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AERONAUTICAL ENGINEERING

A CONTINUING BIBLIOGRAPHY WITH INDEXES

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Introduction

This issue of *Aeronautical Engineering, A Continuing Bibliography with Indexes* (NASA SP-7037) lists 257 reports, articles, and other documents recently announced in the NASA STI Database.

The coverage includes documents on the engineering and theoretical aspects of design, construction, evaluation, testing, operation, and performance of aircraft (including aircraft engines) and associated components, equipment, and systems. It also includes research and development in aerodynamics, aeronautics, and ground support equipment for aeronautical vehicles.

Each entry in the publication consists of a standard bibliographic citation accompanied, in most cases, by an abstract.

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The NASA CASI price code table, addresses of organizations, and document availability information are located at the back of this issue.

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Table of Contents

Records are arranged in categories 1 through 19, the first nine coming from the Aeronautics division of *STAR*, followed by the remaining division titles. Selecting a category will link you to the collection of records cited in this issue pertaining to that category.

01 Aeronautics 1

02 Aerodynamics 2

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

03 Air Transportation and Safety 20

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

04 Aircraft Communications and Navigation 24

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

05 Aircraft Design, Testing and Performance 28

Includes aircraft simulation technology.

06 Aircraft Instrumentation 34

Includes cockpit and cabin display devices; and flight instruments.

07 Aircraft Propulsion and Power 36

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

08 Aircraft Stability and Control 54

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

09 Research and Support Facilities (Air) 57

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

10 Astronautics 60

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.

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. 335)

OCTOBER 1996

01
AERONAUTICS

19960025318; 96N27390 Universal Technology Corp., Dayton, OH USA
Short-Term Propulsion and Power Development Analysis/ Assessment and Information Dissemination - (UTC) ***Final Report, 20 Jul. 1990 - 17 Jun. 1995***
Bethel, Howard, Universal Technology Corp., USA; Jun. 1995; 30p; In English
Contract(s)/Grant(s): F33615-90-C-2031; APPL
Report No.(s): AD-A302114; WL*-TR-95-2124; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This final report documents a summary of all technical accomplishments in performance of the contract 'Short-Term Propulsion & Power Development analysis Assessment and information Dissemination' (Contract Fl36l5-90C-2O3l) in support of the Aero Propulsion and power Directorate, Wright Laboratory, Air Force Materiel Command, Wright-Patterson AFB OH. Work accomplished under this contract was documented by Universal Technology Corporation (UTC) as Project SA77. Individual tasks were identified as Project 5477-01 through Project S477-50. The various tasks accomplished under this contract covered a wide variety of technical fields and areas of study. Examples include polymer battery research, analysis of the operation of the Compressor Research Facility, an assessment of the integrated High Performance Turbine Engine Technology (IHPTET) Program, a drag analysis of a variable flow ducted rocket (VFDR) using the Aerodynamic Preliminary Analysis System (APAS) computer codes, evaluation of Russian scramjet technology and test facilities, test and data reduction procedures for single tube MCH endothermic heat exchanger experiments, generation of residual stress in fan blades using the laser shock peening process, on-site support for the biannual turbine Steering committee meetings, the biannual Turbine Engine Technology Symposium, analyses on combined cycle engines, NASP flowpath thermal loads, and single crystal sheet material for high temperature environments.
DTIC
Aircraft Engines; Propulsion System Performance; Information Dissemination; Ducted Rocket Engines; Engine Design; Gas Turbine Engines

19960025415; 96N27453 Universal Energy Systems, Inc., Dayton, OH USA
Short-Term Propulsion and Power Development Analysis/Assessment and Information Dissemination ***Final Report, 8 Jun. 1990 - 5 Aug. 1995***
Jul. 1995; 41p; In English
Contract(s)/Grant(s): F33615-90-C-2086; APPL
Report No.(s): AD-A302115; UES-246; WL*-TR-95-2125; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This final report documents a summary of all technical work accomplished and information - obtained in performance of the contract 'Short-Term Propulsion and Power Development Analysis/Assessment and Information Dissemination' (Contract No. F33615-90-C-2086) in support of the Aero Propulsion and Power Directorate, - Wright Laboratory, Air Force Material Command, Wright Patterson AFB, OH. Work accomplished under this contract was documented by UES Inc. as Project 246. Individual tasks were identified as Project 246-001 through Project 246-047. The various tasks accomplished under this contract covered a wide variety of technical fields and areas of study. Examples of these areas include design and testing of heat transfer devices and combustors; design, fabrication and installation of an Ethylene Fuel System; design and engineering analysis of sonic and supersonic nozzles; - historical research on the History of the Gas Turbine Engine and the 75th Anniversary History of the Aero Propulsion and Power Directorate; and improve the design and efficiency of airbreathing propulsion systems.
DTIC
Fuel Systems; Sonic Nozzles; Supersonic Nozzles; Information Dissemination; Combustion Chambers; Gas Turbine Engines; Air Breathing Engines

19960027893; 96N29035 NASA Dryden Flight Research Center, Edwards, CA USA
A historical perspective of the YF-12A thermal loads and structures program
Jenkins, Jerald M., NASA Dryden Flight Research Center, USA; Quinn, Robert D., NASA Dryden Flight Research Center, USA; May 1996; 38p; In English
Contract(s)/Grant(s): RTOP 505-70-63
Report No.(s): NASA-TM-104317; NAS 1.15:104317;

H-2079; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Around 1970, the Y-F-12A loads and structures efforts focused on numerous technological issues that needed defining with regard to aircraft that incorporate hot structures in the design. Laboratory structural heating test technology with infrared systems was largely created during this program. The program demonstrated the ability to duplicate the complex flight temperatures of an advanced supersonic airplane in a ground-based laboratory. The ability to heat and load an advanced operational aircraft in a laboratory at high temperatures and return it to flight status without adverse effects was demonstrated. The technology associated with measuring loads with strain gages on a hot structure was demonstrated with a thermal calibration concept. The results demonstrated that the thermal stresses were significant although the airplane was designed to reduce thermal stresses. Considerable modeling detail was required to predict the heat transfer and the corresponding structural characteristics. The overall YF-12A research effort was particularly productive, and a great deal of flight, laboratory, test and computational data were produced and cross-correlated.

Author

Thermal Stresses; Structural Design; Histories; Aircraft Design; Grumman Aircraft; Supersonic Aircraft; Fighter Aircraft

19960029185; 96N29691 Logistics Management Inst., McLean, VA USA

Aircraft Battle Damage Repair (BDR): Analysis methodolology development requirements, Revision 1 ***Topical Report***

Kaplan, Bruce J., Logistics Management Inst., USA; Wallick, Jerry, Logistics Management Inst., USA; Nov. 1994; 69p; In English

Contract(s)/Grant(s): MDA903-90-C-0006

Report No.(s): AD-A304294; LMI-JL301RD1; JTCG/AS-94-D-002; No Copyright; Avail: Issuing Activity (Defense Technical Information Center (DTIC)), Microfiche

The JTCG/AS has been asked by the Office of the Secretary of Defense (OSD) to develop and implement a DoD accepted analysis methodology to address BDR aspects of aircraft system design, development, and acquisition. A BDR analysis methodology workshop was conducted to investigate feasible extensions of existing survivability and logistics support analysis methods to include BDR. No methodology obstacles were identified which would preclude such extensions. Industry and government organizations were visited and solicited for specific details on existing methodology limitations that must be addressed to incorporate appropriate BDR considerations. This report summarizes findings from these visits and interviews and documents a recommended roadmap with near-, mid-, and long-term tasks and projects for developing the required BDR analysis capability.

DTIC

Aircraft Maintenance; Damage Assessment; Logistics Management; Systems Engineering

19960029351; 96N29803 Air Force Scientific Advisory Board, Washington, DC USA

New world vistas: Air and space power for the 21st century, Summary Volume

McCall, Gene, Air Force Scientific Advisory Board, USA; Corder, John A., Air Force Scientific Advisory Board, USA; Jan. 30, 1996; 96p; In English

Report No.(s): AD-A303728; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

New World Vistas is a study about the Air Force. New World Vistas is documented in detail in over 2000 pages of monographs collected in 15 volumes. The study participants are listed, and abstracts of their work are contained in Appendix B.

DTIC

Aircraft; Space Technology Experiments; Computers

02
AERODYNAMICS

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

19960024181; 96N26736 National Renewable Energy Lab., Golden, CO USA

Effects of grit roughness and pitch oscillations on the LS(1)-0417MOD airfoil

Janiszewska, J. M., Ohio State Univ., USA; Ramsay, R. R., Ohio State Univ., USA; Hoffman, M. J., Ohio State Univ., USA; Gregorek, G. M., Ohio State Univ., USA; Jan. 1996; 158p; In English

Contract(s)/Grant(s): DE-AC36-83CH-10093

Report No.(s): NREL/TP-442-7819; DE96-000528; No Copyright; Avail: Issuing Activity (Department of Energy (DOE)), Microfiche

Horizontal axis wind turbine rotors experience unsteady aerodynamics due to wind shear when the rotor is yawed, when rotor blades pass through the support tower wake, and when the wind is gusting. An understanding of this unsteady behavior is necessary to assist in the calculations of rotor performance and loads. The rotors also experience performance degradation caused by surface roughness. These surface irregularities are due to the accumulation of insect debris, ice, and/or the aging process. Wind tunnel studies that examine both the steady and unsteady behavior of airfoils can help define pertinent flow phenomena, and the resultant data can be used to validate analytical computer codes. An LS(l)-0417MOD airfoil model was tested in The Ohio State

University Aeronautical and Astronautical Research Laboratory (OSU/AARL) 3x5 subsonic wind tunnel (3x5) under steady flow and stationary model conditions, as well as with the model undergoing pitch oscillations. to study the possible extent of performance loss due to surface roughness, a standard grit pattern (LEGR) was used to simulate leading edge contamination. After baseline cases were completed, the LEGR was applied for both steady state and model pitch oscillation cases. The Reynolds numbers for steady state conditions were 0.75, 1, 1.25, and 1.5 million, while the angle of attack ranged from -20 deg to +40 deg. With the model undergoing pitch oscillations, data were acquired at Reynolds numbers of 0.75, 1, 1.25, and 1.5 million, at frequencies of 0.6, 1.2, and 1.8 Hz. Two sine wave forcing functions were used, +/- 5.5% deg and +/- 10 deg, at mean angles of attack of 8 deg, 14 deg, and 20 deg. For purposes herein, any reference to unsteady conditions foil model was in pitch oscillation about the quarter chord.
DOE
Airfoils; Surface Roughness; Pitch (Inclination); Wind Turbines; Unsteady Aerodynamics; Airfoil Oscillations

19960024182; 96N26737 National Renewable Energy Lab., Golden, CO USA
Effects of surface roughness and vortex generators on the LS(1)-0417MOD airfoil
Reuss, R. L., Ohio State Univ., USA; Hoffman, M. J., Ohio State Univ., USA; Gregorek, G. M., Ohio State Univ., USA; Dec. 1995; 116p; In English
Contract(s)/Grant(s): DE-AC36-83CH-10093
Report No.(s): NREL/TP-442-6474; DE96-000495; No Copyright; Avail: Issuing Activity (Department of Energy, Washington, DC), Microfiche

An 18-inch constant-chord model of the LS(l)-0417MOD airfoil section was tested under two dimensional steady state conditions in the Ohio State University 7 x 10 Subsonic Wind Tunnel. The objective was to document section lift and moment characteristics under various model and air flow conditions. Surface pressure data was acquired at -60 deg. through +230 deg. geometric angles of attack, at a nominal 1 million Reynolds number. Cases with and without leading edge grit roughness were investigated. The leading edge roughness was used to simulate blade conditions in the field. Additionally, surface pressure data were acquired for Reynolds numbers of 1.5 and 2.0 million, with and without leading edge grit roughness; the angle of attack was limited to a -20 deg. to 40 deg. range. In general, results showed lift curve slope sensitivities to Reynolds number and roughness. The maximum lift coefficient was reduced as much as 29 percent by leading edge roughness. Moment coefficient showed little sensitivity to roughness beyond 50 deg. angle of attack, but the expected decambering effect of a thicker boundary layer with roughness did show at lower angles. Tests were also conducted with vortex generators located at the 30 percent chord location on the upper surface only, at 1 and 1.5 million Reynolds numbers, with and without leading edge grit roughness. In general, with leading edge grit roughness applied, the vortex generators restored 85 percent of the baseline level of maximum lift coefficient but with a more sudden stall break and at a higher angle of attack than the baseline.
DOE
Angle of Attack; Surface Roughness; Vortex Generators; Reynolds Number; Wind Tunnel Tests; Surface Roughness Effects; Wind Turbines; Rotor Blades (Turbomachinery)

19960024188; 96N26740 NASA Langley Research Center, Hampton,VA USA
Development of Semi-Span Model Test Techniques
Milholen, William E., II, North Carolina State Univ., USA; Chokani, Ndaona, North Carolina State Univ., USA; McGhee, Robert J., NASA Langley Research Center, USA; Jun 1996; 18p; In English; AIAA Applied Aerodynamics Conference, 17-20 Jun 1996, New Orleans, LA, USA
Contract(s)/Grant(s): NCC1-169
Report No.(s): NASA-TM-111583; NAS 1.15:111583; Copyright Waived (NASA); Avail: CASI; A03, Hardcopy; A01, Microfiche

A computational investigation was performed to support the development of a semispan model test capability in the NASA Langley Research Center's National Transonic Facility. This capability is desirable for the testing of advanced subsonic transport aircraft at full-scale Reynolds numbers. A state-of-the-art three-dimensional Navier-Stokes solver was used to examine methods to improve the flow over a semispan configuration. First, a parametric study is conducted to examine the influence of the stand-off height on the flow over the semispan model. It is found that decreasing the stand-off height, below the maximum fuselage radius, improves the aerodynamic characteristics of the semi-span model. Next, active sidewall boundary layer control techniques are examined. Juncture region blowing jets, upstream tangential blowing, and sidewall suction are found to improve the flow over the aft portion of the semispan model. Both upstream blowing and suction are found to reduce the sidewall boundary layer separation. The resulting near surface streamline patterns are improved, and found to be quite similar to the full-span results. Both techniques however adversely affect the pitching moment coefficient.
Author
Semispan Models; Transport Aircraft; Subsonic Speed; Navier-Stokes Equation; Boundary Layer Control; Aerodynamic Characteristics; Active Control; Boundary Layer Separation; Computational Fluid Dynamics; Grid Generation (Mathematics)

19960025202; 96N27283 Research Inst. for Advanced Computer Science, Moffett Field,CA USA

An all-at-once reduced Hessian SQP scheme for aerodynamic design optimization

Feng, Dan, Research Inst. for Advanced Computer Science, USA; Pulliam, Thomas H., NASA Ames Research Center, USA; Oct. 1995; 26p; In English

Contract(s)/Grant(s): NAS2-13721

Report No.(s): NASA-CR-201090; RIACS-TR-95-19; NAS 1.26:201090; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This paper introduces a computational scheme for solving a class of aerodynamic design problems that can be posed as nonlinear equality constrained optimizations. The scheme treats the flow and design variables as independent variables, and solves the constrained optimization problem via reduced Hessian successive quadratic programming. It updates the design and flow variables simultaneously at each iteration and allows flow variables to be infeasible before convergence. The solution of an adjoint flow equation is never needed. In addition, a range space basis is chosen so that in a certain sense the 'cross term' ignored in reduced Hessian SQP methods is minimized. Numerical results for a nozzle design using the quasi-one-dimensional Euler equations show that this scheme is computationally efficient and robust. The computational cost of a typical nozzle design is only a fraction more than that of the corresponding analysis flow calculation. Superlinear convergence is also observed, which agrees with the theoretical properties of this scheme. All optimal solutions are obtained by starting far away from the final solution.

Author

Aerodynamics; Design Analysis; Differential Equations; Nonlinear Programming; Nozzle Design; Quadratic Programming; Mathematical Programming

19960025271; 96N27344 Old Dominion Univ., Dept. of Mechanical Engineering., Norfolk, VA USA

Formulation for Simultaneous Aerodynamic Analysis and Design Optimization

Hou, G. W., Old Dominion Univ., USA; Taylor, A. C., III, Old Dominion Univ., USA; Mani, S. V., Old Dominion Univ., USA; Newman, P. A., National Aeronautics and Space Administration. Langley Research Center, USA; Aug. 1993; 46p; In English; Second; U.S. National Congress on Computational Mechanics, 16-18 Aug. 1993, Washington, DC, USA

Contract(s)/Grant(s): NAG1-1265

Report No.(s): NASA-CR-201036; NAS 1.26:201036; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An efficient approach for simultaneous aerodynamic analysis and design optimization is presented. This approach does not require the performance of many flow analyses at each design optimization step, which can be an expensive procedure. Thus, this approach brings us one step closer to meeting the challenge of incorporating computational fluid dynamic codes into gradient-based optimization techniques for aerodynamic design. An adjoint-variable method is introduced to nullify the effect of the increased number of design variables in the problem formulation. The method has been successfully tested on one-dimensional nozzle flow problems, including a sample problem with a normal shock. Implementations of the above algorithm are also presented that incorporate Newton iterations to secure a high-quality flow solution at the end of the design process. Implementations with iterative flow solvers are possible and will be required for large, multidimensional flow problems.

Author

Computational Fluid Dynamics; Design Analysis; Optimization

19960025637; 96N27641 NASA Ames Research Center, Moffett Field,CA USA

ZETA 2 Code for Determining the Flow Around Multielement and Deformable Airfoils

Wang, C. M., Georgia Inst. of Tech., USA; Wu, J. C., Georgia Inst. of Tech., USA; McAlister, Kenneth W., Army Aviation Systems Command, USA; Tung, C., Army Aviation Systems Command, USA; Jul. 1995; 52p; In English

Contract(s)/Grant(s): RTOP 505-59-87

Report No.(s): NASA-TP-3567; A-950064; NAS 1.60:3567; USAATCOM-TR-95-A-004; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

The present NASA-University consortium involves research on the development of an efficient and accurate numerical procedure to treat general unsteady viscous flows. of particular interest in the present work are flows relevant to helicopter rotors. Special efforts are centered on new concepts applied to the rotorblade configuration on the helicopter to improve the rotor aerodynamic characteristics. Among them, airfoil dynamic stall and the aerodynamic noise of vortex-airfoil interactions were extensively studied by using the numerical procedure. The procedure, based on the formulation of an integral representation of the velocity vector and the vorticity transport equation, is used to solve incompressible Navier-Stokes flows. ZETA, a computer code developed at Georgia Tech, is used as the basis for the development of an extended computer code, ZETA 2. The ZETA code is very efficient in the treatment of single, solid airfoils undergoing arbitrary motions. The extended ZETA 2 can treat the two-element airfoil, the deformable airfoil, and vortex-airfoil interaction. New concepts aimed at improving the helicopter maneuverability and susceptibility are studied, including the slatted airfoil and the deformable airfoil. Numerical results obtained by using the ZETA 2 code show that both slatted and deformable airfoils reduce the dynamic stall and that the deformable airfoil reduces the blade-vortex interaction (BVI) noise.

Author

Blade Slap Noise; Blade-Vortex Interaction; Rotary Wings;

Rotor Aerodynamics; Leading Edge Slats; Applications Programs (Computers); Computational Fluid Dynamics; Aerodynamic Stalling; Flexible Wings

19960025648; 96N27648 NASA Langley Research Center, Hampton,VA USA

Experimental Surface Pressure Data Obtained on 65 deg Delta Wing Across Reynolds Number and Mach Number Ranges, Volume 1, Sharp Leading Edge

Chu, Julio, NASA Langley Research Center, USA; Luckring, James M., NASA Langley Research Center, USA; Feb. 1996; 34p; In English

Contract(s)/Grant(s): RTOP 505-59-54-01

Report No.(s): NASA-TM-4645-Vol-1; L-17411A; NAS 1.15:4645-Vol-1; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An experimental wind tunnel test of a 65 deg delta wing model with interchangeable leading edges was conducted in the Langley National Transonic Facility (NTF). The objective was to investigate the effects of Reynolds and Mach numbers on slender-wing leading-edge vortex flows with four values of wing leading-edge bluntness. Experimentally obtained pressure data are presented without analysis in tabulated and graphical formats across a Reynolds number range of 6 x 10(exp 6) to 36 x 10(exp 6) at a Mach number of 0.85 and across a Mach number range of 0.4 to 0.9 at a Reynolds number of 6 x 10(exp 6). Normal-force and pitching-moment coefficient plots for these Reynolds number and Mach number ranges are also presented.

Author

Delta Wings; Mach Number; Reynolds Number; Wind Tunnel Tests; Sharp Leading Edges; Angle of Attack

19960025651; 96N27651 Air Force Inst. of Tech., Wright-Patterson AFB, OH USA

Numerical Simulation of Dynamic-Stall Suppression by Tangential Blowing

Towne, Matthew Carroll, Air Force Inst. of Tech., USA; Jan. 01, 1994; 154p; In English; No Copyright; Avail: Univ. Microfilms Order No. DA9503651, Hardcopy, Microfiche

The use of nearly tangential blowing to suppress the dynamic stall of a pitching airfoil is investigated numerically. The laminar two-dimensional, compressible Navier-Stokes equations are solved time-accurately using a Beam-Warming algorithm. A slot is located at four different positions along the surface of a NACA 0015 airfoil and air is injected in a nearly tangential sense along the upper surface. Suction control is also employed at one of these slot locations to directly compare with tangential-blowing control. Solution sensitivity to grid refinement, time-step size, numerical smoothing, and initial conditions is investigated at a Reynolds number of 24,000. Initial-condition and initial-airfoil-acceleration effects are analyzed for various pitch rates. Compressibility of solutions is investigated for a free-stream Mach number of 0.2. Numerical simulation uncertainties of jet-orientation angle and jet velocity profile are investigated. Studies are conducted to establish the effects of slot position slot width, blowing-initiation angle, blowing velocity, pulsed blowing and blowing at different pitch rates.

Dissert. Abstr.

Digital Simulation; Navier-Stokes Equation; Aerodynamic Stalling; Blowing

19960026082; 96N27828 High Technology Corp., Hampton,VA USA

Turbulent Wall-Pressure Fluctuations: New Model for Off-Axis Cross-Spectral Density

Singer, Bart A., High Technology Corp., USA; Mar. 1996; 16p; In English

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

Report No.(s): NASA-CR-198297; NAS 1.26:198297; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Models for the distribution of the wall-pressure under a turbulent boundary layer often estimate the coherence of the cross-spectral density in terms of a product of two coherence functions. One such function describes the coherence as a function of separation distance in the mean-flow direction, the other function describes the coherence in the cross-stream direction. Analysis of data from a large-eddy simulation of a turbulent boundary layer reveals that this approximation dramatically underpredicts the coherence for separation directions that are neither aligned with nor perpendicular to the mean-flow direction. These models fail even when the coherence functions in the directions parallel and perpendicular to the mean flow are known exactly. A new approach for combining the parallel and perpendicular coherence functions is presented. The new approach results in vastly improved approximations for the coherence.

Author

Wall Pressure; Pressure Distribution; Turbulent Boundary Layer; Mathematical Models

19960026256; 96N27964 NASA Ames Research Center, Moffett Field, CA USA

Flight evaluation of advanced controls and displays for transition and landing on the NASA V/STOL systems research aircraft

Franklin, James A., NASA Ames Research Center, USA; Stortz, Michael W., NASA Ames Research Center, USA; Borchers, Paul F., NASA Ames Research Center, USA; Moralez, Ernesto, III, NASA Ames Research Center, USA; Apr. 1996; 124p; In English

Contract(s)/Grant(s): RTOP 505-68-33

Report No.(s): NASA -TP-3607; NAS 1.60:3607; A-961333; No Copyright; Avail: CASI; A06, Hardcopy; A02, Microfiche

Flight experiments were conducted on Ames Research Center's V/STOL Systems Research Aircraft (VSRA) to

assess the influence of advanced control modes and head-up displays (HUD's) on flying qualities for precision approach and landing operations. Evaluations were made for decelerating approaches to hover followed by a vertical landing and for slow landings for four control/display mode combinations: the basic YAV-8B stability augmentation system; attitude command for pitch, roll, and yaw; flightpath/acceleration command with translational rate command in the hover; and height-rate damping with translational-rate command. Head-up displays used in conjunction with these control modes provided flightpath tracking/pursuit guidance and deceleration commands for the decelerating approach and a mixed horizontal and vertical presentation for precision hover and landing. Flying qualities were established and control usage and bandwidth were documented for candidate control modes and displays for the approach and vertical landing. Minimally satisfactory bandwidths were determined for the translational-rate command system. Test pilot and engineer teams from the Naval Air Warfare Center, the Boeing Military Airplane Group, Lockheed Martin, McDonnell Douglas Aerospace, Northrop Grumman, Rolls-Royce, and the British Defense Research Agency participated in the program along with NASA research pilots from the Ames and Lewis Research Centers. The results, in conjunction with related ground-based simulation data, indicate that the flightpath/longitudinal acceleration command response type in conjunction with pursuit tracking and deceleration guidance on the HUD would be essential for operation to instrument minimums significantly lower than the minimums for the AV-8B. It would also be a superior mode for performing slow landings where precise control to an austere landing area such as a narrow road is demanded. The translational-rate command system would reduce pilot workload for demanding vertical landing tasks aboard ship and in confined land-based sites.
Author
V/STOL Aircraft; Research Aircraft; Vertical Landing; Flight Control; Aircraft Control; Flight Tests; Head-Up Displays

19960026362; 96N28036 National Renewable Energy Lab., Golden, CO USA
Effects of grit roughness and pitch oscillations on the S809 airfoil
Ramsay, R. Reuss, Ohio State Univ., USA; Hoffman, M. J., Ohio State Univ., USA; Gregorek, G. M., Ohio State Univ., USA; Dec. 1995; 165p; In English
Contract(s)/Grant(s): DE-AC36-83CH-10093
Report No.(s): NREL/TP-442-7817; DE96-000503; No Copyright; Avail: CASI; A08, Hardcopy; A02, Microfiche

An S809 airfoil model was tested in The Ohio State University Aeronautical and Astronautical Research Laboratory (OSU/AARL) 3 x 5 subsonic wind tunnel under steady flow and stationary model conditions, and also with the model undergoing pitch oscillations. to study the possible extent of performance loss due to surface roughness, a standard grit pattern (LEGR) was developed to simulate leading edge contamination. After baseline cases were completed, the LEGR was applied for both steady state and model pitch oscillation cases. The Reynolds numbers for steady state conditions were 0.75, 1, 1.25, and 1.5 million, while the angle of attack ranged from -20, to +40 degrees. With the model undergoing pitch oscillations, data were acquired at Reynolds numbers of 0.75, 1, 1.25, and 1.4 million, at frequencies of 0.6, 1.2, and 1.8 Hz. Two sine wave forcing functions were used; +/-5.5 degrees and +/-10 degrees, at mean angles of attack of 8 degrees, 14 degrees, and 20 degrees. For purposes herein, any reference to unsteady conditions means the model was in pitch oscillation about the quarter chord. In general, the unsteady maximum lift coefficient was from 4% to 86% higher than the steady state maximum lift coefficient, and variation in the quarter chord pitching moment coefficient magnitude was from -83% to 195% relative to steady state values at high angles of attack. These findings indicate the importance of considering the unsteady flow behavior occurring in wind turbine operation to obtain accurate load estimates.
DOE
Airfoils; Aerodynamic Coefficients; Pitching Moments; Surface Roughness; Wind Tunnel Tests

19960026419; 96N28069 NYMA, Inc., Brook Park, OH USA
Cessation of Screech in Underexpanded Jets *Final Report*
Raman, Ganesh, NYMA, Inc., USA; Apr. 1996; 34p; In English; 2nd; Aeroacoustics Conference, 6-8 May 1996, State College , PA, USA; Sponsored by American Inst. of Aeronautics and Astronautics, USA
Contract(s)/Grant(s): NAS3-27186; RTOP 505-62-52
Report No.(s): NASA-CR-198475; NAS 1.26:198475; AIAA Paper 96-1719; E-10184; Copyright Waived (NASA); Avail: CASI; A03, Hardcopy; A01, Microfiche

In significantly underexpanded jets, screech inherently ceases to exist. This paper studies screech cessation in a supersonic rectangular jet and provides an explanation of its occurrence. Experimental data are presented for fully expanded jet Mach numbers ranging from 1.1 to 1.9. Screech becomes unsteady beyond jet Mach 1.65 and ceases to exist beyond jet Mach 1.75. Although screech cessation has been observed previously, the reason for its occurrence has remained a mystery. to resolve this mystery, this paper examines three suspects (1) Tam's theory of a frequency mismatch between screech tones and the band of the most amplified jet instability waves. (2) the notion that Mach disk formation disrupts the shock cell structure and weakens the screech producing shocks, and (3) the idea that acoustic feedback and receptivity diminish at high levels of underexpansion. A thorough interrogation of experimental data shows that (1) is not the main cause of screech cessation here although it is a valid theory for hot jets, (2) plays and insignificant role, and (3) appears to have been largely responsible for screech cessa-

tion. Cessation occurs because feedback to the jet lip is diminished due to excessive expansion of the jet boundary. Further, since the jet lip now scatters low intensity sound, the end result is poor receptivity at the initial shear layer. this theory is substantiated by the reactivation of screech when the nozzle lip thickness is made larger than the expanded jet boundary. Finally, increasing lip thickness is seen to produce a systematic shift (to higher jet Mach numbers) of the onset of cessation. The results of this study are of direct relevance to the sonic fatigue problem in aircraft structures, because understanding screech helps prevent such damage.
Author
Acoustic Fatigue; Supersonic Jet Flow; Mach Number; Boundary Layer Thickness; Static Pressure

19960026432; 96N28082 NASA Ames Research Center, Moffett Field, CA USA
Full-scale S-76 rotor performance and loads at low speeds in the NASA Ames 80- by 120-Foot Wind Tunnel, Vol. 1
Shinoda, Patrick M., Army Aviation Systems Command, USA; Apr. 1996; 248p; In English
Contract(s)/Grant(s): RTOP 505-59-36
Report No.(s): NASA-TM-110379; NAS 1.15:110379; A-960974; USAATCOM-TR-96-A-004; No Copyright; Avail: CASI; A11, Hardcopy; A03, Microfiche

A full-scale helicopter rotor test was conducted in the NASA Ames 80- by 120-Foot Wind Tunnel with a four-bladed S-76 rotor system. Rotor performance and loads data were obtained over a wide range of rotor shaft angles-of-attack and thrust conditions at tunnel speeds ranging from 0 to 100 kt. The primary objectives of this test were (1) to acquire forward flight rotor performance and loads data for comparison with analytical results; (2) to acquire S-76 forward flight rotor performance data in the 80- by 120-Foot Wind Tunnel to compare with existing full-scale 40- by 80-Foot Wind Tunnel test data that were acquired in 1977; (3) to evaluate the acoustic capability of the 80- by 120- Foot Wind Tunnel for acquiring blade vortex interaction (BVI) noise in the low speed range and compare BVI noise with in-flight test data; and (4) to evaluate the capability of the 80- by 120-Foot Wind Tunnel test section as a hover facility. The secondary objectives were (1) to evaluate rotor inflow and wake effects (variations in tunnel speed, shaft angle, and thrust condition) on wind tunnel test section wall and floor pressures; (2) to establish the criteria for the definition of flow breakdown (condition where wall corrections are no longer valid) for this size rotor and wind tunnel cross-sectional area; and (3) to evaluate the wide-field shadowgraph technique for visualizing full-scale rotor wakes. This data base of rotor performance and loads can be used for analytical and experimental comparison studies for full-scale, four-bladed, fully articulated rotor systems. Rotor performance and structural loads data are presented in this report.
Author
Blade Slap Noise; Blade-Vortex Interaction; Rotary Wings; Wind Tunnel Tests; Angle of Attack; Rotor Aerodynamics; Aeroacoustics; Full Scale Tests; Low Speed; Helicopter Performance; Test Chambers; Hovering

19960026632; 96N28201 National Renewable Energy Lab., Golden, CO USA
Effects of grit roughness and pitch oscillations on the S801 airfoil
Ramsay, R. R., Ohio State Univ., USA; Hoffman, M.J., Ohio State Univ., USA; Gregorek, G. M., Ohio State Univ., USA; Jan. 1996; 158p; In English
Contract(s)/Grant(s): AC36-83CH10093
Report No.(s): NREL/TP-442-7818; DE96-000517; No Copyright; Avail: CASI; A08, Hardcopy; A02, Microfiche

Horizontal axis wind turbine rotors experience unsteady aerodynamics due to wind shear when the rotor is yawed, when rotor blades pass through the support tower wake, and when the wind is gusting. An understanding of this unsteady behavior is necessary to assist in the calculation of rotor performance and loads. The rotors also experience performance degradation due to surface roughness. These surface irregularities are due to the accumulation of insect debris, ice, and the aging process. Wind tunnel studies that examine both the steady and unsteady behavior of airfoils can help define pertinent flow phenomena, and the resultant data can be used to validate analytical computer codes. A S801 airfoil model was tested in The Ohio State University Aeronautical and Astronautical Research Laboratory (OSU/AARL) 3x5 subsonic wind tunnel (3x5) under steady flow and stationary model conditions, as well as with the model undergoing pitch oscillations. to study the possible extent of performance loss due to surface roughness, a standard grit pattern (LEGR) was used to simulate leading edge contamination. After baseline cases were completed, the LEGR was applied for both steady state and model pitch oscillation cases. The Reynolds numbers used for steady state conditions were 0.75, 1, 1.25, and 1.5 million, while the angle of attack ranged from -20(degrees) to +40(degrees). With the model undergoing pitch oscillations, data were acquired at Reynolds numbers of 0.75, 1, 1.25, and 1.4 million, at frequencies of 0.6, 1.2, and 1.8 Hz. Two sine wave forcing functions were used, (plus minus) 5.5 (degrees)and (plus minus) 10(degrees), at mean angles of attack of 8(degrees) 14(degrees) and 20(degrees) For purposes herein, any reference to unsteady conditions means that the airfoil model was in pitch oscillation about the quarter chord.
DOE
Airfoils; Pitching Moments; Surface Roughness Effects; Wind Tunnel Tests; Steady Flow; Aerodynamic Coefficients; Aerodynamic Characteristics; Grit

19960026700; 96N28263 National Renewable Energy Lab., Golden, CO USA

A study of pitch oscillation and roughness on airfoils used for horizontal axis wind turbines

Gregorek, G. M., Ohio State Univ., USA; Hoffmann, M. J., Ohio State Univ., USA; Ramsay, R. R., Ohio State Univ., USA; Janizewska, J. M., Ohio State Univ., USA; Dec. 1995; 22p; In English

Contract(s)/Grant(s): AC36-83CH10093

Report No.(s): NREL/TP-442-7386; DE96-000504; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Under subcontract XF-1-11009-3 the Ohio State University Aeronautical and Astronautical Research Laboratory (OSU/AARL) with the National Renewable Energy Laboratory (NREL) developed an extensive database of empirical aerodynamic data. These data will assist in the development of analytical models and in the design of new airfoils for wind turbines. to accomplish the main objective, airfoil models were designed, built and wind tunnel tested with and without model leading edge grit roughness (LEGR). LEGR simulates surface irregularities due to the accumulation of insect debris, ice, and/or the aging process.

DOE

Aerodynamics; Wind Turbines; Leading Edges; Airfoils; Surface Roughness

19960026701; 96N28264 National Renewable Energy Lab., Golden, CO USA

Effects of grit roughness and pitch oscillations on the LS(1)-0421MOD airfoil

Reuss, R. L., Ohio State Univ., USA; Hoffman, M. J., Ohio State Univ., USA; Gregorek, G. M., Ohio State Univ., USA; Dec. 1995; 147p; In English

Contract(s)/Grant(s): DE-AC36-83CH-10093

Report No.(s): NREL/TP-442-6473; DE96-000493; No Copyright; Avail: CASI; A07, Hardcopy; A02, Microfiche

An LS(1)-0421 MOD airfoil model was tested in The Ohio State University Aeronautical and Astronautical Research Laboratory (OSU/AARL) 3 x 5 subsonic wind tunnel 3 x 5 under steady flow and stationary model conditions, and also with the model undergoing pitch oscillations. In order to study the possible extent of performance loss due to surface roughness, a leading edge grit roughness (LEGR) pattern was developed to simulate leading edge contamination. After baseline cases were completed, the LEGR was applied for both steady state and model pitch oscillation cases. The Reynolds numbers for steady state conditions were 0.75, 1, and 1.25 million, while the angle of attack ranged from -10 deg. to +40 deg. With the model undergoing pitch oscillations, data was acquired at Reynolds numbers of 0.75, 1, 1.25, and 1.5 million, at frequencies of 0.6, 1.2, and 1.8 Hz. Two sine wave forcing functions were used; +/- 5.5 deg. and +/-10 deg., at mean angles of attack of 8 deg., 14 deg., and 20 deg. For this report, unsteady conditions refer to the model in pitch oscillation. In general, the maximum unsteady lift coefficient was from 10% to 50% higher than the steady state maximum lift coefficient. Variation in the quarter chord pitching moment coefficient was nearly two times greater than steady state values at high angles of attack. These findings indicate the importance of considering the unsteady flow behavior occurring in wind turbine operation for accurate load estimates.

DOE

Airfoils; Angle of Attack; Subsonic Wind Tunnels; Leading Edges; Pitching Moments; Steady Flow

19960026762; 96N28322 NASA Ames Research Center, Moffett Field, CA USA

Full-Scale S-76 Rotor Performance and Loads at Low Speeds in the NASA Ames 80- by 120-Foot Wind Tunnel, Volume 2

Shinoda, Patrick M., Army Aviation Systems Command, USA; 1996; 944p; In English

Contract(s)/Grant(s): RTOP 505-59-36

Report No.(s): NASA-TM-110379-Vol-2; NAS 1.15:110379-Vol-2; USAATCOM-TP-96-A-004-Vol-2; A-960974; No Copyright; Avail: CASI; A99, Hardcopy; A10, Microfiche

A full-scale helicopter rotor test was conducted in the NASA Ames 80- by 120-Foot Wind Tunnel with a four-bladed S-76 rotor system. Rotor performance and loads data were obtained over a wide range of rotor shaft angles-of-attack and thrust conditions at tunnel speeds ranging from 0 to 100 kt. The primary objectives of this test were (1) to acquire forward flight rotor performance and loads data for comparison with analytical results; (2) to acquire S-76 forward flight rotor performance data in the 80- by 120-Foot Wind Tunnel to compare with existing full-scale 40- by 80-Foot Wind Tunnel test data that were acquired in 1977; (3) to evaluate the acoustic capability of the 80- by 120-Foot Wind Tunnel for acquiring blade vortex interaction (BVI) noise in the low speed range and compare BVI noise with in-flight test data; and (4) to evaluate the capability of the 80- by 120-Foot Wind Tunnel test section as a hover facility. The secondary objectives were (1) to evaluate rotor inflow and wake effects (variations in tunnel speed, shaft angle, and thrust condition) on wind tunnel test section wall and floor pressures; (2) to establish the criteria for the definition of flow breakdown (condition where wall corrections are no longer valid) for this size rotor and wind tunnel cross-sectional area; and (3) to evaluate the wide-field shadowgraph technique for visualizing full-scale rotor wakes. This data base of rotor performance and loads can be used for analytical and experimental comparison studies for full-scale, four-bladed, fully articulated rotor systems. Rotor performance and structural loads data are presented in this report.

Author

Angle of Attack; Full Scale Tests; Wind Tunnel Tests; Rotary

Wings; Rotor Dynamics; Blade-Vortex Interaction; Helicopters; Dynamic Loads; Horizontal Flight

19960026768; 96N28328 ESDU International Ltd., London, UK

Effect of wing height on lift and aerodynamic centre for a slender wing-body combination

Mar. 1995; ISSN 0141-397X; 26p; In English

Report No.(s): ESDU-95009; ISBN-0-85679-943-2; No Copyright; Avail: ESDU, Hardcopy, Microfiche

This technical report, ESDU 95009, presents a theoretical method, based on slender-body theory, for estimating the change in lift and aerodynamic center position resulting from moving a wing from a centrally-mounted position to one above or below the center line of a forebody-cylinder combination. It applies to a plane wing with sweptback leading edge, local span increasing (and then possibly constant) streamwise, and an unswept trailing edge. The body has a constant diameter circular section over the wing root chord and the wing may be set at an angle to the body center line. Equations and graphs are given for the ratios of the lift (excluding that on the body alone) due to angle of attack and due to wing setting angle for the combination with wing off-center to those for the combination with net wing removed and remounted centrally. The lift of the combination with centrally-mounted wing may be estimated from ESDU 91007, or from other sources, instead of using the slender-body value. Similarly, the slender-body value for aerodynamic center shift due to moving the wing from centrally-mounted to off-center, for which an equation is given, may be added to the aerodynamic center position found from ESDU 92024. Use of ESDU 91007 and 92024 in this way is expected to give better results than use of slender-body theory for the combination with centrally-mounted wing. The method is illustrated with two practical worked examples.

ESDU

Body-Wing Configurations; Aerodynamic Balance; Forebodies; Wings; Aerodynamics; Lift

19960026771; 96N28331 ESDU International Ltd., London, UK

Drag due to lift for plane swept wings, alone or in combination with a body, up to high angles of attack at subsonic speeds

Nov. 1995; ISSN 0141-397X; 49p; In English

Report No.(s): ESDU-95025; ISBN-0-85679-959-9; No Copyright; Avail: ESDU, Hardcopy, Microfiche

ESDU 95025 applies to wings of straight taper with transition near the leading edge. It uses a semi-empirical correlation to derive the attainable leading-edge suction from the theoretical leading-edge suction on a thin wing presented in ESDU 94037 and it applies between limits appropriate to inviscid attached flow and to fully-separated flow. to obtain the contribution of wing viscous axial force the variation of wing lift with angle of attack is predicted using the leading-edge suction analogy. That lift prediction includes the non-linear contribution from leading-edge vortex flow that is present for highly-swept wings, together with a side-edge vortex contribution, but the method is limited to angles of attack where the full realization of the predicted potential flow contribution is not degraded by flow separation over the main part of the wing. For the body contribution, ESDU 90034 may be used if the body can be approximated by an axi-symmetric shape. The contributions of the various terms are illustrated and comparisons are made with experimental results for a number of cases and suggest the method yields drag coefficient to within 0.01 of lift coefficient up to 70 percent of maximum lift for wings with leading-edge sweep exceeding about 15 degrees.

ESDU

Aerodynamic Drag; Swept Wings; Subsonic Speed; Aerodynamic Coefficients

19960026782; 96N28339 ESDU International Ltd., London, UK

Increments in aerofoil lift coefficient at zero angle of attack and in maximum lift coefficient due to deployment of a double-slotted or triple-slotted trailing-edge flap, with or without a leading-edge high-lift device, at low speeds

Apr. 1995; ISSN 0141-397X; 30p; In English; Supersedes ESDU-85033

Report No.(s): ESDU-94031; ESDU-85033; ISBN-0-85679-918-1; No Copyright; Avail: ESDU, Hardcopy, Microfiche

ESDU 94031 presents an estimation method based on the theory for a thin hinged plate modified using empirical correlation factors to account for the geometry of practical aerofoils and high-lift devices. Some allowance for the effects of chord extension was made by using flap chord ratio and lift coefficients based on airfoil extended chord but further adjustments were required to adapt to the considerable departure for slotted flaps from the model, possibly involving large chord extensions. The data for airfoils with trailing-edge flaps deployed from which the methods were developed were extracted from wind-tunnel tests reported in the literature covering a wide range of practical geometries. Fewer data were available for airfoils with both leading- and trailing-edge devices deployed. The methods apply to Reynolds numbers greater than a million and freestream Mach numbers less than 0.2. The predicted and test data for the lift coefficient increments correlated to within 15 percent. The use of the methods is illustrated by worked examples. to obtain results for an aerofoil with both leading-edge devices and slotted flaps deployed, ESDU 84026 is used in conjunction with this document and ESDU 94027.

ESDU

Aerodynamic Coefficients; Lift; Trailing Edge Flaps

19960026783; 96N28340 ESDU International Ltd., London, UK

Increments in aerofoil lift coefficient at zero angle of attack and in maximum lift coefficient due to deployment of a single-slotted trailing-edge flap, with or without a leading-edge high-lift device, at low speeds

Apr. 1995; ISSN 0141-397X; 21p; In English; Supersedes ESDU-85033

Report No.(s): ESDU-94030; ESDU-85033; ISBN-0-85679-917-3; No Copyright; Avail: ESDU, Hardcopy, Microfiche

ESDU 94030 presents an estimation method based on the theory for a thin hinged plate modified using empirical correlation factors to account for the geometry of practical airfoils and high-lift devices. Some allowance for the effects of chord extension was made by using flap chord ratio and lift coefficients based on airfoil extended chord but further adjustments were required to adapt to the considerable departure for slotted flaps from the model, possibly involving large chord extensions. The data for airfoils with trailing-edge flaps deployed from which the methods were developed were extracted from wind-tunnel tests reported in the literature covering a wide range of practical geometries. Fewer data were available for aerofoils with both leading- and trailing-edge devices deployed. The methods apply to Reynolds numbers greater than a million and freestream Mach numbers less than 0.2. The predicted and test data for the lift coefficient increments correlated to within 15 percent. The use of the methods is illustrated by worked examples. to obtain results for an aerofoil with both leading-edge devices and slotted flaps deployed, ESDU 84026 is used in conjunction with this document and ESDU 94027.

ESDU

Aerodynamic Coefficients; Lift; Trailing Edge Flaps; Slots

19960026784; 96N28341 ESDU International Ltd., London, UK

Wing lift coefficient increment at zero angle of attack due to deployment of double-slotted or triple-slotted flaps at low speeds

Aug. 1995; ISSN 0141-397X; 20p; In English; Supersedes Aero-F.01.01.09

Report No.(s): ESDU-95021; Aero-F.01.01.09; ISBN-0-85679-955-6; No Copyright; Avail: ESDU, Hardcopy, Microfiche

ESDU 95021 presents an empirical method that depends on determining the increment for an aerofoil representative of the wing and applying factors to allow for three-dimensional and part-span effects. ESDU 94031 is used to predict the increment for the airfoil, depending on the extended deployed chord. That increment is factored by the ratio of the wing to airfoil lift curve slopes obtained respectively from ESDU 70011 and Aero W.01.01.05, thereby accounting for aspect ratio, taper and sweep effects. Finally, part-span factors are applied that depend on the spanwise center of pressure of the loading due to incidence obtained from ESDU 83040. For double-slotted flaps, the range of parameters for which the method was validated are tabulated and include aspect ratios of 3.7 to 10, taper ratios from 0.3 to 0.5, leading-edge sweep up to 48 degrees, Mach numbers up to 0.2, first element deflections of 12 to 48 degrees and second element of 13 to 30 degrees. For triple-slotted flaps, no experimental data were found but it is anticipated the method will apply equally well. Sketches show the correlation between experimental results drawn from the literature and predicted results; for all results, 89 percent were correlated within 15 percent. A worked example illustrates the use of the method.

ESDU

Aerodynamic Coefficients; Lift; Slots; Flaps (Control Surfaces); Prediction Analysis Techniques; Wing Profiles

19960026786; 96N28343 ESDU International Ltd., London, UK

Computer program for estimation of spanwise loading of wings with camber and twist in subsonic attached flow

Jun. 1995; ISSN 0141-397X; 28p; In English

Report No.(s): ESDU-95010; ESDUpac-A9510; ISBN-0-85679-944-0; No Copyright; Avail: ESDU, Hardcopy, Microfiche

ESDU 95010 introduces a computer program, ESDUpac A9501, that uses lifting-surface theory based on the Multhopp-Richardson solution. It calculates the distributions of local lift and pitching moment due to incidence, and due to camber and due to twist both at zero incidence. They may be output as individual components or as total loadings at either a specified incidence or lift coefficient. ESDU 70011 is used to calculate a lift-curve slope and applies to a straight-tapered wing only; if a cranked wing is considered it requires the prior calculation of an equivalent wing using the method of ESDU 76003. ESDU 70011 is based on the Multhopp-Garner lifting-surface solution which provides a slightly more accurate value than Multhopp-Richardson. All loadings, forces and moments output by ESDUpac A9510 are factored by the ratio of the lift-curve slopes therefore. Guidance is given on the choice of the number of collocation stations to be used and its effect on accuracy. The method of rounding adopted at the central leading-edge kink is explained. The input is described and a worked example illustrates the output. The Fortran program is provided (uncompiled) on disc in the software volume.

ESDU

Applications Programs (Computers); Subsonic Flow; Pressure Distribution; Aerodynamic Loads; Control Surfaces

19960026791; 96N28348 ESDU International Ltd., London, UK

Example of analysis and representation of static pressure correction data in steady flight

Nov. 1995; ISSN 0141-4054; 26p; In English

Report No.(s): ESDU-95024; ISBN-0-85679-958-0; No Copyright; Avail: ESDU, Hardcopy, Microfiche

ESDU 95024 uses the case of a small military subsonic aircraft with compensated pressure sensing head fitted to a nose boom to illustrate methods of collecting and correlating data within a theoretical framework to provide static pressure corrections as a functional relationship over the flight envelope. The pressure transducer had a precision of 1 lbf/sq ft at a 95 percent confidence level and so only the most significant parameters were considered. Data were obtained from low-speed wind-tunnel testing, low-speed flights past an accurate barometer on a tower, and high-speed flights in formation with a calibrated aircraft. Those last results were reduced by use of transonic similarity. From that relatively modest set of data a valid calibration is derived and is shown through a statistical analysis to be precise to within 0.2 lbf/sq ft and the correlation is effected with a confidence interval of 1.1 lbf/sq ft.
ESDU

Static Pressure; Correction; Subsonic Speed; Flight Envelopes; Aerodynamic Loads; Calibrating; Pressure Heads

19960026797; 96N28354 ESDU International Ltd., London, UK

Transonic data memorandum: Upwash interference for wings in solid-liner wind tunnels using subsonic linearized-theory

Oct. 1995; ISSN 0141-4356; 129p; In English

Report No.(s): ESDU-95014; ISBN-0-85679-948-3; No Copyright; Avail: ESDU, Hardcopy, Microfiche

This technical report, ESDU 95014, gives graphs, for a range of dimensions (relative to the span) of the rectangular section, of the spanwise variation of the interference upwash along the leading- and trailing-edges, and mid-chord line of 66 transformed (to take account of compressibility) plane wing planforms (of which six are swept-forward). Mean values over the planform are also tabulated together with corrections to incidence such that values of lift and (vortex) drag are unchanged between wind tunnel and free-air. The aerodynamic centre positions and their aft shifts due to non-uniformity of upwash are also given. A Fortran program, ESDUpac A9514, provides a database of those results for ease of access and is provided on disc in the software volume. Use of the data for cambered and/or twisted wings and for test sections with corner fillets is considered and they may be applied to half- or full-wing tests. by use of the equivalent wing concept, they may also be used for wings with kinked leading- or trailing-edges and two examples are shown comparing results obtained using that concept with those calculated. The practical applicability of the results is discussed and they are expected to be satisfactory up to Mach numbers of 0.85 even in the presence of shocks or separation on the model.
ESDU

Upwash; Subsonic Speed; Wind Tunnel Tests; Aerodynamics; Wing Planforms; Data Bases; Wind Tunnels

19960026801; 96N28358 ESDU International Ltd., London, UK

Pitching moment of low-aspect-ratio cropped-delta wings up to high angles of attack at subsonic speeds

Sep. 1995; ISSN 0141-397X; 16p; In English

Report No.(s): ESDU-95022; ISBN-0-85679-956-4; No Copyright; Avail: ESDU, Hardcopy, Microfiche

ESDU 95022 presents graphical data for center of pressure position of delta and rectangular wings of aspect ratio 0.25 to 4 for angles of attack up to 90 degrees. They were obtained by correlating results from the literature. However, for cropped-delta wings there were fewer data and it only proved possible to present graphs applying to pre-stall conditions for angles of attack up to 30 degrees, aspect ratio from 0.1 to 4 and taper ratios from 0.1 to 0.5, although a method of extrapolation to taper ratios from 0.5 to unity is suggested. The correlation in that case also involved the aerodynamic center position, a graph for which is reproduced from ESDU 70011. The methods apply at low subsonic speeds; a method of extrapolating to high Mach numbers is given but is tentative for the rectangular and cropped-delta wings although for delta wings it is found to work well at Mach numbers up to 0.95. The method applies to thin, sharp-edged wings without camber or twist. It is accurate to 2 percent of root chord length. The pitching moment is obtained using the normal-force data from ESDU 93034.
ESDU

Pitching Moments; Aerodynamic Balance; Low Aspect Ratio; Subsonic Speed; Aspect Ratio; Delta Wings; Wing Profiles

19960027039; 96N28538 NASA Langley Research Center, Propulsion Aerodynamics Branch., Hampton, VA USA

Investigation of supersonic jet plumes using an improved two-equation turbulence model

Lakshmanan, B., Old Dominion Univ., USA; Abdol-Hamid, Khaled S., Analytical Services and Materials, Inc., USA; Oct. 1994; 8p; Repr. from Journal of Propulsion and Power, v. 10, no. 5, Sep.-Oct. 1994 p736-741; In English; 10th; Applied Aerodynamics Conference, 22-24 Jun. 1992, Palo Alto, CA, USA; Sponsored by American Inst. of Aeronautics and Astronautics, USA

Report No.(s): NASA-TM-111555; NAS 1.15:111555; AIAA Paper 92-2604; Copyright Waived (NASA); Avail: CASI; A02, Hardcopy; A01, Microfiche

Supersonic jet plumes were studied using a two-equation turbulence model employing corrections for compressible dissipation and pressure-dilatation. A space-marching procedure based on an upwind numerical scheme was used to solve the governing equations and turbulence transport equations. The computed results indicate that two-equation models employing corrections for compressible dissipation and pressure-dilatation yield improved agreement with the experimental data. In addition, the numerical study demonstrates that the computed results are sensitive to the effect of grid

refinement and insensitive to the type of velocity profiles used at the inflow boundary for the cases considered in the present study.
Author
Supersonic Jet Flow; Turbulence Models; Computational Fluid Dynamics; Upwind Schemes (Mathematics); Flow Distribution; Compressibility; Pressure Effects

19960027051; 96N28546 Modern Technologies Corp., Middleburg Heights, OH USA
The modelling of symmetric airfoil vortex generators ***Final Report***
Reichert, B. A., Kansas State Univ., USA; Wendt, B. J., Modern Technologies Corp., USA; Jun. 1996; 22p; In English; 34th; Aerospace Sciences Meeting and Exhibit, 15-18 Jan. 1996, Reno, NV, USA
Contract(s)/Grant(s): NAS3-27377; RTOP 505-62-52
Report No.(s): NASA-CR-198501; NAS 1.26:198501; AIAA Paper 96-0807; E-10315; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An experimental study is conducted to determine the dependence of vortex generator geometry and impinging flow conditions on shed vortex circulation and crossplane peak vorticity for one type of vortex generator. The vortex generator is a symmetric airfoil having a NACA 0012 cross-sectional profile. The geometry and flow parameters varied include angle-of-attack alfa, chordlength c, span h, and Mach number M. The vortex generators are mounted either in isolation or in a symmetric counter-rotating array configuration on the inside surface of a straight pipe. The turbulent boundary layer thickness to pipe radius ratio is delta/R = 0. 17. Circulation and peak vorticity data are derived from crossplane velocity measurements conducted at or about 1 chord downstream of the vortex generator trailing edge. Shed vortex circulation is observed to be proportional to M, alfa, and h/delta. With these parameters held constant, circulation is observed to fall off in monotonic fashion with increasing airfoil aspect ratio AR. Shed vortex peak vorticity is also observed to be proportional to M, alfa, and h/delta. Unlike circulation, however, peak vorticity is observed to increase with increasing aspect ratio, reaching a peak value at AR approx. 2.0 before falling off.
Author
Vortex Generators; Vortices; Angle of Attack; Mach Number; Turbulent Boundary Layer

19960027055; 96N28550 NASA Ames Research Center, Moffett Field, CA USA
Flight evaluation of advanced controls and displays for transition and landing on the NASA V/STOL systems research aircraft
Borchers, Paul F., NASA Ames Research Center, USA; Moralez, Ernesto, III, NASA Ames Research Center, USA; Stortz, Michael W., NASA Ames Research Center, USA; Apr. 1996; 124p; In English
Contract(s)/Grant(s): RTOP 505-68-33
Report No.(s): NASA-TP-3607; NAS 1.60:3607; A-961333; No Copyright; Avail: CASI; A06, Hardcopy; A02, Microfiche

Flight experiments were conducted on Ames Research Center's V/STOL, Systems Research Aircraft (VSRA) to assess the influence of advanced control modes and head-up displays (HUD's) on flying qualities for precision approach and landing operations. Evaluations were made for decelerating approaches to hover followed by a vertical landing and for slow landings for four control/display mode combinations: the basic YAV-8B stability augmentation system; attitude command for pitch, roll, and yaw; flightpath/acceleration command with translational rate command in the hover; and height-rate damping with translational-rate command. Head-up displays used in conjunction with these control modes provided flightpath tracking/pursuit guidance and deceleration commands for the decelerating approach and a mixed horizontal and vertical presentation for precision hover and landing. Flying qualities were established and control usage and bandwidth were documented for candidate control modes and displays for the approach and vertical landing. Minimally satisfactory bandwidths were determined for the translational-rate command system. Test pilot and engineer teams from the Naval Air Warfare Center, the Boeing Military Airplane Group, Lockheed Martin, McDonnell Douglas Aerospace, Northrop Grumman, Rolls-Royce, and the British Defense Research Agency participated in the program along with NASA research pilots from the Ames and Lewis Research Centers. The results, in conjunction with related ground-based simulation data, indicate that the flightpath/longitudinal-acceleration command response type in conjunction with pursuit tracking and deceleration guidance on the HUD would be essential for operation to instrument minimums significantly lower than the minimums for the AV-8B. It would also be a superior mode for performing slow landings where precise control to an austere landing area Such as a narrow road is demanded. The translational-rate command system would reduce pilot workload for demanding vertical landing tasks aboard ship and in confined land-based sites.
Author
Flight Tests; Head-Up Displays; V/STOL Aircraft; Research Aircraft; Aircraft Landing; STOVL Aircraft; Transition Flight; Flight Control; Aircraft Control

19960027449; 96N28838 NASA Ames Research Center, Moffett Field, CA USA
Parameter identification and modeling of longitudinal aerodynamics
Aksteter, J. W., Science Applications International Corp., USA; Parks, E. K., Arizona Univ., USA; Bach, R. E., Jr., NASA Ames Research Center, USA; 1995; 8p; Repr. from

Journal of Aircraft, v. 32, no. 4, Jul.-Aug. 1995 p 726-731; In English
Report No.(s): NASA-TM-111272; NAS 1.15:111272; Copyright Waived (NASA); Avail: CASI; A02, Hardcopy; A01, Microfiche

Using a comprehensive flight test database and a parameter identification software program produced at NASA Ames Research Center, a math model of the longitudinal aerodynamics of the Harrier aircraft was formulated. The identification program employed the equation error method using multiple linear regression to estimate the nonlinear parameters. The formulated math model structure adhered closely to aerodynamic and stability/control theory, particularly with regard to compressibility and dynamic manoeuvring. Validation was accomplished by using a three degree-of-freedom nonlinear flight simulator with pilot inputs from flight test data. The simulation models agreed quite well with the measured states. It is important to note that the flight test data used for the validation of the model was not used in the model identification.
Author
Aerodynamic Characteristics; Parameter Identification; Mathematical Models; Nonlinearity; Degrees of Freedom; Control Theory; Stability; Aerodynamics

19960027504; 96N28867 National Renewable Energy Lab., Golden, CO USA
Effects of grit roughness and pitch oscillations on the S810 airfoil
Ramsay, R. Reuss, Ohio State Univ., USA; Hoffman, M. J., Ohio State Univ., USA; Gregorek, G. M., Ohio State Univ., USA; Jan. 1996; 159p; In English
Contract(s)/Grant(s): DE-AC36-83CH-10093
Report No.(s): NREL/TP-442-7816; DE96-000518; No Copyright; Avail: CASI; A08, Hardcopy; A02, Microfiche

An S810 airfoil model was tested in The Ohio State University Aeronautical and Astronautical Research Laboratory 3 x 5 subsonic wind tunnel under steady state and unsteady conditions. The test defined baseline conditions for steady state angles of attack from -20 deg to +40 deg and examined unsteady behavior by oscillating the model about its pitch axis for three mean angles, three frequencies, and two amplitudes. For all cases, Reynolds numbers of 0.75, 1, 1.25, and 1.5 million were used. In addition, the above conditions were repeated after the application of leading edge grit roughness (LEGR) to determine contamination effects on the airfoil performance. Baseline steady state results of the S810 testing showed a maximum lift coefficient of 1.15 at 15.2 deg angle of attack. The application of LEGR reduced the maximum lift coefficient by 12% and increased the 0.0085 minimum drag coefficient value by 88%. The zero lift pitching moment of -0.0286 showed a 16% reduction in magnitude to -0.0241 with LEGR applied. Data were also obtained for two pitch oscillation amplitudes: +/- 5.5 deg and +/- 10 deg. The larger amplitude consistently gave a higher maximum lift coefficient than the smaller amplitude and both sets of unsteady maximum lift coefficients were greater than the steady state values. Stall was delayed on the airfoil while the angle of attack was increasing, thereby causing an increase in maximum lift coefficient. A hysteresis behavior was exhibited for all the unsteady test cases. The hysteresis loops were larger for the higher reduced frequencies and for the larger amplitude oscillations. In addition to the hysteresis behavior, an unusual feature of these data were a sudden increase in the lift coefficient where the onset of stall was expected. As in the steady case, the effect of LEGR in the unsteady case was to reduce the lift coefficient at high angles of attack.
DOE
Surface Roughness; Pitching Moments; Oscillations; Airfoils

19960027883; 96N29025 NASA Langley Research Center, Hampton, VA USA
Wind tunnel test results of a 1/8-scale fan-in-wing model
Wilson, John C., Army Aviation Systems Command, USA; Gentry, Garl L., NASA Langley Research Center, USA; Gorton, Susan A., Army Aviation Systems Command, USA; May 1996; 90p; In English
Contract(s)/Grant(s): RTOP 505-59-87-92; DA Proj. 1L1-61102-AH-45
Report No.(s): NASA-TM-4710; NAS 1.15:4710; ATCOM-TR-96-A-005; L-17448; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

A 1/8-scale model of a fan-in-wing concept considered for development by Grumman Aerospace Corporation for the U.S. Army was tested in the Langley 14- by 22-Foot Subsonic Tunnel. Hover testing, which included height above a pressure-instrumented ground plane, angle of pitch, and angle of roll for a range of fan thrust, was conducted in a model preparation area near the tunnel. The air loads and surface pressures on the model were measured for several configurations in the model preparation area and in the tunnel. The major hover configuration change was varying the angles of the vanes attached to the exit of the fans for producing propulsive force. As the model height above the ground was decreased, there was a significant variation of thrust-removed normal force with constant fan speed. The greatest variation was generally for the height-to-fan exit diameter ratio of less than 2.5; the variation was reduced by deflecting fan exit flow outboard with the vanes. In the tunnel angles of pitch and sideslip, height above the tunnel floor, and wind speed were varied for a range of fan thrust and different vane angle configurations. Other configuration features such as flap deflections and tail incidence were evaluated as well. Though the V-tail empennage provided an increase in static longitudinal stability, the total model configuration remained unstable.
Author
Wind Tunnel Tests; Scale Models; Ground Effect

(Aerodynamics); Ducted Fans; Hovering; Aircraft Configurations; Aerodynamic Stability

19960027894; 96N29036 NYMA, Inc., Engineering Services Div., Brook Park, OH USA

A non-axisymmetric linearized supersonic wave drag analysis: Mathematical theory *Final Report*

Barnhart, Paul J., NYMA, Inc., USA; May 1996; 22p; In English

Contract(s)/Grant(s): NAS3-27186; RTOP 505-60-00

Report No.(s): NASA-CR-198430; NAS 1.26:198430; E-10030; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A Mathematical theory is developed to perform the calculations necessary to determine the wave drag for slender bodies of non-circular cross section. The derivations presented in this report are based on extensions to supersonic linearized small perturbation theory. A numerical scheme is presented utilizing Fourier decomposition to compute the pressure coefficient on and about a slender body of arbitrary cross section.

Author

Wave Drag; Supersonic Drag; Engine Airframe Integration; Perturbation Theory; Slender Bodies

19960027897; 96N29039 National Aeronautics and Space Administration. Langley Research Center, Hampton, VA USA

Blunt body near wake flow field at Mach 6

Horvath, Thomas J., National Aeronautics and Space Administration. Langley Research Center, USA; McGinley, Catherine B., National Aeronautics and Space Administration. Langley Research Center, USA; Hannemann, Klaus, Deutsche Forschungsanstalt fuer Luft- und Raumfahrt, Germany; Jun. 20, 1996; 30p; In English; 27th; Fluid Dynamics Conference, 17-20 Jun. 1996, New Orleans, LA, USA; Sponsored by American Inst. of Aeronautics and Astronautics, USA

Report No.(s): NASA-TM-111601; NAS 1.15:111601; AIAA Paper 96-1935; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Tests were conducted in a Mach 6 flow to examine the reattachment process of an axisymmetric free shear layer associated with the near wake of a 70 deg. half angle, spherically blunted cone with a cylindrical after body. Model angle of incidence was fixed at 0 deg. and free-stream Reynolds numbers based on body diameter ranged from 0.5 x 10(exp 6) to 4 x 10(exp 6). The sensitivity of wake shear layer transition on reattachment heating was investigated. The present perfect gas study was designed to compliment results obtained previously in facilities capable of producing real gas effects. The instrumented blunted cone model was designed primarily for testing in high enthalpy hypervelocity shock tunnels in both this country and abroad but was amenable for testing in conventional hypersonic blowdown wind tunnels as well. Surface heating rates were inferred from temperature - time histories from coaxial surface thermocouples on the model forebody and thin film resistance gages along the model base and cylindrical after body. General flow feature (bow shock, wake shear layer, and recompression shock) locations were visually identified by schlieren photography. Mean shear layer position and growth were determined from intrusive pitot pressure surveys. In addition, wake surveys with a constant temperature hot-wire anemometer were utilized to qualitatively characterize the state of the shear layer prior to reattachment. Experimental results were compared to laminar perfect gas predictions provided by a 3-D Navier Stokes code (NSHYP). Shear layer impingement on the instrumented cylindrical after body resulted in a localized heating maximum that was 21 to 29 percent of the forebody stagnation point heating. Peak heating resulting from the reattaching shear layer was found to be a factor of 2 higher than laminar predictions, which suggested a transitional shear layer. Schlieren flow visualization and fluctuating voltage time histories and spectra from the hot wire surveys across the shear layer substantiate this observation. The sensitivity of surface heating to forebody roughness was characterized for a reattaching shear layer. For example, at R(sub infinity), d = 4 x 10(exp 6), when the shear layer was transitional, the magnitude of peak heating from shear layer impingement was reduced by approximately 24 percent when transition grit was applied to the forebody. The spatial location of the local peak, however, remained unchanged.

Author

Blunt Bodies; Half Cones; Aerobraking; Near Wakes; Hypersonic Speed; Flow Distribution; Compressible Flow; Compressible Boundary Layer; Bow Waves

19960028091; 96N29182 North Carolina State Univ., Dept. of Mechanical and Aerospace Engineering., Raleigh, NC USA

Computational analysis of semi-span model test techniques

Milholen, William E., II, North Carolina State Univ., USA; Chokani, Ndaona, North Carolina State Univ., USA; Mar. 1996; 232p; In English

Contract(s)/Grant(s): NCC1-169; RTOP 505-59-85-01

Report No.(s): NASA-CR-4709; NAS 1.26:4709; No Copyright; Avail: CASI; A11, Hardcopy; A03, Microfiche

A computational investigation was conducted to support the development of a semi-span model test capability in the NASA LaRC's National Transonic Facility. This capability is required for the testing of high-lift systems at flight Reynolds numbers. A three-dimensional Navier-Stokes solver was used to compute the low-speed flow over both a full-span configuration and a semi-span configuration. The computational results were found to be in good agreement with the experimental data. The computational results indicate that the stand-

off height has a strong influence on the flow over a semi-span model. The semi-span model adequately replicates the aerodynamic characteristics of the full-span configuration when a small stand-off height, approximately twice the tunnel empty sidewall boundary layer displacement thickness, is used. Several active sidewall boundary layer control techniques were examined including: upstream blowing, local jet blowing, and sidewall suction. Both upstream tangential blowing, and sidewall suction were found to minimize the separation of the sidewall boundary layer ahead of the semi-span model. The required mass flow rates are found to be practicable for testing in the NTF. For the configuration examined, the active sidewall boundary layer control techniques were found to be necessary only near the maximum lift conditions.
Author
Semispan Models; Aircraft Models; Transonic Wind Tunnels; Navier-Stokes Equation; Flow Distribution; Boundary Layer Control

19960028555; 96N29282 North Carolina State Univ., Raleigh, NC USA
Three-dimensional aerodynamic analysis of a subsonic transport high-lift configuration and comparisons with wind-tunnel test results
Edge, D. Christian, North Carolina State Univ., USA; Perkins, John N., North Carolina State Univ., USA; Jun. 22, 1995; 14p; In English; 13th; AIAA Applied Aerodynamics Conference, 19-22 Jun. 1995, San Diego, CA, USA; Sponsored by American Inst. of Aeronautics and Astronautics, USA
Contract(s)/Grant(s): NCC1-154
Report No.(s): NASA-CR-201457; NAS 1.15:201457; AIAA Paper 95-1846; Copyright Waived (NASA); Avail: CASI; A03, Hardcopy; A01, Microfiche

The sizing and efficiency of an aircraft is largely determined by the performance of its high-lift system. Subsonic civil transports most often use deployable multi-element airfoils to achieve the maximum-lift requirements for landing, as well as the high lift-to-drag ratios for take-off. However, these systems produce very complex flow fields which are not fully understood by the scientific community. In order to compete in today's market place, aircraft manufacturers will have to design better high-lift systems. Therefore, a more thorough understanding of the flows associated with these systems is desired. Flight and wind-tunnel experiments have been conducted on NASA Langley's B737-100 research aircraft to obtain detailed full-scale flow measurements on a multi-element high-lift system at various flight conditions. As part of this effort, computational aerodynamic tools are being used to provide preliminary flow-field information for instrumentation development, and to provide additional insight during the data analysis and interpretation process. The purpose of this paper is to demonstrate the ability and usefulness of a three-dimensional low-order potential flow solver, PMARC, by comparing computational results with data obtained from 1/8 scale wind-tunnel tests. Overall, correlation of experimental and computational data reveals that the panel method is able to predict reasonably well the pressures of the aircraft's multi-element wing at several spanwise stations. PMARC's versatility and usefulness is also demonstrated by accurately predicting inviscid three-dimensional flow features for several intricate geometrical regions.
Author
Three Dimensional Flow; Aerodynamic Configurations; Aerodynamic Characteristics; Flight Conditions; Inviscid Flow; Subsonic Flow; Subsonic Aircraft; Transport Aircraft

19960028556; 96N29283 NASA Langley Research Center, Hampton, VA USA
High Reynolds number analysis of an axisymmetric afterbody with flow separation
Carlson, John R., NASA Langley Research Center, USA; Reubush, David E., NASA Langley Research Center, USA; Jun. 20, 1996; 16p; In English; 19th; AIAA Advanced Measurement and Ground Testing Technology Conference, 17-20 Jun. 1996, New Orleans, LA, USA; Sponsored by American Inst. of Aeronautics and Astronautics, USA
Report No.(s): NASA-TM-111603; NAS 1.15:111603; AIAA Paper 96-2274; Copyright Waived (NASA); Avail: CASI; A03, Hardcopy; A01, Microfiche

The ability of a three-dimensional Navier-Stokes method, PAB3D, to predict nozzle afterbody flow at high Reynolds number was assessed. Predicted surface pressure coefficient distributions and integrated afterbody drag are compared with experimental data obtained from the NASA-Langley 0.3 m Transonic Cryogenic Tunnel. Predicted afterbody surface pressures matched experimental data fairly closely. The change in the pressure coefficient distribution with Reynolds number was slightly over-predicted. Integrated afterbody drag was typically high compared to the experimental data. The change in afterbody pressure drag with Reynolds number was fairly small. The predicted point of flow separation on the nozzle was slightly downstream of that observed from oilflow data at low Reynolds numbers and had a very slight Reynolds number dependence, moving slightly further downstream as Reynolds number increased.
Author
High Reynolds Number; Afterbodies; Boundary Layer Separation; Nozzle Flow; Navier-Stokes Equation; Separated Flow

19960028557; 96N29284 National Aeronautics and Space Administration. Langley Research Center, Hampton, VA USA
High Reynolds number analysis of flat plate and separated afterbody flow using non-linear turbulence models
Carlson, John R., National Aeronautics and Space Administration. Langley Research Center, USA; Jul. 03, 1996; 22p; In English; 32nd; AIAA/ASME/SAE/ASEE Joint Propulsion

Conference and Exhibit, 1-3 Jul. 1996, Lake Buena Vista, FL, USA; Sponsored by Society of Automotive Engineers, Inc., USA
Report No.(s): NASA-TM-111605; NAS 1.15:111605; AIAA Paper 96-2544; Copyright Waived (NASA); Avail: CASI; A03, Hardcopy; A01, Microfiche

The ability of the three-dimensional Navier-Stokes method, PAB3D, to simulate the effect of Reynolds number variation using non-linear explicit algebraic Reynolds stress turbulence modeling was assessed. Subsonic flat plate boundary-layer flow parameters such as normalized velocity distributions, local and average skin friction, and shape factor were compared with DNS calculations and classical theory at various local Reynolds numbers up to 180 million. Additionally, surface pressure coefficient distributions and integrated drag predictions on an axisymmetric nozzle afterbody were compared with experimental data from 10 to 130 million Reynolds number. The high Reynolds data was obtained from the NASA Langley 0.3m Transonic Cryogenic Tunnel. There was generally good agreement of surface static pressure coefficients between the CFD and measurement. The change in pressure coefficient distributions with varying Reynolds number was similar to the experimental data trends, though slightly over-predicting the effect. The computational sensitivity of viscous modeling and turbulence modeling are shown. Integrated afterbody pressure drag was typically slightly lower than the experimental data. The change in afterbody pressure drag with Reynolds number was small both experimentally and computationally, even though the shape of the distribution was somewhat modified with Reynolds number.
Author
Separated Flow; High Reynolds Number; Flow Characteristics; Turbulence Models; Reynolds Stress; Boundary Layer Flow; Three Dimensional Models

19960028952; 96N29555 Stanford Univ., CA USA
Study of the velocity gradient tensor in turbulent flow
Cheng, Wei-Ping, Stanford Univ., USA; Cantwell, Brian, Stanford Univ., USA; Jun. 1996; 230p; In English
Report No.(s): NASA-CR-201449; NAS 1.26:201449; JIAA TR 114; No Copyright; Avail: CASI; A11, Hardcopy; A03, Microfiche

The behavior of the velocity gradient tensor, A(ij)=delta u(i)/delta x(j), was studied using three turbulent flows obtained from direct numerical simulation The flows studies were: an inviscid calculation of the interaction between two vortex tubes, a homogeneous isotropic flow, and a temporally evolving planar wake. Self-similar behavior for each flow was obtained when A(ij) was normalized with the mean strain rate. The case of the interaction between two vortex tubes revealed a finite sized coherent structure with topological characteristics predictable by a restricted Euler model. This structure was found to evolve with the peak vorticity as the flow approached singularity. Invariants of A(ij) within this structure followed a straight line relationship of the form: gamma(sup 3)+gammaQ+R=0, where Q and R are the second and third invariants of A(ij), and the eigenvalue gamma is nearly constant over the volume of this structure. Data within this structure have local strain topology of unstable-node/saddle/saddle. The characteristics of the velocity gradient tensor and the anisotropic part of a related acceleration gradient tensor H(ij) were also studied for a homogeneous isotropic flow and a temporally evolving planar wake. It was found that the intermediate principal eigenvalue of the rate-of-strain tensor of H(ij) tended to be negative, with local strain topology of the type stable-node/saddle/saddle. There was also a preferential eigenvalue direction. The magnitude of H(ij) in the wake flow was found to be very small when data were conditioned at high local dissipation regions. This result was not observed in the relatively low Reynolds number simulation of homogeneous isotropic flow. A restricted Euler model of the evolution of A(ij) was found to reproduce many of the topological features identified in the simulations.
Author
Velocity Distribution; Turbulent Flow; Inviscid Flow; Computerized Simulation; Flow Characteristics; Topology; Tensor Analysis

19960028955; 96N29558 Aeronautical Systems Div., Wright-Patterson AFB, OH USA
The Hypersonic Revolution, Volume 2, From Max Valier to Project Prime *Final Report, 1 Jan. - 1 Dec. 1967*
Hallion, Richard P., Aeronautical Systems Div., USA; Aug. 1995; 676p; In English; Limited Reproducibility: More than 20% of this document may be affected by microfiche quality
Report No.(s): AD-A302634; ASC-TR-95-5010; No Copyright; Avail: Issuing Activity (Defense Technical Information Center (DTIC)), Microfiche

The hypersonic revolution has been a particularly American one, borne of the national pursuit of transonic and supersonic flight technology. True, it does have both domestic and international dimensions, in the prophecy of Robert Goddard, Hermann Oberth, and Konstantin Tsiolkovskiy at the beginning of the twentieth century, and in the prescient (if impractical) studies of Eugen Sanger and Irene Bredt (later Irene Sanger-Bredt) near mid-century. But if its inspiration was sometimes international in flavor, its execution was American - from the early pre-X-15 studies of the 1950's through the pioneering missions of Columbia in 1981. Primarily, the hypersonic revolution grew out of the traditional federal-industrial partnership that had benefitted American aviation since the First World War. It germinated and flourished amidst the laboratories of the Air Force, Navy, the National Advisory Committee for Aeronautics (NACA), and its successor, the National Aeronautics and Space Administration (NASA), and the major aerospace manufacturers. Not merely an aerodynamic revolution, the hypersonic revolution - like the

supersonic breakthrough and the drive for the 'modern,' airplane before it - involved the creative integration and exploitation of diverse technologies.
DTIC
Supersonic Combustion Ramjet Engines; Lifting Bodies; Supersonic Flight; Space Shuttles; Aerospace Planes; Hypersonics

19960029060; 96N29624 North Carolina State Univ., Raleigh, NC USA
Hypersonic Boundary Layer Stability Experiments in a Quiet Wind Tunnel with Bluntness Effects ***Final Report***
Lachowicz, Jason T., North Carolina State Univ., USA; Chokani, Ndaona, North Carolina State Univ., USA; Jan. 01, 1996; 232p; In English
Contract(s)/Grant(s): NCC1-183; RTOP-505-59-50-02
Report No.(s): NASA-CR-198272; NAS 1.26:198272; No Copyright; Avail: CASI; A11, Hardcopy; A03, Microfiche

Hypersonic boundary layer measurements over a flared cone were conducted in a Mach 6 quiet wind tunnel at a free-stream unit Reynolds number of 2.82 million/ft. This Reynolds number provided laminar-to-transitional flow over the cone model in a low-disturbance environment. Four interchangeable nose-tips, including a sharp-tip, were tested. Point measurements with a single hot-wire using a novel constant voltage anemometer were used to measure the boundary layer disturbances. Surface temperature and schlieren measurements were also conducted to characterize the transitional state of the boundary layer and to identify instability modes. Results suggest that second mode disturbances were the most unstable and scaled with the boundary layer thickness. The second mode integrated growth rates compared well with linear stability theory in the linear stability regime. The second mode is responsible for transition onset despite the existence of a second mode subharmonic. The subharmonic disturbance wavelength also scales with the boundary layer thickness. Furthermore, the existence of higher harmonics of the fundamental suggests that nonlinear disturbances are not associated with 'high' free stream disturbance levels. Nose-tip radii greater than 2.7% of the base radius completely stabilized the second mode.
Author
Boundary Layer Stability; Hypersonic Boundary Layer; Nose Tips; Wind Tunnel Tests; Flared Bodies; Conical Bodies; Boundary Layer Transition

19960029099; 96N29642 Research Inst. for Advanced Computer Science, Moffett Field, CA USA
An all-at-once reduced Hessian SQP scheme for aerodynamic design optimization
Feng, Dan, Research Inst. for Advanced Computer Science, USA; Pulliam, Thomas H., Research Inst. for Advanced Computer Science, USA; Oct. 1995; 26p; In English
Contract(s)/Grant(s): NAS2-13721
Report No.(s): NASA-CR-201068; NAS 1.26:201068; RIACS-95-19; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This paper introduces a computational scheme for solving a class of aerodynamic design problems that can be posed as nonlinear equality constrained optimizations. The scheme treats the flow and design variables as independent variables, and solves the constrained optimization problem via reduced Hessian successive quadratic programming. It updates the design and flow variables simultaneously at each iteration and allows flow variables to be infeasible before convergence. The solution of an adjoint flow equation is never needed. In addition, a range space basis is chosen so that in a certain sense the 'cross term' ignored in reduced Hessian SQP methods is minimized. Numerical results for a nozzle design using the quasi-one-dimensional Euler equations show that this scheme is computationally efficient and robust. The computational cost of a typical nozzle design is only a fraction more than that of the corresponding analysis flow calculation. Superlinear convergence is also observed, which agrees with the theoretical properties of this scheme. All optimal solutions are obtained by starting far away from the final solution.
Author
Hessian Matrices; Aerodynamics; Quadratic Programming; Design Analysis; Iterative Solution

19960029105; 96N29648 California Univ., Davis, CA USA
Aerodynamic shape optimization using control theory
Reuther, James, California Univ., USA; May 1996; 260p; In English
Contract(s)/Grant(s): NAS2-13721
Report No.(s): NASA-CR-201064; NAS 1.15:201064; RIACS-96-09; Copyright Waived (NASA); Avail: CASI; A12, Hardcopy; A03, Microfiche

Aerodynamic shape design has long persisted as a difficult scientific challenge due its highly nonlinear flow physics and daunting geometric complexity. However, with the emergence of Computational Fluid Dynamics (CFD) it has become possible to make accurate predictions of flows which are not dominated by viscous effects. It is thus worthwhile to explore the extension of CFD methods for flow analysis to the treatment of aerodynamic shape design. Two new aerodynamic shape design methods are developed which combine existing CFD technology, optimal control theory, and numerical optimization techniques. Flow analysis methods for the potential flow equation and the Euler equations form the basis of the two respective design methods. In each case, optimal control theory is used to derive the adjoint differential equations, the solution of which provides the necessary gradient information to a numerical optimization method much more efficiently then by conventional finite differencing. Each technique uses a quasi-Newton numerical optimization algorithm to drive an aerodynamic objective function toward a minimum. An analytic grid perturbation method is developed to modify body

fitted meshes to accommodate shape changes during the design process. Both Hicks-Henne perturbation functions and B-spline control points are explored as suitable design variables. The new methods prove to be computationally efficient and robust, and can be used for practical airfoil design including geometric and aerodynamic constraints. Objective functions are chosen to allow both inverse design to a target pressure distribution and wave drag minimization. Several design cases are presented for each method illustrating its practicality and efficiency. These include non-lifting and lifting airfoils operating at both subsonic and transonic conditions.
Author
Control Theory; Aerodynamic Configurations; Optimization; Design Analysis; Grid Generation (Mathematics); Differential Equations; Mathematical Models

19960029125; 96N29660 Research Inst. for Advanced Computer Science, Moffett Field, CA USA
Aerodynamic design optimization via reduced Hessian SQP with solution refining
Feng, Dan, Research Inst. for Advanced Computer Science, USA; Pulliam, Thomas H., Research Inst. for Advanced Computer Science, USA; Dec 1995; 24p; In English
Contract(s)/Grant(s): NAS2-13721
Report No.(s): NASA-CR-201067; NAS 1.26:201067; RIACS-95-24; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An all-at-once reduced Hessian Successive Quadratic Programming (SQP) scheme has been shown to be efficient for solving aerodynamic design optimization problems with a moderate number of design variables. This paper extends this scheme to allow solution refining. In particular, we introduce a reduced Hessian refining technique that is critical for making a smooth transition of the Hessian information from coarse grids to fine grids. Test results on a nozzle design using quasi-one-dimensional Euler equations show that through solution refining the efficiency and the robustness of the all-at-once reduced Hessian SQP scheme are significantly improved.
Author
Hessian Matrices; Optimization; Design Analysis; Aerodynamics; Quadratic Programming; Refining; Iterative Solution

19960029247; 96N29744 Naval Postgraduate School, Monterey, CA USA
Strut and wall interference on jet-induced ground effects of a STOVL aircraft in hover
Kristy, Michael H., Naval Postgraduate School, USA; Sep. 1995; 93p; In English
Report No.(s): NASA-TM-111708; NAS 1.15:111708; AD-A303614; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

A small scale ground effect test rig was used to study the ground plane flow field generated by a STOVL aircraft in hover. The objective of the research was to support NASA-Ames Research Center planning for the Large Scale Powered Model (LSPM) test for the ARPA-sponsored ASTOVL program. Specifically, small scale oil flow visualization studies were conducted to make a relative assessment of the aerodynamic interference of a proposed strut configuration and a wall configuration on the ground plane stagnation line. A simplified flat plate model representative of a generic jet-powered STOVL aircraft was used to simulate the LSPM. Cold air jets were used to simulate both the lift fan and the twin rear engines. Nozzle Pressure Ratios were used that closely represented those used on the LSPM tests. The flow visualization data clearly identified a shift in the stagnation line location for both the strut and the wall configuration. Considering the experimental uncertainty, it was concluded that either the strut configuration o r the wall configuration caused only a minor aerodynamic interference.
DTIC
V/STOL Aircraft; STOVL Aircraft; Struts; Wall Flow; Aerodynamic Interference; Arpa Computer Network; Flow Visualization; Ground Effect (Aerodynamics); Hovering

19960029266; 96N29762 NASA Marshall Space Flight Center, Huntsville,AL USA
Assessment of Lifting Body Linear Aerospike Plume Effects on Vehicle Aerodynamics
Ruf, Joseph H., NASA Marshall Space Flight Center, USA; Frost, Alonzo L., NASA Marshall Space Flight Center, USA; Vu, Bruce, NASA Marshall Space Flight Center, USA; Canabal, Francisco, NASA Marshall Space Flight Center, USA; Thirteenth Workshop for Computational Fluid Dynamic Applications in Rocket Propulsion and Launch Vehicle Technology; Mar. 1996, pp. 935-962; In English; Also announced as 19960029254; No Copyright; Avail: CASI; A03, Hardcopy; A06, Microfiche

The lifting body/linear aerospike is one of three configurations being studied for a single stage to orbit (SSTO) vehicle. A preliminary aerodynamics database existed for then current lifting body configurations, however, this database was developed without considering plume effects. A combined effort by the Computational Fluid Dynamics (CFD) and the Experimental Fluids Dynamics Branches was undertaken to determine first order effects of plume/external flow interactions on vehicle aerodynamics of this lifting body/linear aerospike configuration. of interest were plume pumping/entrainment at low Mach numbers and plume induced separation of flow over the vehicle at higher altitudes. The CFD analysis included combinations of four Mach numbers, two angles of attack, and four throttle settings. The majority of the CFD was two dimensional centerline analysis of the lifting body/aerospike. Incremental plume effects were derived by comparing the power-on, power-off, and throttled cases

and were extrapolated to the preliminary aerodynamic database. The plume had little effect on the vehicle aerodynamics for supersonic freestream velocities. At subsonic freestream velocities, the plume affected the vehicle aerodynamics through both jet pumping/entrainment and the jet flap effect.
Author
Computational Fluid Dynamics; Lifting Bodies; Aerodynamic Configurations; Flow Characteristics; Plumes; Single Stage to Orbit Vehicles; Angle of Attack; Pressure Distribution

19960029344; 96N29796 NASA Langley Research Center, Hampton, VA USA
Review of blunt body wake flows at hypersonic low density conditions
Moss, J. N., NASA Langley Research Center, USA; Price, J. M., NASA Langley Research Center, USA; Jun. 20, 1996; 20p; In English; 31; AIAA ThermophysicS Conference, 17-20 Jun.1996, New Orleans, LA, USA
Report No.(s): NASA-TM-111599; NAS 1.15:111599; AIAA Paper 96-1803; Copyright Waived (NASA); Avail: CASI; A03, Hardcopy; A01, Microfiche

Recent results of experimental and computational studies concerning hypersonic flows about blunted cones including their near wake are reviewed. Attention is focused on conditions where rarefaction effects are present, particularly in the wake. The experiments have been performed for a common model configuration (70 deg spherically-blunted cone) in five hypersonic facilities that encompass a significant range of rarefaction and nonequilibrium effects. Computational studies using direct simulation Monte Carlo (DSMC) and Navier-Stokes solvers have been applied to selected experiments performed in each of the facilities. In addition, computations have been made for typical flight conditions in both Earth and Mars atmospheres, hence more energetic flows than produced in the ground-based tests. Also, comparisons of DSMC calculations and forebody measurements made for the Japanese Orbital Reentry Experiment (OREX) vehicle (a 50 deg spherically-blunted cone) are presented to bridge the spectrum of ground to flight conditions.
Author
Wakes; Hypersonic Flow; Blunt Bodies; Rarefaction; Nose Cones; Wind Tunnel Tests; Flow Distribution

19960029445; 96N29863 Advisory Group for Aerospace Research and Development, Fluid Dynamics Panel., Neuilly-Sur-Seine, France
Hypersonic Experimental and Computational Capability, Improvement and Validation, Volume 1 *l'Hypersonique experimentale et de calcul - Capacite, amelioration et validation*
Saric, William S., Editor, Arizona State Univ., USA; Muylaert, Jean, Editor, European Space Agency. European Space Research and Technology Center, ESTEC, Netherlands; Dujarric, Christian, Editor, European Space Agency, France; May 1996; 209p; In English
Report No.(s): AGARD-AR-391-Vol-1; ISBN-92-836-1037-7; Copyright Waived; Avail: CASI; A10, Hardcopy; A03, Microfiche

The results of the phase 1 effort conducted under AGARD Working Group 18 on Hypersonic Experimental and Computational Capability, Improvement and Validation are presented in this report. A discussion of issues and problems classified into seven topic areas precedes the individual sections on Shock/Boundary-Layer Interactions, Laminar-Turbulent Transition, Rarefield Flow, Real-Gas Effects, Facility Calibration Procedures, Extrapolation to Flight and Hypersonic Test Facilities. Recommendations for future work in these areas is included at the end of each section. Conclusions for the work conducted during the phase 2 of this Working Group will be presented in Volume 2.
Author
Hypersonics; Shock Wave Interaction; Transition Flow; Boundary Layer Transition; Calibrating; Computational Fluid Dynamics; Test Facilities; Gas Dynamics; Research Facilities; Hypersonic Flow

19960031935; 96N29926 Air Force Inst. of Tech., School of Engineering., Wright-Patterson AFB, OH USA
An Experimental Investigation of a Sting-Mounted Finite Circulation Control Wing
Bradley, Lorenzo C., Air Force Inst. of Tech., USA; Dec. 1995; 99p; In English
Report No.(s): AD-A303354; AFIT/GAE/ENY/95D-03; No Copyright; Avail: CASI; A05, Hardcopy; A02, Microfiche

This study investigated the lift, drag and pitching moment performance of a circulation control wing in the AFIT 5-ft wind tunnel. The experimental wing model was a 20 percent thick, 8.5 percent camber, partial elliptical cross-section, single blowing slot, rectangular planform wing. The aspect ratios tested were 3.99, 3.77 and 3.75. The variables in the investigation included the slot blowing rate and model configuration. The model was modified by adding a leading edge nose droop, a trailing edge splitter plate and wing tip fences to improve flow at the leading edge, reduce separation effects, and encourage attached flow on the upper surface, respectively. Results showed increased lift due to the splitter plate at low blowing rates. The leading edge nose droop increased the stall angle of attack of the wing model as blowing was increased. The wing tip fences increased the lift coefficient at medium and high blowing rates.
DTIC
Aerodynamic Characteristics; Aircraft Configurations; Angle of Attack; Aspect Ratio; Rectangular Planforms; Fixed Wings; Aerodynamic Drag

19960031956; 96N29944 Naval Postgraduate School, Monterey, CA USA

A study of helicopter rotor dynamics and modeling methods

Hiatt, Daniel S., Naval Postgraduate School, USA; Sep. 1995; 133p; In English; No Copyright; Avail: Issuing Activity (Defense Technical Information Center (DTIC)), Microfiche

The rotor system is the primary source of vibratory forces on a helicopter. Vibratory forces result from the rotor system response to dynamic and aerodynamic loading. This thesis discusses sources of excitation, and investigates rotor system modeling methods. Computer models based on finite element and Myklestad methods are developed and compared for the free and forced vibration cases of a uniform rotor blade. The modeling assumptions and the effects of non-uniform physical parameters are discussed. The Myklestad based computer model is expanded to include coupling effects inherent in modern rotor blades. This rotor modeling program is incorporated into the Dynamics section of the Joint Army/Navy Rotorcraft Analysis and Design (JANRAD) program currently used by the Naval Postgraduate School's helicopter design course (AA43O6) for preliminary helicopter design and analysis. Computer programs are developed as tools to investigate the stability of a rotor s ystem for the specific cases of rotor flapping and ground/air resonance. A rotor flapping stability model, based upon Floquet theory, provides a means of analyzing the effect of increasing advance ratio on the flapping stability of a rotor system.

DTIC

Aerodynamic Loads; Rotary Wing Aircraft; Rotor Dynamics; Helicopter Design; Floquet Theorem; Finite Element Method; Design Analysis; Computerized Simulation

19960033266; 96N30070 Massachusetts Inst. of Tech., Dept. of Aeronautics and Astronautics., Cambridge, MA USA

Aerodynamics of heat exchangers for high-altitude aircraft

Drela, Mark, Massachusetts Inst. of Tech., USA; Apr. 1996; 10p; Repr. from Journal of Aircraft (Washington,D.C.), v.33, no. 2, Mar. - Apr. 1996 p 176-184; In English; 13th; AIAA Applied Aerodynamics Conference, 19-22 Jun. 1995, San Diego, CA, USA; Sponsored by American Inst. of Aeronautics and Astronautics, USA

Contract(s)/Grant(s): NAG2-4008

Report No.(s): NASA-CR-201078; NAS 1.26:201078; AIAA Paper 95-1866; Copyright Waived (NASA); Avail: CASI; A02, Hardcopy; A01, Microfiche

Reduction of convective beat transfer with altitude dictates unusually large beat exchangers for piston- engined high-altitude aircraft The relatively large aircraft drag fraction associated with cooling at high altitudes makes the efficient design of the entire heat exchanger installation an essential part of the aircraft's aerodynamic design. The parameters that directly influence cooling drag are developed in the context of high-altitude flight Candidate wing airfoils that incorporate heat exchangers are examined. Such integrated wing-airfoil/heat-exchanger installations appear to be attractive alternatives to isolated heat.exchanger installations. Examples are drawn from integrated installations on existing or planned high-altitude aircraft.

Author

Heat Exchangers; Aerodynamic Drag; Aircraft Design; Airfoil Profiles; Aerodynamic Heat Transfer; Convective Heat Transfer; Inlet Airframe Configurations; High Altitude

03
AIR TRANSPORTATION AND SAFETY

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

19960025214; 96N27294 Advisory Group for Aerospace Research and Development, NATO, The Aerospace Medical Panel., Neuilly-Sur-Seine, France

Aircraft Disinsection: A Guide for Military & Civilian Air Carriers *Desinsectisation des aeronefs: Un guIde a l'intention des responsables des transports aeriens civils et militaires*

Ellis, R. A., Advisory Group for Aerospace Research and Development, France; Apr. 1996; 84p; In English

Report No.(s): AGARD-AG-340; ISBN 92-836-1036-9; Copyright Waived; Avail: CASI; A05, Hardcopy; A01, Microfiche

Aircraft disinsection is an important tool in preventing the introduction of unwanted pests into a country when an aircraft is returning to its home base. To prevent risks to air crew health, aircraft safety, and industry, Canada's Department of National Defence (DND) has reviewed the importance of aircraft disinsection and the potential problems associated with execution. Over the past two decades, various directives for air crew maintenance personnel, and preventive medicine technicians have been developed and updated periodically. This aircraft disinsection review is part of the latest effort to revise DND's administrative orders on aircraft disinsection. Existing Canadian and foreign legislation, regulations, and recommendations dealing with aircraft disinsection were reviewed. This review also summarises the information that was gathered from various officials involved in the regulation of introduced pests, pesticide registration, and safe pesticide use. Aircraft disinsection technology has evolved over the years since its inception. Practical, up-to-date information on current technologies was gathered, through numerous meetings and correspondence, from researchers, private companies involved in aircraft disinsection, air force personnel, and representatives of civilian air carriers who are active worldwide. The end-result was the development of a current, standard operating procedure for disinsection of Canadian Air

Force aircraft in the form of an Air Command Administrative Order. It may serve as a model for the Air Forces and air carriers of other NATO countries.
Author
Military Aircraft; Decontamination; Commercial Aircraft; Insects; Aircraft Maintenance; Regulations

19960025854; 96N27715 National Aerospace Lab., Structural Mechanics Div., Tokyo, Japan
Construction and Analysis of Aircraft Accident Database and Case Studies Related to Failures of Materials and Structures
Terada, Hiroyuki, National Aerospace Lab., Japan; Ohmachi, Shigeru, Metropolitan Police Dept., Japan; Ohmo, Kayo, National Aerospace Lab., Japan; Arai, Osamu, Shibaura Inst. of Tech., Japan; Nakao, Hiroshi, Miura Industry Co. Ltd., Japan; Nov. 1995; ISSN 0389-4010; 32p; In Japanese
Report No.(s): NAL-TR-1279; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The worldwide aircraft accident database, which covers most of the fatal or hull-loss accidents of aircraft heavier than 5,700 kg since 1954, has been developed to study the most critical factors causing accidents and to improve air transport safety. As examples of accident data analyses, major accident causes, secondary factors inducing pilot errors which are responsible for most accidents, and the chronological changes in accident patterns have been discussed. Moreover, interest was primarily focused on the accidents which were induced by inadequate design or manufacturing or by defects of materials. The essential factors causing structural failure accidents were studied by close examination of case studies of typical accidents. These factors are categorized into four groups: Man, Machine, Management, and Environment. It was pointed out that the carelessness of the maintenance staff, defects during manufacturing and improper management on the part of the operator or the authority are also major affecting elements for the accidents induced by structural failures.
Author
Aircraft Accidents; Data Bases; Structural Failure; Pilot Error; Failure Analysis

19960026459; 96N28106 Crew System Ergonomics Information Analysis Center, Wright-Patterson AFB, OH USA
Flightdeck Crew Alerting Issues: An Aviation Safety Reporting System Analysis *Progress Report, Aug. 1993 - Mar. 1994*
Rehmann, Albert, Crew System Ergonomics Information Analysis Center, USA; Mitman, Robert D., Crew System Ergonomics Information Analysis Center, USA; Neumeier, Mark, Crew System Ergonomics Information Analysis Center, USA; Reynolds, Michael C., Crew System Ergonomics Information Analysis Center, USA; Oct. 1995; 122p; In English
Contract(s)/Grant(s): DLA900-88-D-0393
Report No.(s): AD-A302250; DOT/FAA/CT-TN94/18; No Copyright; Avail: CASI; A06, Hardcopy; A02, Microfiche

This document describes an analysis of the Aviation Safety Reporting System (ASRS) database regarding flightdeck crew alerting deficiencies. The ASRS database contains thousands of reports concerning actual or potential deficiencies, which may compromise the safety of aviation operations in the National Aviation System. This analysis searched the ASRS database for incidents of flight technical errors resulting from confusion distraction, or annoyance associated with the sounds and lights present in the cockpit. The analysis of the ASRS reports produced six major crew alerting problem areas: (1) distraction of alerts; (2) missed alerts; (3) lack of alerts; (4) alert inhibit logic; (5) non-distinguishable alerts; and (6) multiple alerts. These problem areas resulted in a variety of flight technical errors, such as altitude and heading deviations, attention deviations, and aborted takeoffs. The crew alerting problems indicated in the ASRS reports are examined in de tail, and any pertinence to the design of the Data Link system is concluded.
DTIC
Aircraft Safety; Flight Safety; Flight Crews; Human Factors Engineering; Alertness

19960026647; 96N28214 Akron Univ., ,AkronOH USA
Structural analysis and properties of impact ices accreted on aircraft structures
Scavuzzo, R. J., Akron Univ., USA; Chu, M. L., Akron Univ., USA; Kellackey, C. J., Akron Univ., USA; Apr. 1996; 48p; In English
Contract(s)/Grant(s): NAG3-479; RTOP 505-68-10
Report No.(s): NASA-CR-198473; NAS 1.26:198473; E-10195; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This final contractor report presents a summary of work done on the experimental measurements of the mechanical properties of impact ices and analytical studies of ice stresses and mechanical deicing systems. Experimental work on the adhesive shear strength, the peel strength, the bending strength and tensile strength of impact ices. The major analytical studies that were completed are as follows: (1) Ice shedding by the EIDI system, (2) Impact Ice stresses in rotating airfoils, (3) Aerodynamic forces in ice shedding, (4) Interface shear stresses from bending and twisting, and (5) Statistical structural analysis of ice shedding from rotating airfoils. In this report, results and data from each experimental and analytical study are summarized. A total of twenty (20) technical papers and one MS Thesis were written and/or presented from the work done in this project.
Author
Mechanical Properties; Ice; Aircraft Structures; Structural Analysis; Deicers; Airfoils; Ice Prevention

19960027240; 96N28681 Galaxy Scientific Corp., Pleasantville, NJ USA

The Role of Research and Development on Safety Regulation *Final Report*

Westfield, William T., Galaxy Scientific Corp., USA; Oct. 1995; 34p; In English

Report No.(s): AD-A302895; DOT/FAA/AR-95/84; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A review of regulatory actions taken by the Federal Aviation Administration (FAA) over approximately the past thirty years was made to identify which of these actions were preceded by, or triggered by research and development (R&D) programs. The focus of this analysis was limited to those actions and R&D that pertained to aircraft safety. Research was frequently requested to be performed by the industry elements themselves, namely the engine and aircraft manufacturers, independent research organizations, or by academia. As the FAA experience with the operational fleets grew, some research was accomplished by the internal FAA and governmental organizations, such as the Civil Aeromedical Institute or the Technical Center, and the National Aeronautics and Space Administration. The review revealed that the rulemaking actions were supported quite frequently by research. While it is true that major accidents brought the attention to technology unknowns, and the subsequent rulemaking to prevent the problem appeared to be moving too slowly to provide solutions to the need, the review showed many risks have been reduced, (cabin fires, inadvertent collision with terrain, faster evacuation, to name a few) through the performance of research.

DTIC

Aircraft Safety; Commercial Aircraft; Transport Aircraft; Research and Development; Regulations

19960027530; 96N28893 Federal Aviation Administration, Civil Aeromedical Inst., Oklahoma City, OK USA

Aviation accident forensic assessment: Comprehensive single-extraction urine screening procedure *Final Report*

Canfield, Dennis, Federal Aviation Administration, USA; White, Vicky, Federal Aviation Administration, USA; Soper, John, Federal Aviation Administration, USA; Kupiec, Tom, Federal Aviation Administration, USA; May 1996; 16p; In English

Report No.(s): DOT/FAA/AM-96/17; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

One of the missions of the Civil Aeromedical Institute (CAMI), Office of Aviation Medicine (OAM), Federal Aviation Administration (FAA), is to help assess the role of potential medical or drug related pilot impairment in aviation accidents. This requires the ability to identify a wide range of drugs and the medical conditions for which these drugs are prescribed. A single extraction screening procedure was developed to identify as many drugs as possible in urine, with minimal effort and cost. Triamterene, a diuretic, is easily identified using HPLC and TLC, whereas it can not be easily identified using standard GC mass spectroscopy, On the other hand, atenolol is easily identified using HPLC and mass spectroscopy, but is not detected using TLC at therapeutic levels. Methadone is difficult to detect using HPLC with a photo diode array and fluorescence detector, but can easily be detected using TLC and confirmed by mass spectroscopy. Urine specimens are hydrolyzed using betta-glucuronidase, and the specimen is then extracted using commercially purchased TOXI-LAB A (R) extraction tubes. After removing the TOXI-LAB (R) organic phase, the remaining aqueous layer is washed with chloroform. The organic phase and the chloroform wash are combined, evaporated to dryness, and then reconstituted in 5O milliliter of a methanol solution. A 10 milliliter quantity of the methanol solution is then chromatographed, using HPLC with a photo diode array and fluorescence detector. The methanol solution is next analyzed using either TOXI-LAB (R) or GC mass spectroscopy, depending on the compound detected using HPLC. When HPLC is negative, the extract is analyzed using the TOXI-LAB (R) procedure. This method has proven successful in the identification of benzodiazepines (alprazolarn, temazepam, diazepam, nordiazepam, and oxazepam), cardiovascular medications, antihistamines, and other drugs that could have been missed using only one of the methods listed above. The number of positive benzodiazepines identified has increased as a result of using this new method, and the time required to complete cases has been reduced substantially. Using a single aliquot, extraction, and analysis of a urine specimen by several different methods makes it possible to screen for a wide variety of drugs. Screening for these drugs would typically require several different sample aliquots and procedures. The new procedure helps prevent false negative results, that might lead investigators in aviation accidents to the wrong conclusions. The fact that drugs can be uniquely identified by HPLC, and then confirmed using GC mass spectroscopy, reduces the time necessary to complete cases by reducing the number of aliquots needed to specifically identify the drugs present in the body.

Author

Aircraft Accidents; Urine; Aerospace Medicine; Photodiodes; Fluorescence; Liquid Chromatography; Drugs; Mass Spectroscopy

19960027862; 96N29005 Federal Aviation Administration, Airport and Aircraft Safety Research and Development Div., Atlantic City, NJ USA

Proposed Cabin Safety Research Program (Transport Category Airplanes) *Final Report*

Hill, Richard, Federal Aviation Administration, USA; Oct. 1995; 69p; In English

Report No.(s): AD-A301693; DOT/FAA/AR-95/14; TP-12570; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

Cabin safety presents challenges common to all aviation authorities. Related issues and needed research must be accomplished through a totally integrated program. to enhance their respective research capabilities, the U.S. Federal Aviation Administration (FAA) and the Transport Canada Aviation (TCA), the aviation authorities of North America, and the Joint Aviation Authorities (JAA), the aviation authority of Europe, have, as they have been doing in rulemaking, agreed to cooperate in research on transport category airplane cabin safety. The FAA/JAA/TCA Cabin Safety Research Program is the formalization of this agreement. Specifically, the goal of the Cabin Safety Research Program is to provide a mechanism for the coordination of pertinent activities and, as appropriate, the conduct of cooperative, joint, and complementary programs to the benefit of the three authorities. For the purpose of this program, cabin safety is intended to address acute events/conditions which can be dealt with by changes within (or closely associated with) the cabin. Although in-flight issues form an integral part of cabin safety, the primary focus is postcrash survivability, the principal elements of which are structural crashworthiness, fire safety, evacuation, and overwater survival. The formost decision-making tools to identify and assess the potential benefits of needed research (and of past improvements) are a probabilistic risk analysis model and a cabin safety accident/incident information data bank/base.

DTIC

Transport Aircraft; Aircraft Compartments; Aircraft Safety; Crashworthiness; Research; Fire Prevention; Flight Safety

19960028015; 96N29120 NASA Langley Research Center, Hampton, VA USA

A preliminary study of a wake vortex encounter hazard boundary for a B737-100 airplane

Reimer, Heidi M., NASA Langley Research Center, USA; Vicroy, Dan D., NASA Langley Research Center, USA; Apr. 1996; 18p; In English

Contract(s)/Grant(s): RTOP 505-64-13-02

Report No.(s): NASA-TM-110223; NAS 1.15:110223; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A preliminary batch simulation study was conducted to define the wake decay required for a Boeing 737-100 airplane to safely encounter a Boeing 727 wake and land. The baseline six-degree-of-freedom B737 simulation was modified to include a wake model and the strip-theory calculation of the vortex-induced forces and moments. The guidance and control inputs for the airplane were provided by an autoland system. The wake strength and encounter altitude were varied to establish a safe encounter boundary. The wake was positioned such that the desired flight path traversed the core of the port Vortex. Various safe landing criteria were evaluated for defining a safe encounter boundary. A sensitivity study was also conducted to assess the effects of encounter model inaccuracies.

Author

Boeing 737 Aircraft; Vortices; Aircraft Wakes; Flight Paths; Spacing; Aircraft Hazards; Flight Hazards

19960028949; 96N29552 Naval Postgraduate School, Monterey, CA USA

A decision support system for naval aviation mishap investigation and reporting

Emde, Charles E., Naval Postgraduate School, USA; Sep. 1995; 127p; In English

Report No.(s): AD-A303699; No Copyright; Avail: CASI; A07, Hardcopy; A02, Microfiche

This thesis discusses the implementation of a prototype of a decision support system (DSS) for aviation mishap reporting. The Naval Aviation Safety Program, as defined by OPNAVINST 3750.6Q is a complete reference for the reporting requirements for naval aviation mishap reporting, and this thesis augments the reporting requirements by defining a mishap investigation heuristic in the form of a logical model. This model directly addresses problems an investigator may encounter in the course of a mishap investigation such as logical omissions and incomplete deduction or investigation. Typically, mishap investigators are faced with numerous, unorganized pieces of evidence which develop into a complex web of interrelationships which recreate the events which caused the mishap. Our mode l suggests a process which organizes evidence and cause factors, and then we automate the model in a decision support prototype. The system also addresses the ’administrative overhead’ of a mishap by outlining the architecture of a complete system providing facilities for initial and final mishap reporting in addition to the automation of the deliberation model.

DTIC

Aircraft Safety; Flight Safety; Applications Programs (Computers); Artificial Intelligence; Heuristic Methods

19960029316; 96N29776 NASA Lewis Research Center, Cleveland, OH USA

Shear Strength Analysis of the Aluminum/Ice Adhesive Bond

Lynch, Denis A., III, Notre Dame Univ., USA; Ludwiczak, Damian R., NASA Lewis Research Center, USA; May 1996; 66p; In English

Contract(s)/Grant(s): RTOP 274-00-00

Report No.(s): NASA-TM-107162; E-10112; NAS 1.15:107162; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

The shear strength of an aluminum/ice adhesive bond is analyzed for the purpose of gaining an appreciation for the forces necessary to break this bond. The results of this work have a direct bearing on future attempts to develop de-icing systems for aircraft. Results of testing indicate that system tem-

perature, bonding surface conditions, and ice purity profoundly influence the adhesion strength. An explanation of testing parameters and procedures are presented, along with a detailed characterization of the results and suggestions for further study.
Author
Ice Formation; Shear Strength; Adhesive Bonding; Aircraft Icing; Metal Surfaces; Static Tests; Tensile Tests

19960031961; 96N29949 Galaxy Scientific Corp., Atlanta, GA USA
Results of a field study of the prformance enhancement system: A support system for aviation safety inspectors *Final Report*
Shepherd, William T., Federal Aviation Administration, USA; Layton, Charles F., Galaxy Scientific Corp., USA; Dec. 1995; 84p; In English
Contract(s)/Grant(s): DTFA01-92-Y-01005
Report No.(s): AD-A303336; DOT/FAA/AM-95/31; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

The Performance Enhancement System (PENS) is a prototype electronic performance support system for Aviation Safety Inspectors (ASIs). PENS facilitates field data collection, information management, and on-line references, thus eliminating paperwork, redundant data-entry tasks, data errors, slow data-entry and turnaround times, and bulky paper references. PENS allows ASIs to collect field data in the format used by national FAA databases and it allows them to verify data at the time of inspection. A national field study of PENS was conducted. The study consisted of fielding four different computers to four ASIs in each of nine FAA Flight Standards District Offices across the country from November 1993 through March 1994. Study results indicate that the complex information management difficulties encountered by ASIs will not be solved by purchasing computer hardware alone. Rather, appropriate field software applications must be developed that will work in combination with field computer hardware to provide ASIs with a system of tools that supports their daily responsibilities. The ASIs must be closely involved in the continued development of these tools to ensure that they are designed to meet the inspectors' needs.
DTIC
Performance Tests; Data Acquisition; Aircraft Safety

19960031980; 96A62696 NASA Lewis Research Center, Cleveland, OH, USA
Characteristics of surface roughness associated with leading-edge ice accretion
Shin, Jaiwon, NASA Lewis Research Cent, USA; Journal of Aircraft; March 1996; ISSN 0021-8669; vol. 33, no. 2, pp. 316-321; In English; Copyright; Avail: Issuing Activity

Detailed measurements of surface roughness dimensions associated with leading-edge ice accretions are presented to provide information on characteristics of roughness and trends of roughness development with various icing parameters. Data were obtained from icing tests conducted in the Icing Research Tunnel at NASA Lewis Research Center using a NACA 0012 airfoil. Dimensions measured include diameters, heights, and spacing of roughness elements along with chordwise icing limits. Results confirm the existence of smooth and rough ice zones and that the boundary between the two zones (surface roughness transition region) moves upstream towards stagnation region with time. The heights of roughness elements increase as the total temperature and the liquid water content increase; however, the airspeed has little effect on the roughness height. Results also show that the roughness in the surface roughness transition region grows during a very early stage of accretion, but reaches a critical height and then remains fairly constant. Results also indicate that a uniformly distributed roughness model is only valid at a very early stage of the ice accretion process.
Author (EI)
Airfoils; Angle of Attack; Ice; Ice Formation; Surface Roughness; Test Facilities

19960033235; 96N30049 Air Weather Service, Air Force Combat Climatology Center., Scott AFB, IL USA
A comparison of aircraft icing forecast models
Cornell, Daniel, Air Weather Service, USA; Donahue, Christopher A., Air Weather Service, USA; Chan, Keith, Air Weather Service, USA; Dec. 1995; 40p; In English
Report No.(s): AD-A303307; AFCCC-TN-95-004; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This report documents the results of a study comparing forecasted aircraft icing conditions to pilot reports (PIREPs) of aircraft icing. of the three operational forecast algorithms validated (AF Global Weather Central's RAOB routine, icing routines in the Navy's NODDS, and TESS software), RAOB had the overall best comparison to the PIREPs and 6% percent of the type forecasts and 42 percent of the intensity forecasts agreeing with the reported icing. These percentages increase to 73 percent (type) and 53 (intensity) when considering only PIREPs from jet aircraft. The greatest degree of error was associated with forecasts of clear and mixed icing and icing of moderate intensity.
DTIC
Jet Aircraft; Ice Formation; Forecasting; Aircraft Icing

04
AIRCRAFT COMMUNICATIONS AND NAVIGATION

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

19960024304; 96N26832 NASA Langley Research Center, Hampton,VA USA
Flight Evaluation of the CTAS Descent Advisor Trajectory Prediction
Oseguera, Rosa M., NASA Langley Research Center, USA;

Williams, David H., NASA Langley Research Center, USA; Jun. 1995; 10p; In English; American Control Conference, 21-23 Jun. 1995, Seattle, WA, USA
Report No.(s): NASA-TM-111572; NAS 1.15:111572; No Copyright; Avail: CASI; A02, Hardcopy; A01, Microfiche

Issues relating to integration of the Center TRACON Automation System (CTAS), a ground-based system for automated management and control of terminal area traffic, with airborne flight management systems (FMS), are reported. Flight test conducted in Sept. 1994 is described.
CASI
Flight Management Systems; Descent Trajectories; Flight Tests; Air Traffic Control; Systems Integration; Automatic Flight Control

19960024356; 96N26871 Western Aerospace Labs., Inc., Monte Sereno, CA USA
Operational Issues During Low-Visibility Taxi Operations: A Field Study ***Final Report***
Andre, Anthony D., Principal Investigator, Western Aerospace Lab., Inc., USA; May 06, 1996; 9p; In English
Contract(s)/Grant(s): NCC2-486
Report No.(s): NASA-CR-201010; NAS 1.26:201010; No Copyright; Avail: CASI; A02, Hardcopy; A01, Microfiche

Low-visibility conditions present a host of problems for the National Airspace System. While many modem aircraft are equipped with automation that allows them to land under low-visibility conditions, there is no such corresponding technology to aid the pilots in taxiing the aircraft from runway to gate, or vice versa. Consequently, flight throughput and sequencing is severely constrained, especially at the major airports. Current efforts within NASA, the FAA, and the commercial aviation industry are aimed at developing technologies to increase the efficiency and safety of taxi operations under low-visibility conditions. Based on cockpit observations, pilot interviews, and pilot-controller communications, this paper presents an analysis of current problems experienced by pilots during the taxi operations, and presents their views on key issues related to the introduction of electronic taxi map displays in the cockpit. The data were collected by the author while serving as a flight deck observer aboard thirty-five commercial carrier flights. The implications of these data for the justification and design of advanced cockpit displays for taxi operations are discussed, and the importance of including pilot experiences, opinions, and attitudes in the research and design process is stressed.
Author
Safety; Airfield Surface Movements; Low Visibility; Air Traffic Control; Approach Control; Control Systems Design; Taxiing; Aircraft Safety

19960026452; 96N28100 Federal Aviation Administration, Technical Center., Atlantic City, NJ USA
National Simulation Capability Reduced Vertical Separation Minima. Phase 1: Result Report ***Progress Report, Dec. 1993 - Jan. 1994***
Seeger, Diena, Federal Aviation Administration, USA; Roske-Hofstrand, R., Federal Aviation Administration, USA; Aug. 1995; 84p; In English
Report No.(s): AD-A302726; DOT/FAA/CT-TN95/20; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

The Reduced Vertical Separation Minima (RVSM) experiment resulted from the North Atlantic (NAT) System Planning Group's conclusion to carry out studies aimed at achieving early implementation of RVSM in the NAT Region. RVSM is an approved International Civil Aviation Organization concept to reduce aircraft vertical separation from the Conventional Vertical Separation Minima (CVSM) of 2000 feet to 1000 feet. This reduction occurs between flight level 290 to 410, within a designated portion of the NAT Region. Phase I investigated workload changes resulting from the transition of westbound aircraft from RVSM to CVSM before leaving defined non-radar RVSM airspace. The simulated New York Air Route Traffic Control Center Oceanic Sectors D71 and D72 were configured with an Oceanic Display and Planning System position to replicate controller operations, including simulated high frequency and inter- and intra-facility communications. The RVSM procedure increased the amount of available altitudes, thus providing the controller with greater flexibility for managing traffic. However, simulation results indicated that controllers operating under RVSM conditions experienced increa sed coordination requirements, longer display scanning times, and needed additional information from aircraft as compared to CVSM.
DTIC
Air Traffic Control; Air Navigation; Civil Aviation; Controllers; Air Traffic

19960026580; 96N28150 Department of the Navy, Washington, DC USA
Waypoint Navigation Using Exclusion Zones
Jones, Gregory B., Inventor, Department of the Navy, USA; Shaw, Christophe, Inventor, Department of the Navy, USA; Hills, Stacy J., Inventor, Department of the Navy, USA; Jul. 19, 1995; 18p; In English
Patent Info.: US-Patent-Appl-SN-504374
Report No.(s): AD-D017777; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A method is provided for navigating a vehicle. The waypoint exclusion zones are defined as circles whose centers are known position waypoints. The vehicle is steered along a path that is tangential to the 'current' waypoint exclusion zone. This path that is maintained until a relative bearing between the vehicle and the center of the current waypoint exclusion zone is at least 90 deg if the path is left of the center of the current waypoint exclusion zone, and at most - 90 deg if the path is right of the center of the current waypoint exclusion zone. The vehicle is than advanced along the circle of the waypoint

exclusion zone until a heading of the vehicle matches a heading of a path that is tangential to the 'next' waypoint exclusion zone. When the heading of the vehicle matches the heading of the path that is tangential to the next waypoint exclusion zone, the next waypoint exclusion zone matches waypoint exclusion zone fo carrying out the steering, maintaining and advancing on the vehicle.
DTIC
Area Navigation; Steering; Autonomous Navigation; Regions

19960026670; 96N28235 NASA Langley Research Center, Hampton, VA USA
Enhancing pilot situational awareness of the airport surface movement area
Jones, D. R., NASA Langley Research Center, USA; Young, S. D., NASA Langley Research Center, USA; Nov. 03, 1994; 10p; In English; 13; Digital Avionics Systems Conference, 31 Oct. - 3 Nov. 1994, Phoenix, AZ, USA
Report No.(s): NASA-TM-111525; NAS 1.15:111525; No Copyright; Avail: CASI; A02, Hardcopy; A01, Microfiche

Two studies are being conducted to address airport surface movement area safety and capacity issues by providing enhanced situational awareness information to pilots. One study focuses on obtaining pilot opinion of the Runway Status Light System (RSLS). This system has been designed to reduce the likelihood of runway incursions by informing pilots when a runway is occupied. The second study is a flight demonstration of an rate integrated system consisting of an electronic moving map in the cockpit and display of the aircraft identification to the controller. Taxi route and hold warning information will be sent to the aircraft data link for display on the electronic moving map. This paper describes the plans for the two studies.
Author
Airports; Safety; Plat System; Pilot Performance; Air Traffic Control

19960027014; 96N28516 NASA Langley Research Center, Hampton, VA USA
Navigating the airport surface: Electronic vs. paper maps
Batson, Vernon M., NASA Langley Research Center, USA; Harris, Randall L., Sr., NASA Langley Research Center, USA; Hunt, Patricia J., Lockheed Engineering and Sciences Co., USA; Nov. 03, 1994; 10p; In English; 13th; Digital Avionics Systems Airport, 31 Oct. - 3 Nov. 1994, Phoenix, AZ, USA
Report No.(s): NASA-TM-111536; NAS 1.15:111536; No Copyright; Avail: CASI; A02, Hardcopy; A01, Microfiche

Recent advances in the Differential Global Positioning System (DGPS) and ground/aircraft data-links provide a basis for the generation of an accurate cockpit navigational map display including data-linked ATC-cleared ground routes. Such an electronic map may have the potential to improve pilots' situation awareness and taxi performance and thereby lessen runway incursions. The objective of this simulator study was to assess the potential improvements in these areas when using an advanced electronic map (compared to using today's paper map) under two outside scene visibility levels. Results showed average taxi speed increased under both good and poor visibilities, by as much as 24 percent, due in part to eliminating the time used for orientation with the paper map. Pilots made only one-third as many errors as well and commented that they believed that the electronic map gave them better awareness.
Author
Airports; Display Devices; Pilot Performance; Taxiing; Aircraft Instruments; Computer Aided Mapping

19960027229; 96N28670 Dunlap and Associates, Inc., Stamford, CT USA
Evaluation of Prototype Air Carrier Instrument Approach Procedure Charts *Final Report, Jan. 1993 - May 1995*
Blomberg, R. D., Dunlap and Associates, Inc., USA; Bishop, E. W., Dunlap and Associates, Inc., USA; Hamilton, J. W., Dunlap and Associates, Inc., USA; Jul. 1995; 210p; In English
Report No.(s): AD-A302728; DOT/VNTSC-FAA-95-24; DOT/FAA-95/44; No Copyright; Avail: CASI; A10, Hardcopy; A03, Microfiche

The objective of this study was to evaluate the design features of two prototype Instrument Approach Procedure (IAP) charts. The John A. Volpe National Transportation System's Center in cooperation with the Air Transport Association's Chart and Data Display Working Group prepared one of the prototypes, and the other was produced by Jeppesen Sanderson. The study was designed to compare each prototype to the current Jeppesen chart, and to compile the results in the form of recommendations for revision of one of the prototypes or for the design of a new IAP chart. Pilots' opinions of the speed and accuracy with which they could acquire and use the information shown and their preferences for the unique features of each were the primary evaluative measures. Scaling techniques were used to obtain the pilots' opinions, preferences and assessments of each chart's suitability for use in approach briefing, normal approach, and go-around. A total of 91 air carrier pilots from four airlines operating under Part 121 of the Federal Aviation Regulations took part in the study. The results showed that while the pilots were resistant to changes in approach chart design before seeing either of the prototypes, they accepted many of the new features of both prototypes and suggested additional changes after they were introduced to the new ideas. The recommendations for an improved instrument approach chart include features from both prototypes. The recommended chart design is compatible with current chart production and distribution procedures, and it is feasible to introduce it on a phased basis, changing all of the charts at any given airport at one time. It

presents the information needed to brief an approach in a 'Briefing Strip' at the top of the chart. This strip includes a pictorial of the lighting to be expected on the runway being used.
DTIC
Instrument Approach; Civil Aviation; Commercial Aircraft; Aircraft Landing; Human Factors Engineering; Crew Procedures (Inflight)

19960028980; 96N29579 Air Force Inst. of Tech., Wright-Patterson AFB, OH USA
A comparison of loose and tight GPS/INS integration using real INS and GPS data
Nuibe, Warren H., Air Force Inst. of Tech., USA; Dec. 1995; 91p; In English
Report No.(s): AD-A303424; AFIT/GE/ENG/95D-19; No Copyright; Avail: Issuing Activity (Defense Technical Information Center (DTIC)), Microfiche

An extended Kalman filter (EKE) is used to combine the information obtained from a Global Positioning System (GPS) receiver and an Inertial Navigation System (INS) to provide a navigation solution. This research compares the results of a tightly-coupled GPS/INS integrated system with a loosely-coupled integrated system, using real world data. A fair comparison is accomplished by using the same sets of data, and keeping the integration structures as close as possible. Both integrations are feedforward and have the same error states in the navigation Kalman filters. Differences between the two, such as navigation solutions and tuning values, are shown in the research.
DTIC
Kalman Filters; Global Positioning System; Inertial Navigation; Data Integration; Receivers

19960029336; 96N29790 Organisatie voor Toegepast Natuurwetenschappelijk Onderzoek, TNO Human Factors Research Inst., Soesterberg, Netherlands
An ECDIS/track predictor prototype on top of the GEO++ system ***Interim Report***
Passenier, P. O., Organisatie voor Toegepast Natuurwetenschappelijk Onderzoek, Netherlands; Vijlbrief, C., Organisatie voor Toegepast Natuurwetenschappelijk Onderzoek, Netherlands; Nov. 28, 1995; 32p; In English
Contract(s)/Grant(s): A90/KM/332
Report No.(s): TNO-TM 1995 A-54; TD 95-1126; Copyright; Avail: Issuing Activity (TNO Human Factors Research Inst., Kampweg 5, 3769 DE Soesterberg, Netherlands), Hardcopy, Microfiche

Under contract to the Royal Netherlands Navy (contract no. A90/KM/332), the TNO Human Factors Research Institute investigates new user-interface concepts for ECDIS-based navigation support systems. On the basis of a two-stage description of track planning, a user interface has been designed for an integrated ECDIS/track-prediction system which assists the master in the preparation of maneuvers. Both ECDIS display and track-prediction interface conform to existing specifications (IHO respectively OSF/Motif), so that the interface can be integrated with existing (commercial) ECDIS systems. Regarding further research, it is recommended to focus attention on the information presentation during the execution of a maneuver, in order to assist the master in a fast detection of deviations between predictions and observations on the basis of which feedback corrections can be determined.
Author
Navigation; Support Systems; Man Machine Systems; Trajectory Planning; Tracking (Position)

19960029350; 96N29802 Air Force Flight Test Center, Edwards AFB, CA USA
Investigation of Using Global Positioning for Air Data System Calibration of General Aviation Aircraft (HAVE PACER 2) ***Final Report, 15 Sep. - 27 Oct. 1995***
Bailey, William D., Air Force Flight Test Center, USA; Knoedler, Andrew J., Air Force Flight Test Center, USA; Harris, David A., Air Force Flight Test Center, USA; McClintock, Bruce H., Air Force Flight Test Center, USA; Jan. 1996; 63p; In English
Report No.(s): AD-A303524; AFFTC-TR-95-76; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

This report presents the results of an evaluation of the suitability of using handheld Global Positioning System (GPS) receivers using course acquisition code to perform an air data system calibration. The two testbeds were an aerospatiale Tobago and a Rutan Long EZ, both general aviation single engine land aircraft. The test objective was to evaluate the suitability of commercial GPS receivers as measuring devices for general aviation air data system (ADS) calibration. Emphasis was on a handheld GPS and data collection devices. The goal was to develop a technique to accomplish an ADS calibration on a low-speed (under 200 knots calibrated airspeed) aircraft, using only commercial GPS equipment. After an all-altitude airspeed comparison technique was developed, a calibration of the ADS of an uninstrumented general aviation aircraft was completed. Overall, the commercial OPS receivers were suitable as measurement devices for ADS calibrations. Any current, commercially available receiver could be used without the need for differential GPS receivers or rigorous postprocessing of receiver data.
DTIC
Global Positioning System; Air Data Systems; General Aviation Aircraft

19960029448; 96N29866 Naval Postgraduate School, Monterey, CA USA
Differential GPS for precision approach: Commercial technology and Navy/Marine corps requirements
Eipp, Thomas B., Naval Postgraduate School, USA; Jun. 1995; 118p; In English

Report No.(s): AD-A302976; No Copyright; Avail: Issuing Activity (Defense Technical Information Center (DTIC)), Microfiche

Currently, the Department of Defense (DoD) uses several Precision Landing Systems (PLS) including the Instrument Landing System (ILS), Automatic Carrier Landing System (ACLS), and Precision Approach Radar (PAR). Each system requires different avionics, ground station equipment and are not universally implemented in the different services. This has lead to interoperability problems among the services. Additionally, these landing systems have numerous deficiencies, which include deployablility, manpower requirements, and frequency congestion. Therefore, a new Precision Landing System is necessary to meet DoD requirements. An evaluation of several different Differential GPS systems was performed. This evaluation involved a comparison of system capabilities against the requirements established by the Federal Aviation Administration (FAA) and the DoD requirements. The results showed that most Commercial Off The Shelf (COTS) Differential GPS systems meet or exceed the requirements identified by the DoD.

DTIC

Global Positioning System; Microwave Landing Systems; Ground Based Control; Attitude Control; Kalman Filters; Cost Effectiveness; Landing Aids

05
AIRCRAFT DESIGN, TESTING AND PERFORMANCE

Includes aircraft simulation technology.

19960024355; 96N26870 Army Aviation Systems Command, Hampton, VA USA

Free Flight Rotorcraft Flight Test Vehicle Technology Development

Hodges, W. Todd, Army Aviation Systems Command, USA; Walker, Gregory W., Army Aviation Systems Command, USA; 1994; 12p; In English

Report No.(s): NASA-TM-111571; NAS 1.15:111571; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A rotary wing, unmanned air vehicle (UAV) is being developed as a research tool at the NASA Langley Research Center by the U.S. Army and NASA. This development program is intended to provide the rotorcraft research community an intermediate step between rotorcraft wind tunnel testing and full scale manned flight testing. The technologies under development for this vehicle are: adaptive electronic flight control systems incorporating artificial intelligence (AI) techniques, small-light weight sophisticated sensors, advanced telepresence-telerobotics systems and rotary wing UAV operational procedures. This paper briefly describes the system's requirements and the techniques used to integrate the various technologies to meet these requirements. The paper also discusses the status of the development effort. In addition to the original aeromechanics research mission, the technology development effort has generated a great deal of interest in the UAV community for related spin-off applications, as briefly described at the end of the paper. In some cases the technologies under development in the free flight program are critical to the ability to perform some applications.

Author

Rotary Wing Aircraft; Artificial Intelligence; Telerobotics; Man Machine Systems; Free Flight; Flight Test Vehicles; Flight Control; Electronic Control

19960024383; 96N26896 Computer Technology Associates, Inc., McKee City, NJ USA

Report of Study on Airlines' Anticipated Near Future Cockpit Control and Display Capabilities and Plans for Data Link Communication, Part 2

Rehmann, Albert J., Computer Technology Associates, Inc., USA; Townsend, S. L., Computer Technology Associates, Inc., USA; Yastrop, G., Computer Technology Associates, Inc., USA; Oct. 1995; 42p; In English

Report No.(s): AD-A302151; DOT/FAA/CT-TN95/31-Pt-2; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This report reviews the findings of a study conducted by CTA Incorporated for the Federal Aviation Administration (FAA) Technical Center Airborne Systems Technology Branch (ACD-320) Airborne Data Link Program. The following data were studied for each of the seven airlines' anticipated near future fleets: (1) quantity of each aircraft model; (2) aircraft model cockpit configuration; (3) currently available data link services; (4) primary data link device location; and (5) Data link equipment training needs. In addition, information was gathered that pertained to the airlines' interest in linking their simulators to the airlines' specific plans for incorporating Data Link services into their fleets.

DTIC

Commercial Aircraft; Data Links; Airline Operations; Display Devices; Avionics; Aircraft Communication

19960024961; 96N27088 NASA Langley Research Center, Hampton,VA USA

An Evaluation of the Measurement Requirements for an In-Situ Wake Vortex Detection System

Fuhrmann, Henri D., NASA Langley Research Center, USA; Stewart, Eric C., NASA Langley Research Center, USA; May 1996; 26p; In English

Contract(s)/Grant(s): RTOP 505-69-59-04

Report No.(s): NASA-TM-110218; NAS 1.15:110218; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Results of a numerical simulation are presented to determine the feasibility of estimating the location and strength of a wake vortex from imperfect in-situ measurements. These estimates could be used to provide information to a pilot on

how to avoid a hazardous wake vortex encounter. An iterative algorithm based on the method of secants was used to solve the four simultaneous equations describing the two-dimensional flow field around a pair of parallel counter-rotating vortices of equal and constant strength. The flow field information used by the algorithm could be derived from measurements from flow angle sensors mounted on the wing-tip of the detecting aircraft and an inertial navigation system. The study determined the propagated errors in the estimated location and strength of the vortex which resulted from random errors added to theoretically perfect measurements. The results are summarized in a series of charts and a table which make it possible to estimate these propagated errors for many practical situations. The situations include several generator-detector airplane combinations, different distances between the vortex and the detector airplane, as well as different levels of total measurement error.
Author
Aircraft Wakes; Vortex Avoidance; Algorithms; Two Dimensional Flow; Simultaneous Equations; Flow Distribution; Vortices

19960025262; 96N27335 NASA Ames Research Center, Moffett Field,CA USA
Analytical Fuselage and Wing Weight Estimation of Transport Aircraft
Chambers, Mark C., Santa Clara Univ., USA; Ardema, Mark D., Santa Clara Univ., USA; Patron, Anthony P., Santa Clara Univ., USA; Hahn, Andrew S., NASA Ames Research Center, USA; Miura, Hirokazu, NASA Ames Research Center, USA; Moore, Mark D., NASA Ames Research Center, USA; May 1996; 60p; In English
Contract(s)/Grant(s): RTOP 538-08-11
Report No.(s): NASA-TM-110392; A-961451; NAS 1.15:110392; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

A method of estimating the load-bearing fuselage weight and wing weight of transport aircraft based on fundamental structural principles has been developed. This method of weight estimation represents a compromise between the rapid assessment of component weight using empirical methods based on actual weights of existing aircraft, and detailed, but time-consuming, analysis using the finite element method. The method was applied to eight existing subsonic transports for validation and correlation. Integration of the resulting computer program, PDCYL, has been made into the weights-calculating module of the AirCraft SYNThesis (ACSYNT) computer program. ACSYNT has traditionally used only empirical weight estimation methods; PDCYL adds to ACSYNT a rapid, accurate means of assessing the fuselage and wing weights of unconventional aircraft. PDCYL also allows flexibility in the choice of structural concept, as well as a direct means of determining the impact of advanced materials on structural weight. Using statistical analysis techniques, relations between the load-bearing fuselage and wing weights calculated by PDCYL and corresponding actual weights were determined.
Author
Structural Weight; Fuselages; Bearings; Weight Analysis; Computer Programs; Airborne/Spaceborne Computers; Wings

19960026124; 96N27852 NASA Lewis Research Center, Cleveland, OH USA
1995 Workshop on Trajectory Optimization Methods and Applications
Alexander, Steve, NASA Lewis Research Center, USA; Langan, Kevin, Air Force Wright Aeronautical Labs., USA; Apr. 1996; 216p; In English; Atmospheric Flight Mechanics Conference, 7-10 Aug. 1995, Baltimore, MD, USA; Sponsored by American Inst. of Aeronautics and Astronautics, USA
Contract(s)/Grant(s): RTOP 564-09-20
Report No.(s): NASA-CP-10187; NAS 1.55:10187; E-10224; No Copyright; Avail: CASI; A10, Hardcopy; A03, Microfiche

This workshop presented recent advancements in trajectory optimization programming and applications. The developers of trajectory optimization programs were invited to present a 15-20 minute summary of recent improvements or current work. Users of trajectory optimization programs, particularly those supporting flight programs, utilizing new approaches or solving unique problems, were also encouraged to present a short summary of their applications and/or modifications. This workshop provided a unique forum to exchange ideas and information between industry, academia, and government agencies involved in trajectory optimization.
Author
Conferences; Trajectory Optimization; Trajectory Analysis

19960026141; 96N27869 Advisory Group for Aerospace Research and Development, Flight Vehicle Integration Panel., Neuilly-Sur-Seine, France
Flight Vehicle Integration Panel Working Group 21 on Glass Cockpit Operational Effectiveness *L'Efficacite operationnelle du poste de pilotage en verre*
Apr. 1996; 192p; In English; Original contains color illustrations
Report No.(s): AGARD-AR-349; ISBN 92-836-1034-2; Copyright Waived; Avail: CASI; A09, Hardcopy; A02, Microfiche

The members of AGARD FMP Working Group 21, representing aircraft manufacturers, research institutions and operational organizations, met over a two year period, 1993-1995 to address the following objective: Summarize the status of current cockpits, highlight their benefits and weaknesses, and provide guidance for future cockpit design. This report, the result of those meetings, is structured around the series of questions listed below: (1) What are the pilot and

crew required to do to complete a mission successfully; (2a) What do the current glass cockpits consist of; (2b) What are some of the technological highlights and trends of these cockpits; (3) What new technologies are becoming available; (4) How can we tailor the cockpit to be the most suitable for the human operator; (5) How can and how does the use of glass cockpits change the required aircrew training process; (6) What are the key problem issues with the current design process and what suggestions can be made to improve it and; (7) What are the cockpit concepts being considered to improve the operational effectiveness of future aircraft. With consideration of the human factors issues in design as a major basis for this report, this document provides an in depth discussion of the cockpit of today's aircraft and can serve as a foundation upon which to develop a more optimized pilot-vehicle-system interface of tomorrow.
Author
Fighter Aircraft; Helicopters; Cockpits; Human Factors Engineering; Man Machine Systems

19960026578; 96N28148 Galaxy Scientific Corp., Pleasantville, NJ USA
An Analysis of Ground-Flight Loads Measured on the Instrumented B-727 N40 ***Final Report***
Cavage, William M., Galaxy Scientific Corp., USA; DeFiore, Tom, Galaxy Scientific Corp., USA; Barnes, Terence, Galaxy Scientific Corp., USA; Oct. 1995; 120p; In English
Contract(s)/Grant(s): DTFA03-89-C-00043
Report No.(s): AD-A302165; DOT/FAA/AR-95/82; No Copyright; Avail: CASI; A06, Hardcopy; A02, Microfiche

The Federal Aviation Administration Technical Center owns and operates an instrumented B-727 aircraft for in-house test and assessment of runway friction and to examine landing gear impact on runway surfaces. Numerous strain gages were installed on the landing gear. Prior analysis of the data was concerned primarily with supporting the FAA's airports research program. It was recognized that this data set also has application in correlating the aircraft internal gear loads to measured ground-flight loads for specific takeoff and landing conditions and runway operations. This in-house research affords the National Aging Aircraft Research Program an opportunity to assess new data, independently obtained, which is both of high quality and having unusually high sampling rates to examine the landing gear loads for a limited number of takeoff, landing, and ground maneuvers. Sixty sample-per-second time history traces were available for each of 72 events with the following breakdown: (1) takeoff analysis, 22 events; (2) landing analysis, 18 events; (3) runway exit analysis, 9 events; (4) braking analysis, 6 events; (5) s-turn analysis, 9 events; and (6) minimum-radius turn analysis, 8 events.
DTIC
Landing Gear; Aircraft Landing; Braking; Takeoff; Aircraft Maneuvers; Friction; Flight Tests; Load Tests

19960027036; 96N28535 Lockheed Martin Engineering and Sciences Co., Hampton, VA USA
Flight test maneuvers for closed loop lateral-directional modeling of the F-18 High Alpha Research Vehicle (HARV) using forebody strakes
Morelli, E. A., Lockheed Martin Engineering and Sciences Co., USA; May 1996; 28p; In English
Contract(s)/Grant(s): NAS1-19000; RTOP 505-64-52-01
Report No.(s): NASA-CR-198270; NAS 1.26:198270; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Flight test maneuvers are specified for the F-18 High Alpha Research Vehicle (HARV). The maneuvers were designed for closed loop parameter identification purposes, specifically for lateral linear model parameter estimation at 30, 45, and 60 degrees angle of attack, using the Actuated Nose Strakes for Enhanced Rolling (ANSER) control law in Strake (S) model and Strake/Thrust Vectoring (STV) mode. Each maneuver is to be realized by applying square wave inputs to specific pilot station controls using the On-Board Excitation System (OBES). Maneuver descriptions and complete specification of the time/amplitude points defining each input are included, along with plots of the input time histories.
Author
Angle of Attack; Research Vehicles; F-18 Aircraft; Aircraft Maneuvers; Thrust Vector Control; Feedback Control; Trajectory Planning; Parameter Identification

19960027241; 96N28682 Naval Postgraduate School, Monterey, CA USA
An analysis of single-engine rate-of-climb capabilities and thrust requirements of the S-3 and ES-3 aircraft in support of the TF34 Engine Component Improvement Program
Micklewright, Alan J., Naval Postgraduate School, USA; Jun. 01, 1995; 117p; In English
Report No.(s): AD-A302899; No Copyright; Avail: Issuing Activity (Defense Technical Information Center (DTIC)), Microfiche

This thesis provides an analytical look at the performance of the TF34 engine installed on the Navy's 5-3 and ES-3 aircraft. The objective of the thesis is to provide information to assist in the effective management of proposals and improvements being considered under the TF34 Engine Component Improvement Program (CIP). Historical flight data, simulator flight and thrust data, historical operational engine data, and data from aircrew surveys were all analyzed to determine the significance of TF34 engine failures in critical flight situations and the degree of engine performance enhancement available. Based on the research, it was determined that a valid thrust deficiency exists with regard to single-engine rate-of-climb performance of the ES-3A aircraft. Suggestions to help solve this deficiency are presented. The most promising recommendation for increasing performance with a minimal ini-

tial cost outlay is to increase the engine interturbine temperature (ITT) operating limit.
DTIC
S-3 Aircraft; Engine Failure; Flight Simulators; Surveys; Flight Crews; Thrust

19960027320; 96N28754 Naval Surface Warfare Center, Dahlgren Div., Dahlgren, VA USA
Extension of the NSWCDD aeroprediction code to the roll position of 45 degrees *Final Report*
Moore, Frank, Naval Surface Warfare Center, USA; McInville, Roy, Naval Surface Warfare Center, USA; Dec. 01, 1995; 156p; In English
Report No.(s): AD-A302427; NSWCDD-TR-95-160; No Copyright; Avail: CASI; A08, Hardcopy; A02, Microfiche

The NSWCDD Aeroprediction Code has been extended to the roll position of 45 degrees (or fins in cross or 'x' orientation). New technology developed includes wing body and body wing interference factors for this roll orientation and to angles of attack of 90 degrees; new semi-empirical wing tail interference models for both the fins in 0 and 45 degrees roll positions; and approximate methods to estimate wing alone center of pressure shift at high angles of attack and 45 degrees roll position as well as 'fin choking' at high mach number. This new technology allows aerodynamics to be estimated with average accuracy levels of +/- 10 percent on normal and axial force coefficients and +/- 4 percent of body length on center of pressure. An exception to this is at subsonic mach number and high angle of attack where wind tunnel sting interference effects are present and at high Mach number and angle of attack where internal shock interactions from a forward fin onto an aft mounted fin are present. Results are presented for several wing body and wing body tail configurations at various Mach numbers and angles of attack to support this average accuracy level conclusion.
DTIC
Body-Wing and Tail Configurations; Roll; Angle of Attack; Shock Wave Interaction; Interactional Aerodynamics; Aerodynamic Interference

19960027357; 96N28786 Air Force Inst. of Tech., Dept. of Engineering., Wright-Patterson AFB, OH USA
Comparison of two shock-capturing methods for calculation of transonic airfoil flutter
Buxton, Bradley J., Air Force Inst. of Tech., USA; Dec. 01, 1995; 108p; In English
Report No.(s): AD-A302254; AFIT/GAE/ENY/95D-04; No Copyright; Avail: Issuing Activity (Defense Technical Information Center (DTIC)), Microfiche

The flutter of a pitch-and-plunge airfoil (PAPA) in transonic flow is simulated using two shock-capturing methods. Each method solves a coupled system of equations involving the unsteady Euler equations and a two-degree-of-freedom structural model. The first method, TVDntiAE, uses an explicit Total-Variation-Diminishing (TVD) algorithm to compute the flowfield for a 2-D airfoil. The second method, ENS3DAE, uses an implicit Beam-Warming approach to compute the flowfleld on a 3-D rectangular wing while enforcing 2-D flow. Results of the schemes are compared with each other and against limited published data to extend the validation of the software to unsteady fluid/structure interactions. Results are compared for three airfoil configurations: (1) static; (2) forced oscillation; and (3) PAPA. Computation of flutter points is of special interest. ENS3DAE predicts a flutter onset speed for the NACA 64A-006 airfoil at M=0.87 that is 1% greater than the value predicted by TVDntiAE and 2% smaller than the published data. Variations in model predictions most likely due to dissipation-model differences and grid and time-step sensitivities are evaluated.
DTIC
Algorithms; Transonic Flow; Airfoils; Degrees of Freedom; Euler Equations of Motion; Flow Distribution; Mathematical Models

19960027445; 96N28834 Air Force Inst. of Tech., Wright-Patterson AFB, OH USA
Comparison of The Ralph Smith and the Time Domain Flying Qualities Criteria
Carlucci, Lori Ann, Air Force Inst. of Tech., USA; Dec. 1995; 177p; In English
Report No.(s): AD-A301706; AFIT/GAE/ENY/95D-05; No Copyright; Avail: CASI; A09, Hardcopy; A02, Microfiche

Aircraft pitch response is a crucial element of piloted vehicle flying qualities.The short term pitch response has created controversy over the form and substance of any requirements. Currently there are six different methods for evaluation in MIL-STD-1797A. There are many other methods which have been proposed. The biggest problem is that many of these methods often give conflicting results. The overall goal of the present effort is to compare and contrast the Time Domain criterion and the Ralph Smith criterion. by examining these methods on common grounds, areas of agreement and discrepancies can be found. Parametric studies are performed and trends identified.
DTIC
Aircraft Performance; Flight Characteristics

19960027920; 96N29061 Air Force Inst. of Tech., Wright-Patterson AFB, OH USA
Reliability Centered Prediction Technique for Diagnostic Modeling and Improvement
Murphy, Michael D., Air Force Inst. of Tech., USA; Dec. 27, 1995; 100p; In English
Report No.(s): AD-A302791; AFIT-95-138; No Copyright; Avail: CASI; A05, Hardcopy; A02, Microfiche

Design phase decisions based on diagnosability lead to lower system costs and, in turn, higher quality products by means of reducing maintenance time and increasing system

reliability. A case for diagnosability is presented. Functions of diagnosability are expounded upon including life cycle costs, statistical analysis, and design criterion to emphasize the necessity of diagnosability analysis early in the design phase. A diagnosability prediction metric is developed for system modeling of component failure rates and unjustified removals.The metric emphasizes ambiguity of system component indications as well as system structure. The metric is evaluated using historical data from the bleed air control system (BACS) on the Boeing 737-300. Seven design changes are suggested based on improving system diagnosability by changing component functions, modifying indications, and adding or changing sensors. The resulting designs are compared via Boeing's life cycle cost mechanism, DEPCOST model, based on cost improvements. It is shown that system improvements based on this prediction technique will increase the quality of a product since increased diagnosability decreases life cycle costs.
DTIC
Mathematical Models; Systems Analysis; Cost Analysis; Reliability; Maintainability; Performance Prediction; Life Cycle Costs

19960027954; 96N29094 Arizona State Univ., Tempe, AZ USA
Genetic algorithms in conceptual design of a light-weight, low-noise, tilt-rotor aircraft *Final Report*
Wells, Valana L., Arizona State Univ., USA; Jun. 1996; 8p; In English
Contract(s)/Grant(s): NAG2-882
Report No.(s): NASA-CR-201404; NAS 1.26:201404; No Copyright; Avail: CASI; A02, Hardcopy; A01, Microfiche

This report outlines research accomplishments in the area of using genetic algorithms (GA) for the design and optimization of rotorcraft. It discusses the genetic algorithm as a search and optimization tool, outlines a procedure for using the GA in the conceptual design of helicopters, and applies the GA method to the acoustic design of rotors.
Author
Genetic Algorithms; Aircraft Design; Tilt Rotor Aircraft; Acoustic Properties; Computer Aided Design; Tilting Rotors; Helicopter Tail Rotors; Blade Slap Noise

19960029238; 96N29735 Air Force Inst. of Tech., School of Engineering., Wright-Patterson AFB, OH USA
Wing Flexure Compensation for Aircraft Attitude and Position Determination in an Inverted Carrier-Phase Positioning System
Mahlum, Bradley W., Air Force Inst. of Tech., USA; Dec. 1995; 104p; In English
Report No.(s): AD-A303320; AFIT/GE/ENG/95D-14; No Copyright; Avail: CASI; A06, Hardcopy; A02, Microfiche

In response to the ever increasing accuracies in inertial navigation systems, the U. S. Air Force must develop higher accuracy reference systems. These reference systems must also be small enough to be utilized in the testing of navigation systems onboard fighter aircraft. One such proposed system utilizes carrier-phase Global Positioning System (CPGPS) transmitters mounted on MM-9 pods with receivers on the ground. This research examines one possible method of utilizing this system to determine the attitude and position of the aircraft, given position estimates for the transmitter's locations. The transmitter positioning algorithm showed that the geometry will be problematic for this configuration. However, if given the estimates of transmitter positions within the desired accuracy, the aircraft attitude and position algorithms worked effectively.
DTIC
Inertial Navigation; Global Positioning System; Positioning; Fighter Aircraft; Reference Systems; Wings

19960031942; 96N29932 NASA Langley Research Center, Hampton, VA USA
An aeroelastic analysis of helicopter rotor blades incorporating piezoelectric fiber composite twist actuation
Wilkie, W. Keats, NASA Langley Research Center, USA; Park, K. C., Colorado Univ., USA; May 1996; 70p; In English
Contract(s)/Grant(s): RTOP 505-63-36-02
Report No.(s): NASA-TM-110252; NAS 1.15:110252; ARL-MR-328; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

A simple aeroelastic analysis of a helicopter rotor blade incorporating embedded piezoelectric fiber composite, interdigitated electrode blade twist actuators is described. The analysis consist of a linear torsion and flapwise bending model coupled with a nonlinear ONERA based unsteady aerodynamics model. A modified Galerkin procedure is performed upon the rotor blade partial differential equations of motion to develop a system of ordinary differential equations suitable for numerical integration. The twist actuation responses for three conceptual full-scale blade designs with realistic constraints on blade mass are numerically evaluated using the analysis. Numerical results indicate that useful amplitudes of nonresonant elastic twist, on the order of one to two degrees, are achievable under one-g hovering flight conditions for interdigitated electrode poling configurations. Twist actuation for the interdigitated electrode blades is also compared with the twist actuation of a conventionally poled piezoelectric fiber composite blade. Elastic twist produced using the interdigitated electrode actuators was found to be four to five times larger than that obtained with the conventionally poled actuators.
Author
Aeroelasticity; Helicopters; Fiber Composites; Rotary

Wings; Unsteady Aerodynamics; Differential Equations; Galerkin Method; Numerical Integration

19960033194; 96N30013 NYMA, Inc., Brook Park, OH USA
High Altitude Towed Glider ***Final Report***
Colozza, Anthony J., NYMA, Inc., USA; Jun. 1996; 16p; In English
Contract(s)/Grant(s): NAS3-27186; RTOP 537-10-20
Report No.(s): NASA-CR-198493; E-10296; NAS 1.26:198493; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The concept of using an unmanned towed glider for high altitude scientific research had been previously proposed. This paper examines the feasibility of this concept by determining what impact the various characteristics of the tow line, glider and tow aircraft have on tow line drag. A description of the analysis and computer code used to generate the results is given. The parameters examined were glider altitude, tow aircraft glider separation distance, velocity, tow line drag coefficient and tow line material properties. The results from the analysis show that the tow line drag increases significantly with tow aircraft/glider separation. The drag increased from 940 N (211 lb) with a tow aircraft/glider separation of 3 km to 11,970 N (2691 lb) with a tow aircraft/glider separation of 10 km. The results also show that by varying some of the initial assumptions significant reductions in tow line drag and weight can be obtained. The variables which had the greatest effect on reducing the tow line drag were the decrease in tow aircraft/glider separation distance, the increase in tow line strength and the decrease in glider Mach number.
Author
Gliders; Aerodynamic Drag; High Altitude; Drag Reduction; Remotely Piloted Vehicles; Towed Bodies

19960033210; 96N30024 Army Research Lab., Aberdeen Proving Ground, MD USA
Dynamic analysis of helicopter bearingless main rotor with hub flexbeam damage configurations ***Final Report, May 1993 - Sep. 1994***
Kim, Ki C., Army Research Lab., USA; 1996; 43p; In English
Contract(s)/Grant(s): ARL Proj. 1L162618AH8
Report No.(s): AD-A303362; ARL-TR-92; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This report documents a dynamic analysis of a helicopter bearingless main rotor system with damaged hub flexbeam configurations. The analysis was performed using a comprehensive helicopter aeroelastic code based on finite element/blade element theory. The bearingless main rotor (BMR) system, including flexbeams, torque tubes, and main rotor blades, is modeled as a number of elastic beam finite elements, wherein each beam element undergoes flap bending, lag bending, elastic twist, and axial deflections. Aerodynamic forces on rotor blades are calculated using quasisteady aerodynamic theory with a linear in-flow model. Flexbeam ballistic damage is simulated by changes in the span-wise distribution of the mass, bending and torsional stiffness of flexbeam element. Results are first calculated for a soft in-plane, five-bladed, bearingless rotor, with an undamaged (baseline) configuration. Results are then calculated for this rotor system with damage representation. The effects of this damage on rotor and helicopter system performances are determined in terms of blade modal shapes and frequencies, rotor system aeroelastic response and loads variations. Ballistic damage to the hub flexbeam can significantly affect the dynamic behavior of the bearingless rotor system.
DTIC
Aerodynamic Forces; Bearingless Rotors; Dynamic Characteristics; Finite Element Method; Mass Distribution; Rotary Wings

19960033216; 96N30030 Air Force Inst. of Tech., School of Engineering., Wright-Patterson AFB, OH USA
Vehicle Thermal Control with a Variable Area Inlet
Layne, Thomas R., Air Force Inst. of Tech., USA; Dec. 1995; 110p; In English
Report No.(s): AD-A303215; AFIT/GAE/ENY/95D-15; No Copyright; Avail: CASI; A06, Hardcopy; A02, Microfiche

This study developed a variable area inlet and controller that regulated the temperature of an electrical component with ram air. The intent of the variable area inlet was to reduce vehicle drag by eliminating inefficiencies associated with component cooling and fixed area inlets. These inefficiencies arise from vehicles moving at varying speeds through varying air temperatures. The hardware model consisted of an electrical component mounted inside a right-circular cylindrical duct. The variable area inlet, mounted in the front of the duct, consisted of a butterfly valve that was actuated by a stepper controller acted on the feedback signal of a thermocouple that was mounted on the electrical component. The system was successful in regulating the component temperature. A nonlinear simulation model was built and the thermal plant in the simulation was based on the electrical components empirically derived Nusselt number. Proportional, Proportional-Derivative (PD), and Proportional-Integral-Derivative (PID) controllers were built and tested. The PD and PID controllers did not appear to need any gain scheduling for the varying speed and temperature conditions. Lastly, a general design process was detailed.
DTIC
Temperature Control; Controllers; Air Intakes; Air Ducts; Control Systems Design; Conductive Heat Transfer

19960033227; 96N30041 NAVMAR, Inc., Warminister, PA USA
Metallurgical examination of a damaged V-22 hydraulic Line ***Final Report***
Agarwala, V. V., NAVMAR, Inc., USA; Pregger, B. A., NAV-

MAR, Inc., USA; Frazier, W. E., Naval Air Warfare Center, USA; Sep. 01, 1995; 23p; In English
Report No.(s): AD-A303293; NAWCADWAR-95045-4.3; No Copyright; Avail: Issuing Activity (Defense Technical Information Center (DTIC)), Microfiche

A preliminary metallurgical investigation of the damaged, Ti-3A1-2.5V, V-22 System 1 Pump Suction Tube of the V-22 Osprey aircraft, identified as 901-081-904-101, has been completed. The 1.0 inch (2.54 cm) diameter tube sustained localized damage where it was secured by a rubber clamp to the transmission. That portion of the tube was excised, sectioned, and examined using metallography and a Scanning Electron Microscope (SEM). Qualitative chemical analysis utilizing Energy Dispersive Spectroscopy (EDS) was performed on the clamp and alloy. Results obtained indicate that the damage was due to fretting. No conclusive evidence was found for corrosion, such as the presence of by-products or an alteration in microstructure.
DTIC
V-22 Aircraft; Metallography; Fretting; Damage Assessment

19960033256; 96N30060 Advisory Group for Aerospace Research and Development, Neuilly-Sur-Seine, France
Integrated vehicle management systems
Apr. 1996; 144p; In English
Report No.(s): AGARD-AR-343; ISBN 92-836-1035-0; Copyright Waived; Avail: CASI; A07, Hardcopy; A02, Microfiche

Major trends in technology, weapon system performance goals and affordability for aerospace systems are occurring simultaneously. For avionic systems this performance and affordability can be achieved by functional and physical integration. 'Functionally' integrated subsystems to achieve higher performance has been greatly aided by advances in computer technology. The desire to minimize costs for these systems has been accomplished through a 'physical' integration concept based upon common modules tied through a high speed backplane. The concept, called integrated avionics, has been used on new aircraft such as the US Air Force F-22 fighter and the Boeing 777 commercial transport. Vehicle management systems provide the management of crucial flight functions and systems for advanced aerospace vehicles. These systems must have high integrity, safety, and overall fault tolerance. Low cost modular avionics are unproven for such fault tolerant systems. This becomes a key issue for investigation. This report deals with the key problems in fault tolerance for modular computer based systems. New techniques, only recently applied, provide exciting possibilities to reduce avionics costs and maintain high integrity and safety. These techniques and more are discussed in this report sponsored by the Mission Systems Panel of the AGARD.
Author
Fault Tolerance; Functional Integration; Avionics; Cost Reduction; Modularity; Aircraft Design; Flight Control; Computer Aided Design; Systems Health Monitoring

19960033297; 96N30095 NASA Ames Research Center, Moffett Field, CA USA
Rotorcraft Blade-Vortex Interaction Controller
Schmitz, Fredric H., Inventor, NASA Ames Research Center, USA; Dec. 26, 1995; 23p; In English; Division of US-Patent-Appl-SN-972790, filed 6 Nov. 1992 (US-Patent-5,437,419)
Patent Info.: NASA-Case-ARC-11992-2; US-Patent-5,478,029; US-Patent-Appl-SN-439838; US-Patent-Appl-SN-972790; No Copyright; Avail: US Patent and Trademark Office, Hardcopy, Microfiche

Blade-vortex interaction noises, sometimes referred to as 'blade slap', are avoided by increasing the absolute value of inflow to the rotor system of a rotorcraft. This is accomplished by creating a drag force which causes the angle of the tip-path plane of the rotor system to become more negative or more positive.
Official Gazzette of the U.S. Patent and Trademark
Blade-Vortex Interaction; Rotary Wing Aircraft; Controllers; Aerodynamic Drag

06
AIRCRAFT INSTRUMENTATION

Includes cockpit and cabin display devices; and flight instruments.

19960025636; 96N27640 NASA Langley Research Center, Hampton,VA USA
Airline Transport Pilot Preferences for Predictive Information
Trujillo, Anna C., NASA Langley Research Center, USA; Feb. 1996; 52p; In English
Contract(s)/Grant(s): RTOP 505-64-13-22
Report No.(s): NASA-TM-4702; L-17454; NAS 1.15:4702; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

This experiment assessed certain issues about the usefulness of predictive information: (1) the relative time criticality of failures, (2) the subjective utility of predictive information for different parameters or sensors, and (3) the preferred form and prediction time for displaying predictive information. to address these issues, three separate tasks were administered to 22 airline pilots. As shown by the data, these pilots preferred predictive information on parameters they considered vital to the safety of the flight. These parameters were related to the checklists performed first for alert messages. These pilots also preferred to know whether a parameter was changing abnormally and the time to a certain value being reached. Furthermore, they considered this information most useful during the cruise, the climb, and the descent phases of flight. Lastly,

these pilots preferred the information to predict as far ahead as possible.
Author
Predictions; Information Management; Pilots (Personnel); Crew Procedures (Inflight)

19960025770; 96N27687 Civil Aeromedical Inst., Oklahoma City,OK USA
A Comparison of the Effects of Navigational Display Formats and Memory Aids on Pilot Performance *Final Report*
Beringer, Dennis B., Civil Aeromedical Inst., USA; Harris, Howard C., Jr., Civil Aeromedical Inst., USA; May 1996; 14p; In English
Report No.(s): DOT/FAA/AM-96/16; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A great deal of effort has been invested in examining integrated instrumentation for advanced cockpits, but little comparable effort has been directed toward the greatest number of aircraft presently flying - those in the general aviation environment. This series of studies examined the benefits of a simple and widely available integrated instrument, the horizontal situation indicator (HSI), in the performance of simple navigational and orientational tasks by private pilots and instructor pilots. Tested in the context of the multiple-processor Basic General Aviation Research Simulator (BGARS), the private pilots exhibited significantly fewer navigational reversals and orientational errors when using the HSI (in comparison with their performances when using the traditional VOR and directional gyro combination). These results were consistent with but even more definitive than those obtained for the instructor pilots. Similar benefits in procedural error reduction were also found when instrument index markers, or 'bugs,' were used as short-term memory aids.
Author
Pilot Performance; Display Devices; Navigation Aids; Air Navigation; Flight Simulation

19960026413; 96N28063 Federal Aviation Administration, Technical Center., Atlantic City, NJ USA
Los Angeles International Airport Instrument Landing System Approach, phase 2 and 3 *Final Report, Jan. - Jul. 1992*
DiMeo, K., Federal Aviation Administration, USA; Melville, B., Federal Aviation Administration, USA; May 1995; 91p; In English
Report No.(s): AD-A302727; DOT/FAA/CT-93/61; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

This report describes the methodology and results of Phase 2 and 3 of a four-phase program. The objective of the four-phase program is to measure and model navigational performance of aircraft making Instrument Landing System (ILS) approaches at distances from 10 to 32 nautical miles from the runway threshold. Navigation of the localizer at these distances is envisioned for simultaneous instrument approaches into parallel triple and quadruple runways. Knowledge of navigational performance will facilitate evaluation of triple and quadruple approach concepts and may identify ways to reduce Total Navigational System Error, which, in turn, may make triple and quadruple approach concepts more viable. The objective of Phase 2 was to quantify the bias in airport surveillance radar (ASR)-9 target position reports, and to provide correction factors to account for this bias in the Phase 1 calculation of cross-track deviation (CTD). The objective of Phase 3 was to determine the location of the ILS localizer course centerline (CL), half-scale, and full-scale course deviation indicator signals relative to the extended runway centerline, and to provide data on localizer CL location for use in Phase 1 CTD calculations.
DTIC
Instrument Landing Systems; Instrument Approach; Aircraft Instruments

19960026761; 96N28321 Army Aviation Systems Command, Aeroflightdynamics Directorate., Moffett Field, CA USA
Longbow Apache warning icons study. Phase 2: LBA warning icon design logic and implementation. Phase 3: State-of-the-art design requirements for aircraft auditory and visual warning icons
Gardner, Donald R., DGA, Inc., USA; Williams, Douglas H., Psycho-Linguistic Research Associates, USA; Simpson, Carol A., Psycho-Linguistic Research Associates, USA; 1996; 140p; In English
Contract(s)/Grant(s): NAS2-13188
Report No.(s): NASA-CR-4743; NAS 1.26:4743; USAATCOM-TR--96-A-002; A-961729; No Copyright; Avail: CASI; A07, Hardcopy; A02, Microfiche

U.S. Army Aeroflightdynamics Directorate (AFDD) conducted a human factors study of the Warning Icons for the Longbow Apache (WILBA) for the Program Manager-Longbow Apache. Phase 1, entitled; Longbow Apache Warning Icons Study, Phase 1: Detailed Problem Statements, was completed in 1994 (USAATCOM TR-96-A-003). This report covers Phase 2, a description of the current LBA warning icons, and Phase 3, state-of-the-art design requirements for aircraft alerting systems. The current LBA system was designed contractually in accordance with 1970 standards and is a collection of federated components. The visual alerts are relatively well integrated. However, for audible alerts there is no integration of tactical alerts with the aircraft systems alerts nor with the flight parameter alerts. This results in asynchronous delivery of audible alerts with the potential for simultaneous audible alerts and auditory overload for the crew. Also, the Longbow's digitized compressed voice technology does not meet the requirements for cockpit alerting systems. This study recommends currently available technology for unlimited speech message generation, based on an operational inte-

grated warning caution and advisory (IWCA) system in AFDD's Crew Station R&D Facility. The IWCA would address the current problems with the LBA alerting system and would provide for cost-effective growth for Longbow and other Army aircraft.
Author
Human Factors Engineering; Cockpits; Synchronism; Crew Workstations

19960029338; 96N29792 Organisatie voor Toegepast Natuurwetenschappelijk Onderzoek, TNO Human Factors Research Inst., Soesterberg, Netherlands
Tactical situation displays: integrated versus separate information presentation ***Interim Report***
Passenier, P. O., Organisatie voor Toegepast Natuurwetenschappelijk Onderzoek, Netherlands; Van Delft, J. H., Organisatie voor Toegepast Natuurwetenschappelijk Onderzoek, Netherlands; Nov. 30, 1995; 19p; In English
Contract(s)/Grant(s): A90/KLu/320
Report No.(s): TNO-TM 1995 A-73; TD 95-1491; Copyright; Avail: Issuing Activity (TNO Human Factors Research Institute, Kampweg 5, 3769 DE Soeterberg, Netherlands), Hardcopy, Microfiche

In the scope of a study on tactical situation displays in the cockpit of modern fighter aircraft, a simulator experiment has been conducted on integrated ('highway in the sky') versus separate presentation of flight information. Results of the experiment show that the 'highway in the sky' especially for 'slow' tasks (slow in relation to aircraft dynamics) leads to more accurate flight performance because of the integrated feedback of he aircraft state, when compared to separate information presentation. However, for 'rapid tasks', where the emphasis is on feedforward control, the separate information presentation is to be preferred because of the more constant preview as a result of the decoupling of the information presentation along different dimensions.
Author
Display Devices; Feedforward Control; Human Factors Engineering; Multisensor Applications; Information Adaptive System; Flight Control; Pilot Performance; Cockpit Simulators; Knowledge Based Systems

19960031892; 96N29888 Federal Aviation Administration, Technical Center., Atlantic City, NJ USA
Mark 20 category 2/3 instrument landing system operational test and evaluation, functional and performance test report ***Progress Report, Jul. 1994 - Mar. 1995***
Jones, Jesse, Federal Aviation Administration, USA; Bennett, Benjamin, Raytheon Service Co., USA; Robbins, Byron, Science Applications International Corp., USA; Oct. 1995; 76p; In English
Report No.(s): AD-A303409; DOT/FAA/CT-TN95/44; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

This test report documents the results of the Operational Test and Evaluation (OT&E) Functional and Performance tests conducted on the Category 2/3 Mark 20 Instrument Landing System (ILS) at the FAA Technical Center, Atlantic City International Airport, Atlantic City, NJ. The Mark 20 ILS modular design is based on a new generation of microprocessors and software. With Remote Maintenance Monitoring (RMM) capability and the concept of 'remove and replace' maintenance, it will provide major airway facilities throughout the U.S. with a more reliable and easily maintained system. The report contains the system configuration, test descriptions, test equipment used, data collection and analysis methods, test results, and conclusions. Based on testing performed at the FAA Technical Center, it is recommended that the Mark 20 ILS be accepted for deployment.
DTIC
Instrument Landing Systems; Airports; Systems Engineering; Applications Programs (Computers)

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.

19960024781; 96N26942 Cincinnati Univ., OH USA
Preliminary C++ Class Definitions for Object-Oriented Engine Simulation ***Final Report, Oct. 1993 - Jan. 1995***
Walker, Bruce K., Cincinnati Univ., USA; May 1996; 8p; In English
Report No.(s): NASA-CR-200981; NAS 1.26:200981; No Copyright; Avail: CASI; A02, Hardcopy; A01, Microfiche

The objective of this work was to define and code some preliminary object classes in C++ for eventual use in the National Propulsion System Simulator (NPSS) program at NASA Lewis Research Center (LeRC). The class definition phase of the project involved collaboration with LeRC researchers and with researchers from General Electric Aircraft Engines (GEAE) on the class hierarchy and the selection and definition of the classes that were needed. The coding phase involved the writing, compilation, and testing of C+ + code to implement some of the preliminary classes, with interface requirements specified primarily by GEAE. The original goal was to define and code classes for the implementation of simple functional tasks like table lookups for gas properties and interfacing with a variable table class that was under construction at LeRC and to implement some simple components such as ducts and bends. This was to lead to a follow-on project that involved implementing in the new modular C+ + structure a compressor rig simulation, where the compressor simulation to be mimicked was to be ether one that had previously been implemented in FORTRAN by GEAE or the NEPP compressor that had been examined by LeRC. The motivation there was the a compressor rig simulation involves

essentially all of the complexity that would be required for simulating any other engine component and therefore that other components could mimic the structure for a compressor rig simulation.
Derived from text
Aircraft Engines; Air Breathing Engines; C++ (Programming Language); Data Structures; Object-Oriented Programming

19960024804; 96N26963 Advisory Group for Aerospace Research and Development, Propulsion and Energetics Panel., Neuilly-Sur-Seine, France
Loss Mechanisms and Unsteady Flows in Turbomachines ***Les mecanismes des pertes et les ecoulements instationnaires dans les turbomachines***
Jan. 1996; 578p; In English; In French; 85; Propulsion and Energetics Panel Symposium, 8-12 May 1995, Derby, UK; Also announced as 19960024804 through 19960024846; Original contains color illustrations
Report No.(s): AGARD-CP-571; ISBN-92-836-0020-7; Copyright Waived; Avail: CASI; A25, Hardcopy; A06, Microfiche

The Conference Proceedings contains 42 papers presented at the Propulsion and Energetics Panel 85th Symposium on Loss Mechanisms and Unsteady Flows in Turbomachines which was held from 8-12 May 1995, in Derby, UK. Sessions: Cascade Steady Flow Mechanisms and Losses (9 papers); Steady Flow Clearance and Injection Flow Mechanisms and Losses (6); Steady Secondary Flows and Shock Boundary Layer Interaction (3); Blade Section-Wake Boundary layer Interaction (6); Management of Unsteady Flows (10); Blade Interactions (8). The Symposium contributes to improving the efficiency and operation of turbomachinery also in off-design conditions by application of experimental and computational fluid dynamics methods. Multi-stage methods and positive loss management in the design stage have been identified as next steps. This will yield optimized engines of high efficiency for expanded flight envelopes including active control of the plane movement at high reliability levels.
Author
Computational Fluid Dynamics; Boundary Layer Transition; Shock Layers; Engine Design; Aircraft Engines; Turbomachinery

19960024805; 96N26964 Technische Univ., Inst. of Fluid Mechanics., Brunswick, Germany
Endwall Boundary Layer Separations and Loss Mechanisms in Two Compressor Cascades of Different Stagger Angle
Stark, U., Technische Univ., Germany; Bross, S., KSBAG, Germany; Loss Mechanisms and Unsteady Flows in Turbomachines; Jan. 1996; 16p; In English; Also announced as 19960024804; Copyright Waived; Avail: CASI; A03, Hardcopy; A06, Microfiche

The paper describes experimental and theoretical investigations on the effect of stagger angle on the endwall flow in two compressor cascades, one with a low stagger angle (1 = 30 degrees) and one with a high stagger angle (1 = 50 degrees). The results of flow visualization tests, wake measurements and inviscid secondary flow calculations are shown for both cascades. The more important differences between the two sets of results are discussed with particular attention to 3D endwall boundary layer separations and loss mechanisms.
Author
Cascade Flow; Flow Visualization; Wall Flow; Boundary Layer Flow; Wakes

19960024806; 96N26965 National Technical Univ., Lab. of Thermal Turbomachines., Athens, Greece
Losses Prediction in Axial Flow Compressor Cascades Using an Explicit K-Epsilon Navier-Stokes Solver
Vassilopoulos, C., National Technical Univ., Greece; Simandirakis, G., National Technical Univ., Greece; Giannakoglou, K. C., National Technical Univ., Greece; Papailiou, K. D., National Technical Univ., Greece; Loss Mechanisms and Unsteady Flows in Turbomachines; Jan. 1996; 10p; In English; Also announced as 19960024804; Copyright Waived; Avail: CASI; A02, Hardcopy; A06, Microfiche

Flow patterns and losses prediction in a controlled diffusion airfoil cascade of an industrial axial compressor are demonstrated by means of a two-dimensional, explicit, time-marching fractional-step solver. The fractional-step algorithm presents certain advantages which are mainly related to its simplicity and the fact that greater time-steps are allowed, in comparison to other explicit solvers. In the present analysis, turbulence is modeled through both a low Reynolds k-E model and the algebraic Baldwin-Lomax one. The former is combined with an efficient artificial viscosity scheme, leading thus to a smooth E-field close to the solid walls. From a numerical point of view, the Baldwin-Lomax model cooperates perfectly with the basic numerical kernel. The k-E model, when used in conjunction to an explicit solver, is prone to instabilities and a particular semi-implicit treatment of the source terms is required. On the basis of the examined cases, differences in the predictive capabilities of the two turbulence models are presented and discussed.
Author
K-Epsilon Turbulence Model; Navier-Stokes Equation; Turbocompressors; Flow Distribution; Cascade Flow; Kernel Functions

19960024807; 96N26966 Manchester Coll. of Science and Technology, Dept. of Mechanical Engineering., UK
Computational Modelling of Cascade-Blade Flow with Linear and Non-Linear Low-Re Eddy-Viscosity Models
Chen, W. L., Manchester Coll. of Science and Technology,

UK; Lien, F. S., Manchester Coll. of Science and Technology, UK; Leschziner, M. A., Manchester Coll. of Science and Technology, UK; Loss Mechanisms and Unsteady Flows in Turbomachines; Jan. 1996; 14p; In English; Also announced as 19960024804; Copyright Waived; Avail: CASI; A03, Hardcopy; A06, Microfiche

A computational study is presented of the flow around a controlled-diffusion compressor-cascade blade at design as well as off-design conditions. In this flow, features associated with by-pass transition close to the leading edge - probably including laminar leading-edge separation contribute significantly to the evolution of the boundary layer on the blade surface. This has motivated the additional investigation of transition in a number of flows which are highly pertinent to the principal blade flow: two flat-plate boundary layers, a flow over a curved blade-like bump in a channel, and a flow past a rounded leading edge involving laminar separation and transitional reattachment. It is demonstrated that conventional eddy-viscosity models, based on the linear Bousinesq stress--strain relations, are able to capture by-pass transition in simple shear, but are unable to resolve transitional features in complex strain, especially those associated with leading edge separation, unless a mechanism is introduced to suppress excessive normal-strain-induced production of turbulence energy. The non-linear model is shown to return a considerably improved representation of the blade flow, resolving the leading-edge processes and returning a much improved variation of the loss as a function of the inlet angle of incidence.
Author

Cascade Flow; Compressor Blades; Leading Edges; Turbulence Models; Angles (Geometry)

19960024808; 96N26967 Technical Univ. of Lodz, Inst. of Turbomachinery., Lodz, Poland

Boundary Layer and Loss Studies on Highly Loaded Turbine Cascade

Smolny, A., Technical Univ. of Lodz, Poland; Blaszczak, J., Technical Univ. of Lodz, Poland; Loss Mechanisms and Unsteady Flows in Turbomachines; Jan. 1996; 14p; In English; Also announced as 19960024804; Copyright Waived; Avail: CASI; A03, Hardcopy; A06, Microfiche

The test results of a boundary layer developing on the suction side of the highly loaded turbine vane are presented. The tests of the boundary layer development were performed with hot film transducers deposited on a polyimide foil and glued to the profile wall. Special attention was paid to a formation of a separation region. Total pressure losses with regard to the boundary layer condition on the suction side of the profile were determined.
Author

Boundary Layer Flow; Cascade Flow; Aerodynamic Characteristics; Reynolds Number; Turbulent Flow; Vanes; Leading Edges

19960024809; 96N26968 Allison Engine Co., Indianapolis, IN USA

The Effect of Vane-Blade Spacing on Transonic Turbine Stage Performance

Kelecy, F. J., Allison Engine Co., USA; Griffin, J. W., Allison Engine Co., USA; Delaney, R. A., Allison Engine Co., USA; Loss Mechanisms and Unsteady Flows in Turbomachines; Jan. 1996; 14p; In English; Also announced as 19960024804 Contract(s)/Grant(s): NAG3-1645; Copyright Waived; Avail: CASI; A03, Hardcopy; A06, Microfiche

This paper presents the results of a computational study on the effect of axial spacing between the vane and blade rows of a transonic turbine stage. The study was performed on the mid-span section of a high-pressure turbine stage using a quasi-3D, unsteady Navier-Stokes solver that provides a fully interactive vane-blade unsteady flow solution. Three different cases were considered, corresponding to axial spacings of 20%, 40%, and 60% of the vane axial chord. The calculated vane and blade pressure distributions for the 40 percent case were found to compare favorably with experimental measurements acquired in a short-duration shock tunnel. In addition, the analysis shows a marked increase in the amplitude of the unsteady pressure fluctuations on the vane and blade surfaces as the spacing decreases. Time-averaged stage adiabatic efficiency predictions for each case are presented to show the effect of spacing on aerodynamic performance.
Author

Aerodynamic Characteristics; Navier-Stokes Equation; Turbine Blades; Unsteady Flow; Turbulence Models; Supersonic Turbines

19960024810; 96N26969 Von Karman Inst. for Fluid Dynamics, Rhode-Saint-Genese, Belgium

A Review of the Research on Unsteady Turbine Blade Wake Characteristics

Cicatelli, G., Von Karman Inst. for Fluid Dynamics, Belgium; Sieverding, C. H., Von Karman Inst. for Fluid Dynamics, Belgium; Loss Mechanisms and Unsteady Flows in Turbomachines; Jan. 1996; 14p; In English; Also announced as 19960024804; Copyright Waived; Avail: CASI; A03, Hardcopy; A06, Microfiche

The research on unsteady wake flows behind turbine blades is the orphan of turbomachinery research. The statement that blade wake flows are characterized by the presence of so called von Karman vortex streets, is a common-place, but yet, only very few researchers concluded that this phenomenon deserved their attention. The paper attempts to summarize the available bits and pieces and provides the reader with the state of art of our present knowledge in this domain. In many circumstances, the authors have to rely on research results on flows behind cylinder. Schlieren, interferometry and smoke visualizations are used to illustrate the existence of vortex street in a wide variety of cascade and flow conditions. The difference in the Strouhal number describing the vortex

shedding frequency behind turbine blades and cylinders is explained and the effect of the vortical motion on the base pressure is discussed. A description of the vortex generation mechanism and the wake structure, developed for flows over cylinders, proves useful in the interpretation of the instantaneous stream lines in time varying turbine blade wakes containing large coherent structures.
Author
Base Pressure; Vortex Streets; Turbine Blades; Cascade Flow; Vortex Shedding; Unsteady Flow

19960024811; 96N26970 Ancona Univ., Dipt. di Energetica., Ancona, Italy
Secondary Flows in a Turbine Cascade at Off-Design Conditions
Bassi, F., Ancona Univ., Italy; Savini, M., Bergamo Univ., Italy; Loss Mechanisms and Unsteady Flows in Turbomachines; Jan. 1996; 10p; In English; Also announced as 19960024804; Copyright Waived; Avail: CASI; A02, Hardcopy; A06, Microfiche

In this work a finite volume full 3-D Navier-Stokes solver equipped with the two equation k-w turbulence model is described. The code is applied to the analysis of the secondary flows occurring in a low aspect ratio turbine cascade both at design and at highly off-design incidence angle. Numerical results and experimental data concerning loss, flow angle, and turbulent kinetic energy distributions are compared and discussed. Aim of the work is to investigate the capability of the physical models, namely of the turbulence model, to simulate the marked three-dimensional flow features that characterize the secondary flows at off-design conditions
Author
Navier-Stokes Equation; Three Dimensional Flow; Turbine Blades; Turbulence Models; Cascade Flow; Computational Grids

19960024812; 96N26971 Durham Univ., School of Engineering., UK
Turbulence and Transition in Secondary Flows in a Turbine Cascade
Moore, H., Durham Univ., UK; Gregory-Smith, D. G., Durham Univ., UK; Loss Mechanisms and Unsteady Flows in Turbomachines; Jan. 1996; 12p; In English; Also announced as 19960024804; Copyright Waived; Avail: CASI; A03, Hardcopy; A06, Microfiche

The accurate prediction of flows in turbomachinery blade rows requires realistic modelling of turbulence and transition, particularly for loss estimation. The complex secondary flow within a large scale cascade of turbine rotor blades has been measured in detail. In particular hot wire measurements have been made close to the end walls in order to characterize the three-dimensional boundary layers. Although at inlet on the end-wall there is a well developed turbulent boundary layer, this is rolled up into a loss core by the secondary velocities and a new highly skewed boundary layer is formed which exhibits significant laminar characteristics. The physical insights gained by this work have implications for the development of turbulence and transition modelling as well as providing test data for the verification of CFD codes.
Author
Three Dimensional Boundary Layer; Turbomachine Blades; Turbulent Boundary Layer; Cascade Flow; Turbulence Models; Navier-Stokes Equation; Hot-Wire Anemometers

19960024813; 96N26972 Marine Technical Univ., Saint Petersburg, Russia
Three Dimensional Modelling and Secondary Flows Control in Gas Turbines
Tikhomirov, B. A., Marine Technical Univ., Russia; Pogodin, J. M., Marine Technical Univ., Russia; Loss Mechanisms and Unsteady Flows in Turbomachines; Jan. 1996; 12p; In English; Also announced as 19960024804; Copyright Waived; Avail: CASI; A03, Hardcopy; A06, Microfiche

Three-dimensional model of vortex flow in a passage is based on Clebsh's Transformation. The associated computer codes consist of the following modules: finite elements mesh automatic generation, finite elements optimum renumeration, calculation of flow, processing of results, analysis of losses. Losses analysis uses typical elements losses models: friction, turn, expansion losses and losses due to equalization throughout the flow. Two last components of losses are calculated by use of integrating throughout the streamlines of the passage element where the results of 3-D flow calculations are known. The model of mass flux averaged flow in turbine rows with end wall inclination, tangential lean or curvature of the blades has been developed. It is made due to the additional boundary condition along geometrical throats line of blade to blade passage. The boundary condition is radial distribution of meridional stream lines inclination which is based on the principle of flow mass flux maximum. Correlations between different flow models has been developed. Research of secondary flows control problem focuses on the effect of end wall fences, wall and blade ledges use for cross flows limitation. Combination of wall ledge and suction from ledge surface, blade through slots and holes used for gas transportation from pressure to suction blade surface in secondary flows area are also experimentally studied in axial turbines.
Author
Axial Flow Turbines; Turbine Blades; Three Dimensional Models; Vortex Sheets; Secondary Flow; Flow Characteristics; Turbulence Models

19960024814; 96N26973 Vrije Univ., Dept. of Fluid Mechanics., Brussels, Belgium
Tip Clearance Flow and Loss in Axial Compressor Cascades
Kang, Shun, Vrije Univ., Belgium; Hirsch, Charles, Vrije Univ., Belgium; Loss Mechanisms and Unsteady Flows in

Turbomachines; Jan. 1996; 16p; In English; Also announced as 19960024804; Original contains color illustrations; Copyright Waived; Avail: CASI; A03, Hardcopy; A06, Microfiche

The tip clearance flow and loss production in a low speed linear compressor cascade with a moving endwall is investigated. The analysis is based on numerical computations with a 3D viscous code, at one tip clearance level. Comparisons with experimental data are given for the stationary wall case. The moving wall modifies significantly the secondary flow field near the wall, by generating the skewing of the inlet endwall boundary layer and dragging the tip leakage vortex towards the pressure side. It is found that the leakage flow near the wall is collateral over most of the chord. Close to the tip surface, however, the flow is highly skewed due to the vena contracta. With the moving wall, the leakage mass flow and pressure drop are noticeably increased.

Author

Wall Flow; Flow Distribution; Leakage; Secondary Flow; Cascade Flow; Three Dimensional Boundary Layer; Turbocompressors; Turbine Blades; Tip Vanes

19960024815; 96N26974 National Technical Univ., Lab. of Thermal Turbomachines., Athens, Greece

Prediction of Losses Due to the Tip Clearance Presence in Axial Flow Machines

Nikolos, I. K., National Technical Univ., Greece; Douvikas, D. I., National Technical Univ., Greece; Papailiou, K. D., National Technical Univ., Greece; Loss Mechanisms and Unsteady Flows in Turbomachines; Jan. 1996; 16p; In English; Also announced as 19960024804

Contract(s)/Grant(s): AER2-CT92-0039-AC3A; Copyright Waived; Avail: CASI; A03, Hardcopy; A06, Microfiche

Theoretical models for the investigation of the phenomena connected with the tip clearance are presented. A model was developed for the description of the flow through the gap, which recognizes the basic flow characteristics inside the gap. The mass flow rate as well as the pressure losses inside the gap are predicted with sufficient accuracy. A new method, based on the moment of momentum equation, is proposed for the calculation of the shed vorticity and the formation of the leakage vortex. A diffusion model for the vorticity distribution is used for the calculation of the pressure deficit field, so that the total pressure loss profile due to the presence of the leakage vortex, is derived. An algorithm is being set up for the implementation of the above models in a secondary flow calculation method. A complete theoretical procedure was developed, which calculates the crcumferentially averaged flow quantities and their radial variation due to the tip clearance effects. The calculation takes place in successive planes, where a Poisson equation is solved, in order to provide the kinematic field. The self induced velocity is used for the positioning of the leakage vortex. The calculated pressure deficit due to the vortex presence is used, through an iterative procedure, in order to modify the pressure difference in the tip region. Calculation results are compared to experimental ones for axial flow machines. The agreement between theory and experiment is good.

Author

Poisson Equation; Flow Characteristics; Axial Flow Turbines; Tip Vanes; Pressure Distribution; Mass Flow Rate; Vorticity

19960024816; 96N26975 Turbine Specialties, Inc., Salina, KS USA

Tip Leakage Loss Development in a Linear Turbine Cascade

Peters, David W., Turbine Specialties, Inc., USA; Moore, John, Virginia Polytechnic Inst. and State Univ., USA; Loss Mechanisms and Unsteady Flows in Turbomachines; Jan. 1996; 14p; In English; Also announced as 19960024804; Copyright Waived; Avail: CASI; A03, Hardcopy; A06, Microfiche

Tip leakage losses were studied in a large-scale linear turbine cascade with a tip clearance gap equal to 2.1 percent of blade height. The cascade was located at the exit of a low speed wind tunnel; the blade exit Reynolds number based upon blade axial chord was 4.5x10(exp 5). The flow was measured at a plane 0.96 axial chords downstream from the blade leading edge. Earlier studies performed at the tip gap exit and at a downstream plane 1.4 axial chords from the blade leading edge were utilized with the present study to understand loss development better. The effect of tip leakage and the corresponding loss production mechanisms involved as the flow mixes out were analyzed. The flow properties at the measurement plane were numerically integrated. to estimate the maximum potential loss of the cascade, the flow was mixed-out through a momentum analysis. The loss at the measurement plane due to tip leakage was found to be equal to the sum of the total pressure loss within the tip gap and the dissipated tip gap secondary kinetic energy. As the flow proceeded downstream, losses were attributed to dissipation of secondary kinetic energy, trailing edge wake mixing, endwall losses, and primary flow mixing.

Author

Cascade Flow; Flow Characteristics; Low Speed Wind Tunnels; Leakage; Tip Vanes; Energy Dissipation; Turbine Blades; Vortices

19960024817; 96N26976 BMW Rolls-Royce AeroEngines G.m.b.H., Dahlewitz, Germany

Reduction of Tip Clearance and Tip Clearance Noise in Axial-Flow Machines

Kameier, Frank, BMW Rolls-Royce AeroEngines G.m.b.H., Germany; Neise, Wolfgang, Deutsche Forschungsanstalt fuer Luft- und Raumfahrt, Germany; Loss Mechanisms and Unsteady Flows in Turbomachines; Jan. 1996; 14p; In English; Also announced as 19960024804; Copyright Waived; Avail: CASI; A03, Hardcopy; A06, Microfiche

An experimental study is described to investigate the negative effects of the tip clearance gap on the aerodynamic and acoustic performance of axial turbomachines. In addition to the increased broadband levels reported in the literature when the tip clearance is enlarged, significant level increases were observed within narrow frequency bands below the blade passing frequency. Measurements of the pressure and velocity fluctuations in the vicinity of the blade tips reveal that the tip clearance noise is associated with a rotating flow instability at the blade tip which in turn is only present under reversed flow conditions in the tip clearance gap. A turbulence generator inserted into tip clearance gap is found to be effective in eliminating the tip clearance noise and in improving the aerodynamic performance.
Author
Acoustic Properties; Aerodynamic Characteristics; Axial Flow; Blade Tips; Turbine Blades; Axial Flow Turbines; Noise Reduction; Reversed Flow

19960024818; 96N26977 Deutsche Forschungsanstalt fuer Luft- und Raumfahrt, Inst. fuer Stroemungsmechanik., Goettingen, Germany
Wake Development Downstream of a Transonic Turbine Inlet Guide Vane with Trailing Edge Ejection
Kapteijn, Cornelis, Deutsche Forschungsanstalt fuer Luft- und Raumfahrt, Germany; Loss Mechanisms and Unsteady Flows in Turbomachines; Jan. 1996; 8p; In English; Also announced as 19960024804
Contract(s)/Grant(s): AER2-CT92-0044; Copyright Waived; Avail: CASI; A02, Hardcopy; A06, Microfiche

The temperature at the inlet of modern high pressure turbines requires an effective cooling of the nozzle guide vanes. This project investigates an annular cascade of internally cooled nozzle guide vanes with bleeding through a slot at the pressure side near the trailing edge. Purpose of the actual work is the determination of the downstream flow field undisturbed by the presence of the rotor with emphasis on the mixing process between main flow and coolant flow. During the second part of this project the investigation of a complete stage will take place, where the present results will enable the elucidation of the flow field deviations due to the rotor presence.
Author
Annular Flow; Cascade Flow; Cooling; Guide Vanes; Supersonic Turbines; Wakes; Multiphase Flow; Inlet Nozzles

19960024819; 96N26978 Cincinnati Univ., Dept. of Aerospace Engineering and Engineering Mechanics., OH USA
Flow Predictions in Transonic Turbines Using Nonperiodic Grids
Hamed, A., Cincinnati Univ., USA; Yeuan, J. J., Cincinnati Univ., USA; Tabakoff, W., Cincinnati Univ., USA; Loss Mechanisms and Unsteady Flows in Turbomachines; Jan. 1996; 10p; In English; Also announced as 19960024804; Copyright Waived; Avail: CASI; A02, Hardcopy; A06, Microfiche

Numerical results are presented for viscous flow through transonic turbine blades using different turbulence models and H-type grids. The numerical schemes for the solution of the compressible Navier Stokes equations are developed with an option of conservative zonal approach for interpolation across non-periodic boundaries to minimize numerical errors. This makes it possible to use nonperiodic type grids which are more orthogonal and less skewed for improved accuracy in the prediction of turbine blade performance. The results obtained using traditional and nonperiodic H-type grids are compared with the experimental results for transonic turbine cascades and turbine stator vanes.
Author
Navier-Stokes Equation; Turbine Blades; Supersonic Turbines; Turbulence Models; Inlet Nozzles; Three Dimensional Flow; Performance Prediction

19960024820; 96N26979 Wright Lab., Wright-Patterson AFB, OH USA
Development of an Off-Design Loss Model for Transonic Compressor Design
Bloch, Gregory S., Wright Lab., USA; Copenhaver, William W., Wright Lab., USA; OBrien, Walter F., Virginia Polytechnic Inst. and State Univ., USA; Loss Mechanisms and Unsteady Flows in Turbomachines; Jan. 1996; 14p; In English; Also announced as 19960024804; Copyright Waived; Avail: CASI; A03, Hardcopy; A06, Microfiche

A numerical investigation has been conducted to determine the changes in shock geometry that will occur with throttle setting for a modern transonic fan. The passage shock was found to be approximately normal near stall, with increasing obliqueness at higher mass flows. At peak efficiency, the shock system consists of an oblique passage shock followed by a normal shock, with the 2 shocks coalescing near the suction surface. As the back pressure is reduced further, the rotor becomes completely choked and the second shock moves downstream, becoming a separate, full-passage shock. The Mach number upstream of the second shock increases steadily as the shock moves downstream with decreasing back pressure. For all values of back pressure, the second passage shock is approximately normal to the predominant flow direction. Because the flow is supersonic downstream of the first passage shock, there are no changes in the first passage shock along this portion of the characteristic. A simple, but fundamentally based model has been developed which captures the qualitative trends of shock performance over the entire range of design-speed mass flows for a transonic fan. Model predictions are compared with measured data at multiple spanwise locations. The results of the comparison suggest that shock losses contribute a significant share of the total loss, but other mechanisms must also be considered.
Author

Transonic Compressors; Oblique Shock Waves; Mathematical Models; Mass Flow; Turbine Blades; Blade Tips

19960024821; 96N26980 Deutsche Forschungsanstalt fuer Luft- und Raumfahrt, Inst. fuer Antriebstechnik., Cologne, Germany

Shock-Wave Turbulent Boundary Layer Interaction in a Highly Loaded Transonic Fan Blade Cascade

Schreiber, Heinz-Adolf, Deutsche Forschungsanstalt fuer Luft- und Raumfahrt, Germany; Loss Mechanisms and Unsteady Flows in Turbomachines; Jan. 1996; 14p; In English; Also announced as 19960024804; Copyright Waived; Avail: CASI; A03, Hardcopy; A06, Microfiche

Results of experimental investigations on strong shock-wave turbulent boundary layer interactions in a transonic compressor cascade are presented. The tests have been conducted in a linear cascade facility at inlet Mach numbers, ranging from 1.37 to 1.59, for freestream Reynolds numbers of about 2.7x10(exp 6). The cascade tested is typical for a fan blade tip section with little flow turning, providing a static pressure ratio of more than 2.0. Main interest is concentrated on the blade suction surface, where the oblique extension of the bow shock of the neighboring blade forms a Mach reflection in combination with a boundary layer separation. Structure and loss mechanism of the shock system within the blade passage is discussed. Detailed measurements of the boundary layer within the interaction region and the rear part of the blade were performed. The results show that shock induced boundary layer separation on the tested blade can be controlled for inlet Mach numbers up to 1.42 . At preshock Mach numbers of 1.32 to 1.44, there are only local boundary layer separations with reattachments. Complete boundary layer separation cannot be avoided beyond inlet Mach numbers of 1.45. However, the magnitude of the separation and thus the total pressure losses can be controlled by increasing the axial stream sheet contraction of the blade channel.

Author

Boundary Layer Separation; Shock Waves; Turbulent Boundary Layer; Fan Blades; Cascade Flow; Turbocompressors; Transonic Flow

19960024822; 96N26981 Roketsan A.S. Ankara, Turkey

Loss Prediction on Axial Flow Cascades Using a Navier-Stokes Solver

Tinaztepe, H. Tugrul, Roketsan A.S. Ankara, Turkey; Akmandor, I. Sinan, Middle East Technical Univ., Turkey; Ucer, Ahmet S., Middle East Technical Univ., Turkey; Loss Mechanisms and Unsteady Flows in Turbomachines; Jan. 1996; 14p; In English; Also announced as 19960024804; Copyright Waived; Avail: CASI; A03, Hardcopy; A06, Microfiche

In this paper, results of a 3D Navier-Stokes flow solver which uses a cell vertex finite volume method are given. The code uses Lax-Wendroff time integration method and Baldwin-Lomax turbulence model. Validation of the code are presented briefly in terms of modeling accuracy due to viscous effects. An attempt is made to predict shock boundary layer interaction on an airfoil. The paper also gives loss distributions. Effects like separation and secondary flow due to a high turning turbine cascade are discussed. Physical aspects of secondary flow and loss generation mechanisms are investigated in the predicted flow field.

Author

Axial Flow; Cascade Flow; Navier-Stokes Equation; Turbulence Models; Flow Distribution; Three Dimensional Flow; Axial Flow Turbines; Shock Wave Profiles

19960024823; 96N26982 British Columbia Univ., Dept. of Mechanical Engineering., Vancouver, British Columbia Canada

Rotor-Stator Interaction Effects on Turbulent Boundary Layers

Evans, R. L., British Columbia Univ., Canada; Holland, R. M., British Columbia Univ., Canada; Loss Mechanisms and Unsteady Flows in Turbomachines; Jan. 1996; 8p; In English; Also announced as 19960024804; Copyright Waived; Avail: CASI; A02, Hardcopy; A06, Microfiche

Compressor and turbine blade boundary layers in axial-flow turbomachines are subject to periodically disturbed flow. This study modelled these conditions in a wind tunnel with circular cylinders traversing in front of a flat plate. Turbulent boundary layer velocity profiles on the flat plate were measured with a hot-wire anemometer. The ensemble-averaged turbulence intensity in the boundary layer was found to be significantly increased by the unsteady free stream flow, resulting in increases in the characteristic boundary layer thicknesses and the skin friction coefficient. The vortex structure within the unsteady cylinder wakes was found to explain the behavior of the velocity and phase angle profiles in the boundary layer.

Author

Rotor Blades (Turbomachinery); Stator Blades; Turbulent Boundary Layer; Turbulence Models; Compressor Blades; Hot-Wire Anemometers; Boundary Layer Separation

19960024824; 96N26983 Paris VI Univ., Orsay, France

Experimental and Numerical Investigation of Unsteady Flow Properties in a Stator of Multistage Axial Flow compressor

Miton, Hubert, Paris VI Univ., France; Belhabib, Mustapha, Paris VI Univ., France; Kus, Umit, Paris VI Univ., France; Loss Mechanisms and Unsteady Flows in Turbomachines; Jan. 1996; 18p; In English; Also announced as 19960024804; Copyright Waived; Avail: CASI; A03, Hardcopy; A06, Microfiche

This analysis has been performed from tests realized in a four stage low speed axial flow compressor (4000 rpm). First, two unsteady velocities components has been measured,

using hot wire anemometry, directly upstream and downstream of the first stage stator in the (r,O) plane. Simultaneously, unsteady flow characteristics has been collected at four blade surfaces. Experimental techniques consisted first in pressure transducers fitted in the inner part of the profiles, then, in series of hot films (29), bounded, at two different radial positions, to the blade surfaces. These films were uniformly distributed orthogonaly to the axial direction, from leading to trailing edges. Data have been processed in view to identify the multiple origins of excitation from the other parts of the machine and to describe the behavior of the boundary layer along the blade surfaces. Comparison of the experimental results with the ones given by a two-dimensional blade to blade computation method, taking into account the development of the boundary layer, has been performed.
Author
Unsteady Flow; Hot-Wire Anemometers; Turbocompressors; Turbine Blades; Boundary Layer Equations

19960024825; 96N26984 Universitaet der Bundeswehr Muenchen, Inst. fuer Strahlantriebe, Neubiberg, Germany
The Effect of Incoming Wakes on Boundary Layer Transition of a Highly Loaded Turbine Cascade
Engber, M., Universitaet der Bundeswehr Muenchen, Germany; Fottner, L., Universitaet der Bundeswehr Muenchen, Germany; Loss Mechanisms and Unsteady Flows in Turbomachines; Jan. 1996; 12p; In English; Also announced as 19960024804; Copyright Waived; Avail: CASI; A03, Hardcopy; A06, Microfiche

Extensive pressure and thin-film measurements have been performed on a linear turbine cascade with variations of inlet angle, inlet Reynolds number, and inlet turbulence intensity at homogeneous and inhomogeneous inflow conditions. Especially the occurrence of laminar separation bubbles was investigated. With both measuring techniques their existence could be verified and the characteristic points of a bubble could be located. Incoming wakes similar to those in real turbomachines were simulated by a cascade of cylindrical bars which was positioned in front of the turbine cascade. Diameter and distance of the bars to the blades were chosen to produce nonuniformities in the cascade inlet plane similar to those in real engines. The wake impingement point could be varied by moving the cylindrical bars in circumferential direction parallel to the turbine cascade front With this set-up the so called rotor-rotor or stator-stator interaction of a multistage turbine can be simulated. Measurements of direction, quantity and actuation of the velocity inside the boundary layer have been conducted with a Laser-2-Focus measuring technique. The analysis of the experiments results in a better understanding of the transitional phenomena on blade profiles and allows to improve empirical correlations for the calculation of transition in boundary layer.
Author
Boundary Layer Transition; Bubbles; Cylindrical Bodies; Cascade Flow; Turbomachinery; Wakes; Rotor Blades (Turbomachinery); Stator Blades; Turbulence Models

19960024826; 96N26985 Oxford Univ., Dept. of Engineering Science., Oxford, UK
Unsteady Pressure and Heat Transfer Measurements on a Rotating Blade Surface in a Transient Flow Facility
Moss, R. W., Oxford Univ., UK; Sheldrake, C. D., Oxford Univ., UK; Ainsworth, R. W., Oxford Univ., UK; Smith, A. D., Rolls-Royce Ltd., UK; Dancer, S. N., Rolls-Royce Ltd., UK; Loss Mechanisms and Unsteady Flows in Turbomachines; Jan. 1996; 12p; In English; Also announced as 19960024804; Copyright Waived; Avail: CASI; A03, Hardcopy; A06, Microfiche

The Oxford rotating turbine facility has been used to obtain measurements of blade surface heat flux, static pressure and relative inlet velocity. The measurements provide time resolved data which is compared with predictions from a two-dimensional unsteady flow calculation.
Author
Turbulent Heat Transfer; Unsteady Flow; Turbine Blades; Rotor Blades (Turbomachinery); Two Dimensional Flow

19960024827; 96N26986 Cambridge Univ., Cambridge, UK
Wake Passing in LP Turbine Blades
Banieghbal, M. R., Cambridge Univ., UK; Curtis, E. M., Cambridge Univ., UK; Denton, J. D., Cambridge Univ., UK; Hodson, H. P., Cambridge Univ., UK; Huntsman, I., Cambridge Univ., UK; Schulte, V., Cambridge Univ., UK; Harvey, N. W., Rolls-Royce Ltd., UK; Steele, A. B., Rolls-Royce Ltd., UK; Loss Mechanisms and Unsteady Flows in Turbomachines; Jan. 1996; 12p; In English; Also announced as 19960024804; Copyright Waived; Avail: CASI; A03, Hardcopy; A06, Microfiche

A study of the effect of wake passing on the boundary layer of a modern low pressure (LP) turbine is presented. Typically, such blading operates at low Reynolds numbers. As a consequence, significant regions of laminar flow might be expected to occur together with transition via laminar separation bubbles. This paper examines the behavior of the surface boundary layers in environments typical of modern low pressure turbines. Results were obtained from arrays of surface mounted hot-film anemometers in the 3rd stage stator of a LP turbine. The results are considered in relation to more detailed experiments carried out in a steady flow high-speed linear turbine cascade and also with reference to tests carried out in a low-speed linear turbine cascade fitted with a mechanism that simulates the presence of upstream rotor wakes. It is found that, although there are some differences in details, the results from the turbine are consistent with the cascade observations. The suction surface boundary layer is little affected by wakes over the forward half of the surface and remains laminar. Downstream of peak suction, laminar separation occurs and

the main effect of the wakes appears to be to promote transition in the separated region. Nevertheless, even in the presence of wakes, laminar flow persists for part of the time right up to the trailing edge. Over most of the Reynolds number range, the wake effects add to the loss, but at the lowest Reynolds numbers ((Re less than 1x1O)(exp 5)), the wakes reduce the profile loss as compared to the steady flow situation, probably by assisting reattachment. The large element of laminar flow, even though there is considerable diffusion over the rear part of the suction surface, leads to low losses for this profile.
Author
Turbine Blades; Wakes; Laminar Flow; Cascade Flow; Boundary Layer Transition; Wind Tunnel Tests; Rotor Blades (Turbomachinery); Stators

19960024828; 96N26987 Defence Research Agency, Pyestock, UK
Unsteady Measurements in an Axial Flow Turbine
Hilditch, M. A., Defence Research Agency, UK; Smith, G. C., Defence Research Agency, UK; Anderson, S. J., Defence Research Agency, UK; Chana, K. S., Defence Research Agency, UK; Jones, T. V., Oxford Univ., UK; Ainsworth, R. W., Oxford Univ., UK; Oldfield, M. L. G., Oxford Univ., UK; Loss Mechanisms and Unsteady Flows in Turbomachines; Jan. 1996; 10p; In English; Also announced as 19960024804; Copyright Waived; Avail: CASI; A02, Hardcopy; A06, Microfiche

Fast response instrumentation has been used to measure the unsteady flow field in an axial flow turbine mounted in a transient test facility. The results show significant fluctuations in static pressure and heat transfer rate at mid-span on the rotor blade. The rotor is shown to have a limited influence on the heat transfer to the nozzle guide vane at midspan.
Author
Axial Flow Turbines; Flow Distribution; Guide Vanes; Unsteady Flow; Rotor Blades (Turbomachinery); Pressure Measurement

19960024829; 96N26988 Royal Military Coll. of Canada, Kingston, Ontario Canada
Rotating Stall in Turbojet Engine Compressors
Borys, W. J., Royal Military Coll. of Canada, Canada; Moffatt, W. C., Royal Military Coll. of Canada, Canada; Loss Mechanisms and Unsteady Flows in Turbomachines; Jan. 1996; 12p; In English; Also announced as 19960024804; Copyright Waived; Avail: CASI; A03, Hardcopy; A06, Microfiche

Losses in an axial compressor are generally ascribed to the effects of shock waves or viscosity, the latter including boundary layer growth, separation, wakes, and secondary flows. Rotating stall is a viscosity-related phenomenon whose effects are well understood, but whose origins and prediction have defied definitive analysis. Experimental approaches to understanding such stalls have for very practical reasons been performed primarily on laboratory scale compressors; there is little information in the open literature on installed units. This paper presents the results of several studies of rotating stall characteristics for a compressor in a turbojet engine, including determination of the conditions in which stall is found, the characteristics of the stall pattern, some details of the flows within the cells, and the effects of inlet distortions on the nature of the stall. Flow characteristics were found using hot film anemometer probes for all the tests. Also included is a brief description of an experimental evaluation of the effects of temperature changes within the compressor on the accuracy of constant temperature hot film anemometer measurements.
Author
Aerodynamic Stalling; Boundary Layer Separation; Temperature Measurement; Turbojet Engines; Viscosity; Rotating Stalls; Wakes; Turbocompressors

19960024830; 96N26989 Motoren- und Turbinen-Union Muenchen G.m.b.H., Munich, Germany
Stall Inception and Surge in High-Speed Axial Flow Compressors
Breuer, T., Motoren- und Turbinen-Union Muenchen G.m.b.H., Germany; Servaty, S., Motoren- und Turbinen-Union Muenchen G.m.b.H., Germany; Loss Mechanisms and Unsteady Flows in Turbomachines; Jan. 1996; 18p; In English; Also announced as 19960024804
Contract(s)/Grant(s): AER2-CT92-0039; Copyright Waived; Avail: CASI; A03, Hardcopy; A06, Microfiche

The results of experimental and numerical studies to examine the inception process of rotating stall and surge are presented. Unsteady pressure measurements have been carried out in a 3-stage high speed compressor to reveal characteristic features of instability onset. Tests have been conducted with clean intake as well as a circumferential inlet distortion. Analysis of the data shows that instability starts from small amplitude disturbances rotating in rotor direction, finally leading to either rotating stall or surge. A numerical model is used to calculate the instability onset in a compression system comprising of ducts, blade rows and volumes. The model is based on the unsteady two-dimensional Euler-equations in conservation law form. The influence of blade rows is simulated by source terms to account for pressure loss and energy input. Some results of computations are compared to the experimental results.
Author
Rotating Stalls; Turbocompressors; Mathematical Models; Surges; Rotor Blades (Turbomachinery); Unsteady Flow; Active Control

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