minutes of large and small inlet condition changes.
Dissert. Abstr.
Adaptive Control; Combustion Chambers; Combustion Control; Optimal Control; Pressure Oscillations; Combustion Temperature

19960015938; 96N22129 NYMA, Inc., Brook Park, OH USA
An Experimental Determination of Losses in a 3-Port Wave Rotor ***Final Report***
Wilson, Jack, NYMA, Inc., USA; Feb. 1996, pp. 20; In English; Turbo Expo 1996, 10-13 Jun. 1996, Birmingham, England, USA; Also announced as 19960015938
Contract(s)/Grant(s): NAS3-27186; RTOP 505-62-75
Report No.(s): NASA-CR-198456; E-10079; NAS 1.26:198456; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Wave rotors, used in a gas turbine topping cycle, offer a potential route to higher specific power and lower specific fuel consumption. In order to exploit this potential properly, it is necessary to have some realistic means of calculating wave rotor performance, taking losses into account, so that wave rotors can be designed for good performance. This in turn requires a knowledge of the loss mechanisms. The experiment reported here was designed as a statistical experiment to identify the losses due to finite passage opening time, friction, and leakage. For simplicity, the experiment used a 3-port, flow divider, wave cycle, but the results should be applicable to other cycles. A 12 inch diameter rotor was used, with two different lengths, 9 inches and 18 inches, and two different passage widths, 0.25 inch and 0.54 inch, in order to vary friction and opening time. to vary leakage, moveable end-walls were provided so that the rotor to end-wall gap could be adjusted. The experiment is described, and the results are presented, together with a parametric fit to the data. The fit shows that there will be an optimum passage width for a given wave rotor, since, as the passage width increases, friction losses decrease, but opening-time losses increase, and vice-versa. Leakage losses can be made small at reasonable gap sizes.
Author
Rotors; Gas Turbines; Losses; Leakage; Energy Storage; Ports (Openings); Friction Factor

19960016115; 96N22153 NASA Lewis Research Center, Cleveland, OH USA
Macroscopic balance model for wave rotors
Welch, Gerard E., NASA Lewis Research Center, USA; Jan. 1996, pp. 22; In English; 34th Aerospace Sciences Meeting and Exhibit, 15-18 Jan. 1996, Reno, NV, USA; Sponsored by American Inst. of Aeronautics and Astronautics, USA; Also announced as 19960016115
Contract(s)/Grant(s): RTOP 505-62-75; DA Proj. IL1-61102-AH-45
Report No.(s): NASA-TM-107114; ARL-TR-925; AIAA Paper 96-0243; E-10022; NAS 1.15:107114; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A mathematical model for multi-port wave rotors is described. The wave processes that effect energy exchange within the rotor passage are modeled using one-dimensional gas dynamics. Macroscopic mass and energy balances relate volume-averaged thermodynamic properties in the rotor passage control volume to the mass, momentum, and energy fluxes at the ports. Loss models account for entropy production in boundary layers and in separating flows caused by blade-blockage, incidence, and gradual opening and closing of rotor passages. The mathematical model provides a basis for predicting design-point wave rotor performance, port timing, and machine size. Model predictions are evaluated through comparisons with CFD calculations and three-port wave rotor experimental data. A four-port wave rotor design example is provided to demonstrate model applicability. The modeling approach is amenable to wave rotor optimization studies and rapid assessment of the trade-offs associated with integrating wave rotors into gas turbine engine systems.
Author
Rotors; Balance; Mathematical Models; Gas Turbine Engines; Tradeoffs

19960016375; 96N22181 NASA Lewis Research Center, Cleveland, OH USA
An integration of the turbojet and single-throat ramjet
Trefny, C. J., NASA Lewis Research Center, USA; Benson, T. J., NASA Lewis Research Center, USA; Nov. 1995, pp. 14; In English; 1995 Airbreathing Propulsion Subcommittee Meeting, 5-9 Dec. 1995, Tampa, FL, USA, sponsored by the Joint Army-Navy-NASA-Air Force Interagency Propulsion Committee; Also announced as 19960016375
Contract(s)/Grant(s): RTOP 466-02-01
Report No.(s): NASA-TM-107085; E-9963; NAS 1.15:107085; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A turbine-engine-based hybrid propulsion system is described. Turbojet engines are integrated with a single-throat ramjet so as to minimize variable geometry and eliminate redundant propulsion components. The result is a simple, lightweight system that is operable from takeoff to high Mach numbers. Non-afterburning turbojets are mounted within the ramjet duct. They exhaust through a converging-diverging (C-D) nozzle into a common ramjet burner section. At low speed the ejector effect of the C-D nozzle aerodynamically isolates the relatively high pressure turbojet exhaust stream from the ramjet duct. As the Mach number increases, and the turbojet pressure ratio diminishes, the system is biased naturally toward ramjet operation. The common ramjet burner is fueled with hydrogen and thermally choked, thus avoiding the weight and complexity of a variable geometry, split-flow exhaust system. The mixed-compression supersonic inlet and subsonic diffuser are also common to both the turbojet and ramjet cycles. As the compressor face total temperature limit is approached, a two-position flap within the inlet is actuated, which closes off the turbojet inlet and provides increased internal contraction for ramjet operation. Similar actuation of the turbojet C-D nozzle flap completes the enclosure of the turbojet. Performance of the hybrid system is compared herein to that of the discrete turbojet and ramjet engines from takeoff to Mach 6. The specific impulse of the hybrid system falls below that of the non-integrated turbojet and ramjet because of ejector and Rayleigh losses. Unlike the discrete turbojet or ramjet however, the hybrid system produces thrust over the entire Mach number range. An alternate mode of operation for takeoff and low speed is also described. In this mode the C-D nozzle flap is deflected to a third position, which closes off the ramjet duct and eliminates the ejector total pressure loss.
Author
Convergent-Divergent Nozzles; Hybrid Propulsion; Ramjet Engines; Turbojet Engines; Throats

19960016953; 96N22560 NASA Lewis Research Center, Cleveland, OH USA
Quantification of Gear Tooth Damage by Optimal Tracking of Vibration Signatures
Choy, F. K., Akron Univ., USA; Veillette, R. J., Akron Univ., USA; Polyshchuk, V., Akron Univ., USA; Braun, M. J., Akron Univ., USA; Hendricks, R. C., NASA Lewis Research Center, USA; Mar. 1996, pp. 22; In English; 6th International Symposium on Transport Phenomena and Dynamics of Rotating Machinery, 25-29 Feb. 1996, Honolulu, HA, USA, Sponsored by the Pacific Center of Thermal Fluids Engineering and the US Turbo and Power Machinery Research Center; Also announced as 19960016953
Contract(s)/Grant(s): RTOP 242-20-06
Report No.(s): NASA-TM-107100; E-10029; NAS 1.15:107100; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This paper presents a technique for quantifying the wear or damage of gear teeth in a transmission system. The procedure developed in this study can be applied as a part of either an onboard machine health-monitoring system or a health diagnostic system used during regular maintenance. As the developed methodology is based on analysis of gearbox vibration under normal operating conditions, no shutdown or special modification of operating parameters is required during the diagnostic process. The process of quantifying the wear or damage of gear teeth requires a set of measured vibration data and a model of the gear mesh dynamics. An optimization problem is formulated to determine the profile of a time-varying mesh stiffness parameter for which the model output approximates the measured data. The resulting stiffness profile is then related to the level of gear tooth wear or damage. The procedure was applied to a data set generated artificially and to another obtained experimentally from a spiral bevel gear test rig. The results demonstrate the utility of the procedure as part of an overall health-monitoring system.
Author
Gear Teeth; Structural Vibration; Damage Assessment; Systems Health Monitoring; Time Dependence

19960017007; 96N22613 Cornell Univ., Ithaca, NY USA
An analytical and numerical study of passive enhancement of stability margin in axial flow compression systems
West, William E, Jr., Cornell Univ., USA; 1993, pp. 121; Also announced as 19960017007; No Copyright; Avail: Univ. Microfilms Order No. DA9401876, Hardcopy, Microfiche

The provision of adequate stability margin allowing maximum performance of axial flow compression systems is of great importance to the engine designer. It is shown that these systems initially lose stability to nonaxisymmetric disturbances as the throttle is closed past the peak of the steady performance curve. The growth of these disturbances may eventually lead to surge or rotating stall depending primarily on the plenum parameter. Passive control strategies based on variation of inlet and compressor geometry to inhibit nonaxisymmetric disturbance growth are evaluated. A parallel shear flow inlet distortion model is developed. The parameter representing reciprocal blade passage fluid inertia is found to significantly effect stability margin and steady performance losses for the case of constant amplitude distortion. In addition, a study of circumferential variation of reciprocal passage lag yields further improvements in distortion tolerance implying that stator stagger may be varied to minimize the compatibility problems associated with particular inlet and compressor combinations. A survey of numerical methods applied to the compression system analysis is presented. The limitations of Spectral and Pseudospectral methods in the representation of nonlinearity induced by the compressor characteristic is discussed. A Point Collocation method is developed that avoids these problems. The numerical method is validated by application to the calculation of equilibrium rotating

stall performance. The reciprocal passage lag parameter is shown to be influential in the determination of stall cell shape and steady performance. A two mode analysis is presented that gives good agreement with numerical results suggesting that the influence of reciprocal passage lag is primarily due to inhibition and phase shifting of higher harmonic stall cell components. An analysis is presented suggesting that stagnation of circumferential velocity in the inlet duct would inhibit nonaxisymmetric disturbances. A singularity distribution method based on thin-airfoil theory is developed to model the unsteady inlet flow due to a finite chord divider placed upstream of the compressor. The reciprocal passage lag parameter is again found to be influential in determination of optimum stall inhibiting divider chord and gap combinations. Dissert. Abstr.

Axial Flow; Compressors; Flow Distortion; Systems Analysis; Systems Stability; Tolerances (Mechanics); Unsteady Flow

19960017152; 96N22744 Allison Engine Co., Engineering Dept., Indianapolis, IN USA

Advanced Turbine Technology Applications Project (ATTAP) ***Annual Report, 1992***

Dec. 1993, pp. 106; In English; Also announced as 19960017152

Contract(s)/Grant(s): DEN3-336; DE-A101-90CE-50300; RTOP 778-32-21

Report No.(s): NASA-CR-195446; E-9499; NAS 1.26:195446; DOE/NASA/0336-5; EDR-17038; No Copyright; Avail: CASI; A06, Hardcopy; A02, Microfiche

The Advanced Turbine Technologies Application Project (ATTAP) is in the fifth year of a multi-year development program to bring the automotive gas turbine engine to a state at which industry can make commercialization decisions. Activities during the past year included reference powertrain design updates, test-bed engine design and development, ceramic component design, materials and component characterization, ceramic component process development and fabrication, ceramic component rig testing, and test-bed engine fabrication and testing. Engine design and development included mechanical design, combustion system development, alternate aerodynamic flow testing, and controls development. Design activities included development of the ceramic gasifier turbine static structure, the ceramic gasifier rotor, and the ceramic power turbine rotor. Material characterization efforts included the testing and evaluation of five candidate high temperature ceramic materials. Ceramic component process development and fabrication, with the objective of approaching automotive volumes and costs, continued for the gasifier turbine rotor, gasifier turbine scroll, extruded regenerator disks, and thermal insulation. Engine and rig fabrication, testing, and development supported improvements in ceramic component technology. Total test time in 1992 amounted to 599 hours, of which 147 hours were engine testing and 452 were hot rig testing.

Author

Ceramics; Automobile Engines; Engine Design; Engine Tests; Gas Turbine Engines; Design Analysis; Refractory Materials

19960017556; 96N23137 NASA Lewis Research Center, Cleveland, OH USA

Performance Benefits for Wave Rotor-Topped Gas Turbine Engines

Jones, Scott M., NASA Lewis Research Center, USA; Welch, Gerard E., Army Research Lab., USA; Mar. 1996, pp. 16; In English; 41st Turbo Expo 1996, 10-13 Jun. 1996, Birmingham, UK; Sponsored by American Society of Mechanical Engineers, USA; Also announced as 19960017556

Contract(s)/Grant(s): RTOP 505-62-75

Report No.(s): NASA-TM-107193; E-10168; NAS 1.15:107193; ARL-TR-1065; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The benefits of wave rotor-topping in turboshaft engines, subsonic high-bypass turbofan engines, auxiliary power units, and ground power units are evaluated. The thermodynamic cycle performance is modeled using a one-dimensional steady-state code; wave rotor performance is modeled using one-dimensional design/analysis codes. Design and off-design engine performance is calculated for baseline engines and wave rotor-topped engines, where the wave rotor acts as a high pressure spool. The wave rotor-enhanced engines are shown to have benefits in specific power and specific fuel flow over the baseline engines without increasing turbine inlet temperature. The off-design steady-state behavior of a wave rotor-topped engine is shown to be similar to a conventional engine. Mission studies are performed to quantify aircraft performance benefits for various wave rotor cycle and weight parameters. Gas turbine engine cycles most likely to benefit from wave rotor-topping are identified. Issues of practical integration and the corresponding technical challenges with various engine types are discussed.

Author

Turbofan Engines; Thermodynamic Cycles; Rotors; Turbomachinery; Turboshafts; Propulsion System Performance

19960017557; 96N23138 NASA Lewis Research Center, Cleveland, OH USA

Two-Dimensional Computational Model for Wave Rotor Flow Dynamics

Welch, Gerard E., Army Research Lab., USA; Mar. 1996, pp. 14; In English; 41st Turbo Expo 1996, 10-13 Jun. 1996, Birmingham, UK; Sponsored by American Society of Mechanical Engineers, USA; Also announced as 19960017557; Original contains 3 color illustrations

Contract(s)/Grant(s): RTOP 505-62-75

Report No.(s): NASA-TM-107192; E-10164; NAS

1.15:107192; ARL-TR-924; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A two-dimensional (theta,z) Navier-Stokes solver for multi-port wave rotor flow simulation is described. The finite-volume form of the unsteady thin-layer Navier-Stokes equations are integrated in time on multi-block grids that represent the stationary inlet and outlet ports and the moving rotor passages of the wave rotor. Computed results are compared with three-port wave rotor experimental data. The model is applied to predict the performance of a planned four-port wave rotor experiment. Two-dimensional flow features that reduce machine performance and influence rotor blade and duct wall thermal loads are identified. The performance impact of rounding the inlet port wall, to inhibit separation during passage gradual opening, is assessed.

Author

Rotor Dynamics; Multiblock Grids; Navier-Stokes Equation; Finite Volume Method; Two Dimensional Models; Computational Fluid Dynamics; Two Dimensional Flow; Gas Turbine Engines

08
AIRCRAFT STABILITY AND CONTROL

Includes aircraft handling qualities; piloting; flight controls; and autopilots.

19960015788; 96N21572 Stanford Univ., CA USA

Inverse and optimal control for aircraft maneuvers

Bryson, Arthur E., Stanford Univ., USA; Jan. 01, 1995, pp. 176; In English; Also announced as 19960015788; No Copyright; Avail: Univ. Microfilms Order No. DA9516871, Hardcopy, Microfiche

This dissertation determines control input histories for flying 3-D precision aerobatic and minimum time 180 deg turning maneuvers using inverse-control and optimization methods. Inverse control involves determining the input histories that produce the desired output histories. Choosing cylindrical coordinates for the position vector and spherical coordinates for the velocity vector yields very simple inverse solutions for most of the standard aerobatic maneuvers. The inverse solutions are nonlinear feedback laws i.e. one finds the controls as functions of the states and parameters of the maneuver. Two coordinate systems are used. The first treats motions along a horizontal helix, which includes barrel rolls, loops, Immelman turns, and split-S maneuvers. The second treats motions along a vertical helix, which includes steep turns, Chandelles, and wing-overs. In some cases, the inverse control histories or secondary output histories (e.g. normal load factor) may violate constraints, and/or the final states may be far from those desired. Optimal control methods may be used to modify the inverse control histories to satisfy the constraints at the expense of small deviations from the desired path. The dynamic model used has six states (three position, three velocity) with thrust, angle of attack, and bank angle as the control variables. There are bounds on the maximum angle of attack, thrust, and normal load factor. This dissertation also presents numerical results for loop, barrel roll, and minimum time turning maneuvers with three different sets of terminal conditions for an F-4H aircraft. The results of this study could be used in evaluation of pilot performance and in improving pilot training.

Dissert. Abstr.

Aircraft Maneuvers; Pilot Training; Pilot Performance; Angle of Attack; Thrust; Spherical Coordinates

19960016758; 96N22331 California Univ., Los Angeles, CA USA

A study of aeromechanical problems with active controls *Final Report, 15 Nov. 1987 - 30 Sep. 1994*

Friedmann, Peretz P., California Univ., USA; Sep. 30, 1994, pp. 4; In English; Also announced as 19960016758

Contract(s)/Grant(s): NAG2-477

Report No.(s): NASA-CR-200213; NAS 1.26:200213; No Copyright; Avail: CASI; A01, Hardcopy; A01, Microfiche

The principal accomplishments, under the grant were the development of the first sophisticated aeroelastic simulation of higher harmonic control (HHC) using time domain aerodynamics. The principal finding in this study was that vibration reduction in hingeless rotors using HHC is much more difficult than in articulated rotors. In addition, the first rotor/flexible fuselage model including HHC was developed. A four bladed hingeless rotor, with nonlinear coupled flap-lag-torsional dynamics was combined with a flexible fuselage represented by four bending and two torsional modes. The principal findings of this study were: vibration reduction at the hub due to HHC is not equivalent to acceleration reduction at specific locations in the flexible fuselage; simultaneous reduction of hub vibratory loads and fuselage acceleration requires a combination of 2, 3, 4, and 5/rev. pitch inputs in the rotating reference frame.

CASI

Rotary Wing Aircraft; Rigid Rotors; Vibration Damping; Vibratory Loads

19960017566; 96N23145 Lockheed Martin Engineering and Sciences Co., Hampton, VA USA

Performance Validation of Version 152.0 ANSER Control Laws for the F-18 HARV

Messina, Michael D., Lockheed Martin Engineering and Sciences Co., USA; Jan. 1996, pp. 176; In English; Also announced as 19960017566

Contract(s)/Grant(s): NAS1-19000; RTOP 505-68-30-05

Report No.(s): NASA-CR-198250; NAS 1.26:198250; No Copyright; Avail: CASI; A09, Hardcopy; A02, Microfiche

The Actuated Nose Strakes for Enhanced Rolling (ANSER) Control Laws were modified as a result of Phase 3 F/A-18 High Alpha Research Vehicle (HARV) flight testing.

The control law modifications for the next software release were designated version 152.0. The Ada implementation was tested in the Hardware-In-the-Loop (HIL) simulation and results were compared to those obtained with the NASA Langley batch Fortran implementation of the control laws which are considered the 'truth model.' This report documents the performance validation test results between these implementations for ANSER control law version 152.0.
Author
F-18 Aircraft; Ada (Programming Language); Research Aircraft; Longitudinal Control; Lateral Control

19960017716; 96N23247 National Aerospace Lab., Tokyo, Japan
H(sub infinity) Control for Aircraft Take off in Windshear
Wang, Lixin, National Aerospace Lab., Japan; Jul. 1995; ISSN 0389-4010, pp. 12; In English; Also announced as 19960017716
Report No.(s): NAL-TR-1273T; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Results are presented from an application of the most recently developed H(sub infinity) control design methodology to a stabilizing controller for an aircraft during take off in a windshear. The emphasis is on the formulation of H(sub infinity) optimal control synthesis problem in the state space. Simulation tests are performed in a six degree-of-freedom flight simulator for different windshear histories and different takeoff conditions, these results demonstrate that the controlled aircraft could take off in windshear safely, and the designed controller provides stability robustness to both external windshear disturbance as well as the model parameters variation with changes in flight conditions.
Author
Wind Shear; Optimal Control; Aircraft Control; Aircraft Stability

19960021251; 96N23230 National Aerospace Lab., Tokyo, Japan
Inverse Dynamics Control for Aircraft Take Off and Landing in Windshear
Wang, Lixin, National Aerospace Lab., Japan; Jul. 1995; ISSN 0389-4010, pp. 16; In English; Also announced as 19960021251
Report No.(s): NAL-TR-1274T; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This paper is concerned with the guidance of an aircraft flight in the presence of windshear. Based on following a nominal approach path relative to the ground, and subject to a minimum airspeed constraint, two guidance laws are synthesized by inverse dynamics technique for takeoff and landing respectively. Simulation tests are performed in a six-degree-of-freedom flight simulator for different windshear histories and different flight conditions, these results illustrate that the controlled aircraft could takeoff and safely and follow the nominal landing path approximately in windshear, and the designed robust controllers are insensitive to both external windshear disturbance and the model parameters variation with change in flight conditions. The different results between manual operations and autopilot control for against windshear are also compared.
Author
Wind Shear; Aircraft Control; Aircraft Landing; Flight Control; Inverse Kinematics

09
RESEARCH AND SUPPORT FACILITIES (AIR)

Includes airports, hangars and runways; aircraft repair and overhaul facilities; wind tunnels; shock tubes; and aircraft engine test stands.

19960015861; 96N21640 NASA Lewis Research Center, Cleveland, OH USA
Mach-6 Integrated Systems Tests of the NASA Lewis Research Center Hypersonic Tunnel Facility
Thomas, Scott R., NASA Lewis Research Center, USA; Woike, Mark R., NASA Lewis Research Center, USA; Pack, William D., NASA Lewis Research Center, USA; Jan. 01, 1996, pp. 18; In English; 1995 Airbreathing Propulsion Subcommittee Meeting, 5-8 Dec. 1995, Tampa, FL, USA, Sponsored by the Joint Army-Navy-NASA-Air Force Interagency Propulsion Committee; Also announced as 19960015861
Contract(s)/Grant(s): RTOP 505-70-62
Report No.(s): NASA-TM-107083; E-9960; NAS 1.15:107083; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This report documents the results for a series of 15 Integrated Systems Tests which were conducted at the NASA Lewis Research Center's Hypersonic Tunnel Facility (HTF) with test conditions simulating up to Mach-6 flight conditions. Facility stagnation conditions up to 3050 R and 1050 psia were obtained with typical test times of 20 to 45 sec. The HTF is a free-jet, blowdown propulsion test facility that can simulate up to Mach-7 flight conditions with true air composition. Mach-5, -6, and -7 facility nozzles, each with a 42-in. exit diameter, are available. The facility, without modifications, can accommodate models approximately 10 ft long. Nitrogen is heated using a graphite core induction heater, ambient oxygen is added downstream to produce simulated air. The combination of clean-air, large-scale, and Mach-7 capabilities is unique to the HTF. Reactivation of the HTF was accomplished between 1990 and May 1994. This activity included refurbishing the graphite heater, the steam generation plant, the gaseous oxygen system, and all control systems. All systems were checked out and recertified, and environmental systems were upgraded to meet current standards. The data systems were also upgraded to current standards and a communication link with NASA-wide computers was added. In

May 1994 a short-duration integrated systems test (approx. 2 sec) was conducted to verify facility operability. This test activity identified several modifications and corrections to the HTF which were required to improve overall facility performance. From the end of 1994 to April 1995 these items were completed, and the 3000-ft long, 30-in.-diam. steam supply line was insulated to improve system efficiency and allow operation in all weather conditions. Through May 1995 the test series which will be described in this report were conducted. During this activity, significant test experience was gained. The graphite storage heater was used at up to the maximum operating temperature of 5000 R, several improvements were made to various facility systems and test procedures, and some operational problems experienced in the past were resolved (i.e., elimination of water backflow during shutdown). The HTF was operated with significant run times for the first time since being reactivated, and for the first time in more than 20 yr. Overall this test program resulted in smooth, relatively trouble-free facility operation and served to successfully demonstrate the operating capability and reliability of the HTF.
Author
Hypersonic Wind Tunnels; System Effectiveness; Systems Integration; Flight Simulation; Test Facilities; Hypersonic Speed; Performance Tests

19960015885; 96N21663 NYMA, Inc., Brook Park, OH USA
High Response Dew Point Measurement System for a Supersonic Wind Tunnel *Final Report*
Blumenthal, Philip Z., NYMA, Inc., USA; Feb. 1996, pp. 16; In English; 42nd International Instrumentation Symposium, 5-9 May 1996, San Diego, CA, USA, Sponsored by the Aerospace Industries and Test Measurement Divisions of the Instrument Society of America; Also announced as 19960015885
Contract(s)/Grant(s): NAS3-27186; RTOP 505-62-82
Report No.(s): NASA-CR-198453; E-10111; NAS 1.26:198453; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A new high response on-line measurement system has been developed to continuously display and record the air stream dew point in the NASA Lewis 10 x 10 supersonic wind tunnel. Previous instruments suffered from such problems as very slow response, erratic readings, and high susceptibility to contamination. The system operates over the entire pressure level range of the 10 x 10 SWT, from less than 2 psia to 45 psia, without the need for a vacuum pump to provide sample flow. The system speeds up tunnel testing, provides large savings in tunnel power costs and provides the dew point input for the data-reduction subroutines which calculate test section conditions.
Author
Air Flow; Dew Point; Supersonic Wind Tunnels; On-Line Systems; Hygrometers; Real Time Operation

19960016875; 96N22504 George Washington Univ., Washington, DC USA
An automated maintenance management system for international airports
Khozeimeh, Issa D., George Washington Univ., USA; Jan. 01, 1994, pp. 297; In English; Also announced as 19960016875; No Copyright; Avail: Univ. Microfilms Order No. DA9512248, Hardcopy, Microfiche

This study was undertaken to develop and document the functional requirements for an automated maintenance management system (AMMS) suitable for international airports. The need for such a system was identified in 1990 by the Maintenance Committee of the Airport Association Council International (AACI). As an integral part of this research, an in-depth investigation was made concerning the availability, adaptability, and feasibility of modifying any commercially available AMMS that might have been developed for a similarly sized industry. to develop the detailed requirements for the maintenance management module, a formal questionnaire was created and distributed worldwide to international airports. The survey inquired about the automation level of their maintenance management efforts, the specific requirement for various modules, and management information reports. to validate the questionnaire, a pilot study was conducted by reviewing the questions during interviews with personnel at other U.S. airports. Consequently, the final questionnaire was modified and sent to international airports worldwide. The survey results were analyzed using descriptive statistical techniques including frequency distribution of responses; central tendency and dispersion measures such as mean, standard deviation, and range of criterion variable; and the maintenance management functional requirements developed. Analysis of the returned questionnaire identified several factors considered minimum requirements for the AMMS and for types of management information reports that would contribute to more efficient and effective operation. The respondents were also asked to provide additional comments concerning maintenance procedures at their facilities and possible improvement in particular areas. These participants were commonly concerned about management budgetary problems, staff and equipment management problems, inadequate computer systems and software for maintenance management applications, inadequate training on current management systems, inadequate level of standardized training for maintenance staff, imbalance of funds allocation, improper methods for evaluating existing equipment depreciation, and difficulties in upgrading maintenance management systems to interface with the older fragmented systems in place. The study also examined organizational structure of the Engineering and Maintenance Division of Washington Dulles International Airport and compared it with airports of similar size as well

as with other complex maintenance organizations. This examination revealed a need for realignment of various maintenance branches for more effective management in allocation of labor and resources.
Dissert. Abstr.
Airports; Management Information Systems; Surveys; Aircraft Maintenance; User Requirements

19960017102; 96N22698 Dayton Univ., OH USA
Design in aeroelastically scaled wind tunnel models using sensitivity-based parameter identification
French, Richard Mark, Dayton Univ., USA; 1993, pp. 220; In English; Also announced as 19960017102; No Copyright; Avail: Univ. Microfilms Order No. DA9333076, Hardcopy, Microfiche

Designing aeroelastically scaled wind tunnel models is a common task in flight vehicle development programs. Model design methods, however, often rely heavily on repeated analysis, experience and intuition. This work presents a procedure which uses numerical optimization techniques to automate the model design process. The proposed method offers many advantages over current ones. The model structure can be made as complex as necessary. Also, no unrealistic structural assumptions are required of the designer; any structure which can be discretized using a finite element approximation may be considered. The method uses readily available flexibility and modal response data from the full scale structure to size elements in the wind tunnel model structure automatically, eliminating repeated analysis. Several variations on the optimization-based approach are evaluated here using a low-aspect ratio fighter wing as a sample problem. Desired flutter characteristics are compared to those calculated for the final model design. to verify the numerical work, a sample model structure has been fabricated and tested using laser metrology techniques. Results of static and modal tests compare well with numerical predictions.
Dissert. Abstr.
Parameter Identification; Wind Tunnel Models; Finite Element Method; Aeroelasticity; Scale Models; Aircraft Models

19960017138; 96N22731 NASA Lewis Research Center, Cleveland, OH USA
Compact solar simulator with a small subtense angle and controlled magnification optics
Jensen, Kenneth A., Inventor, NASA Lewis Research Center, USA; Curren, Arthur N., Inventor, NASA Lewis Research Center, USA; Roman, Robert F., Inventor, NASA Lewis Research Center, USA; 11 Oct. 1994, pp. 48; In English; Also announced as 19960017138
Patent Info.: NASA-Case-LEW-15462-1; US-PATENT-APPL-SN-322862; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The present invention is directed to a method of simulating a pseudosun using a solar simulator. In the present invention the collector and lens of a lamp are designed to properly focus a plurality of light beams onto a segmented turning mirror. The path of light rays are traced from the lamp to the collector and then finally to the lens to control the solid and tangential magnification of the solar simulator. The segmented turning mirror is located at the focal point of the light beam and redirects the light into a vacuum chamber.
NASA
Lenses; Light Beams; Solar Simulators; Vacuum Chambers; Optical Equipment; Solar Collectors; Segmented Mirrors

19960017165; 96N22757 NYMA, Inc., Brook Park, OH USA
Enhancing the Capabilities of NASA Lewis' 8x6/9x15 Wind Tunnel Complex Through Drive System Modifications *Final Report*
Becks, Edward A., NYMA, Inc., USA; Feb. 1996, pp. 20; In English; 42nd International Instrumentation Symposium, 5-9 May 1996, San Diego, CA, USA, Sponsored by the Aerospace Industries and Test Measurement Div. of the Instrument Society of America; Also announced as 19960017165
Contract(s)/Grant(s): NAS3-27186; RTOP 505-62-82
Report No.(s): NASA-CR-198455; NAS 1.26:198455; E-10121; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This paper describes the conclusions to date derived from a series of activities that began in January 1995 to determine the potential power savings from running a three motor tunnel drive system on less than a full complement of three motors. This paper also discusses the now proven fact that one motor's minimum operating speed in this type of drive train is inherently slower than the minimum speed of three motors running at the same set of operating characteristics, as well as the benefits derived from this method of operation. The tunnel drive consists of three mechanically-coupled 29,000 HP wound rotor induction motors driving an axial flow compressor. Liquid rheostats are used to vary the impedance of the individual motor rotor circuits, thus varying the speed of the drive system. The resistance of the electrolyte in the rheostat tanks is inversely proportional to temperature. This paper discusses the manipulation of these factors to enhance the overall operation of the facility.
Author
Control Systems Design; Supersonic Wind Tunnels; Subsonic Wind Tunnels; Subsonic Flow; Wind Tunnel Drives; Mechanical Drives

19960017256; 96N22848 Aircraft Research Association Ltd., Bedford, UK
Reinventing T and E Internationally: The View from an Independent European Wind Tunnel
Green, J. E., Aircraft Research Association Ltd., UK; Oct. 1995, pp. 22; In English; ITEA 1995 Symposim, 2-5 Oct.

1995, Huntsville, AL, USA; Also announced as 19960017256
Report No.(s): ARA-TM-418; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The paper discusses the current status and future outlook for wind tunnel testing from an international perspective. More particularly, it presents the view of an independent, commercial, European wind tunnel operator. The technical, economic, and political factors influencing the future evolution of wind tunnel testing are considered and the challenge from computational fluid dynamics and the importance of continued investment in facility enhancement are highlighted. The present and future balance between supply and demand is discussed and efforts by the European research establishments to deal with problems of overcapacity are noted. Finally, an appeal is made to accelerate progress towards an end state, which the author believes to be inevitable, in which most wind tunnel testing is done in facilities which are free of government restraint and operate without government subsidy.

Author

Wind Tunnels; Technology Assessment; Technological Forecasting; Commerce; Europe; Constraints; Economic Factors; Politics

19960017571; 96N23150 Federal Aviation Administration, Civil Aeromedical Inst., Oklahoma City, OK USA

Qualification Guidelines for Personal Computer-Based Aviation Training Devices: Instrument Training *Final Report*

Williams, Kevin W., Federal Aviation Administration, USA; Feb. 1996, pp. 80; In English; Also announced as 19960017571

Report No.(s): DOT/FAA/AM-96/8; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

This is a report of the analytical development of qualification guidelines for personal computer-based aviation training devices (PCATD's) for use in instrument flight training. The report includes a task analysis of instrument flight tasks, along with baseline and task-specific guidelines that were developed as a result of the analysis. In addition a PCATD qualification tool is presented in the appendix for use in approving a Part 141 flight school curriculum that incorporates a PCATD for use as a ground training aid. The purpose of this paper is to provide the reader with an understanding of the process and reasoning behind the development of the PCATD qualification guidelines.

Author

Flight Training; Training Devices; Flight Instruments; Aircraft Control; Pilot Training; Pilot Performance

10
ASTRONAUTICS

Includes astronautics (general); astrodynamics; ground support systems and facilities (space); launch vehicles and space vehicles; space transportation; space communications, spacecraft communications, command and tracking; spacecraft design, testing and performance; spacecraft instrumentation; and spacecraft propulsion and power.

19960017101; 96N22697 Michigan Univ., Ann Arbor, MI USA

Optimal trajectories of reentry vehicle by serial and asymptotic matching

Han, Dongsuk, Michigan Univ., USA; 1993, pp. 115; In English; Also announced as 19960017101; No Copyright; Avail: Univ. Microfilms Order No. DA9332078, Hardcopy, Microfiche

With the advent of aeroassist technology for orbital transfers, it is of interest to have explicit optimal control laws for aerodynamic control during atmospheric passage of the space vehicle. Due to the nonlinearity of the equations of motion the exact analytic solutions are not available. Previously, analytic solutions were available only for the simplified problems, usually separated into two different regions namely, for the orbital flight and atmospheric flight. In this thesis, an analytic optimal control law uniformly valid in both regions is obtained. The differential equations for the optimal lift control coupled with the state equations can be fully integrated analytically by neglecting either the gravity or the aerodynamic force terms. A serial matching technique for the skip trajectory is proposed using the two controls obtained in these two separate regions. The switching condition between the two controls is derived and the solution is compared with the exact solution for a sample trajectory. This technique shows a good estimation of the cost function and provides a good physical explanation of the lift control as well. A matched asymptotic expansion method is used, coupled with the full knowledge of the physical properties of the exact integrals of motion, to derive the explicit optimal control laws for lift and bank modulation during the entry phase. The results from this control strategy by the matched asymptotic expansion method are compared with the exact numerical solutions for a variety of skip and glide entry trajectories. Its performance is excellent in the sense that the control exhibits the same characteristic behavior as the exact solution and it also leads to nearly the same cost function.

Dissert. Abstr.

Optimal Control; Transfer Orbits; Differential Equations; Reentry Vehicles; Trajectory Control; Asymptotic Series; Series Expansion; Equations of State; Lift

Subject Term Index

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