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AERONAUTICAL ENGINEERING

A CONTINUING BIBLIOGRAPHY WITH INDEXES

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Typical Report Citation and Abstract

❶ 19970001126 NASA Langley Research Center, Hampton, VA USA
❷ **Water Tunnel Flow Visualization Study Through Poststall of 12 Novel Planform Shapes**
❸ Gatlin, Gregory M., NASA Langley Research Center, USA Neuhart, Dan H., Lockheed Engineering and Sciences Co., USA;
❹ Mar. 1996; 130p; In English
❺ Contract(s)/Grant(s): RTOP 505-68-70-04
❻ Report No(s): NASA-TM-4663; NAS 1.15:4663; L-17418; No Copyright; Avail: CASI; A07, Hardcopy; A02, Microfiche
❼ To determine the flow field characteristics of 12 planform geometries, a flow visualization investigation was conducted in the Langley 16- by 24-Inch Water Tunnel. Concepts studied included flat plate representations of diamond wings, twin bodies, double wings, cutout wing configurations, and serrated forebodies. The off-surface flow patterns were identified by injecting colored dyes from the model surface into the free-stream flow. These dyes generally were injected so that the localized vortical flow patterns were visualized. Photographs were obtained for angles of attack ranging from 10' to 50', and all investigations were conducted at a test section speed of 0.25 ft per sec. Results from the investigation indicate that the formation of strong vortices on highly swept forebodies can improve poststall lift characteristics; however, the asymmetric bursting of these vortices could produce substantial control problems. A wing cutout was found to significantly alter the position of the forebody vortex on the wing by shifting the vortex inboard. Serrated forebodies were found to effectively generate multiple vortices over the configuration. Vortices from 65' swept forebody serrations tended to roll together, while vortices from 40' swept serrations were more effective in generating additional lift caused by their more independent nature.
❽ Author
❾ *Water Tunnel Tests; Flow Visualization; Flow Distribution; Free Flow; Planforms; Wing Profiles; Aerodynamic Configurations*

Key

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2. Title
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9. Subject Terms

AERONAUTICAL ENGINEERING

A Continuing Bibliography (Suppl. 389)

DECEMBER 11, 1998

01 AERONAUTICS

19980236619 Purdue Univ., School of Aeronautics and Astronautics, West Lafayette, IN USA
1997 NASA Academy in Aeronautics ***Final Report***
Andrisani, Dominick, II, Purdue Univ., USA; Jun. 1998; 29p; In English
Contract(s)/Grant(s): NAG4-114
Report No.(s): NASA/CR-1998-208054; NAS 1.26:208054; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The NASA Academy in Aeronautics at the Dryden Flight Research Center (DFRC) was a ten-week summer leadership training program conducted for the first time in the summer of 1997. Funding was provided by a contract between DFRC and Purdue University. Mr. Lee Duke of DFRC was the contract monitor, and Professor Dominick Andrisani was the principal investigator. Five student research associates participated in the program. Biographies of the research associates are given in Appendix 1. Dominick Andrisani served as Dean of the NASA Academy in Aeronautics. NASA Academy in Aeronautics is a unique summer institute of higher learning that endeavors to provide insight into all of the elements that make NASA aeronautical research possible. At the same time the Academy assigns the research associate to be mentored by one of NASA!s best researchers so that they can contribute towards an active flight research program. Aeronautical research and development are an investment in the future, and NASA Academy is an investment in aeronautical leaders of the future. The Academy was run by the Indiana Space Grant Consortium at Purdue in strategic partnership with the National Space Grant College and Fellowship Program. Research associates at the Academy were selected with help from the Space Grant Consortium that sponsored the research associate. Research associate stipend and travel to DFRC were paid by the students' Space Grant Consortium. All other student expenses were paid by the Academy. Since the Academy at DFRC had only five students the opportunity for individual growth and attention was unique in the country. About 30% of the working time and most of the social time of the students were be spent as a "group" or "team." This time was devoted to exchange of ideas, on forays into the highest levels of decision making, and in executing aeronautical research. This was done by interviewing leaders throughout the aerospace industry, seminars, working dinners, and informal discussions. The other 70% of the working time was spent working on the technical research project with the engineering mentors. Abstracts of those projects are given in Appendix 4.
Author
Education; Decision Making; Students; Universities; Leadership

19980236949 NASA Langley Research Center, Hampton,VA USA
Aeronautical Engineering: A Continuing Bibliography with Indexes, Supplement 387
Nov. 13, 1998; 95p; In English
Report No.(s): NASA/SP-1998-7037/SUPPL387; NAS 1.21:7037/SUPPL387; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

This supplemental issue of Aeronautical Engineering, A Continuing Bibliography with Indexes (NASA/SP-1998-7037) lists 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.
CASI
Aerodynamics; Aeronautical Engineering; Bibliographies; Indexes (Documentation)

02
AERODYNAMICS

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

19980231974 NASA Langley Research Center, Hampton, VA USA

An Experimental Study at a Mach Number of 3 of the Effect of Turbulence Level and Sandpaper Type Roughness on Transition on a Flat Plate

Jones, Robert A., NASA Langley Research Center, USA; Mar. 1959; 48p; In English

Report No.(s): NASA-MEMO-2-9-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been conducted at a Mach number of 3 of the effect of turbulence level and sandpaper-type roughness on transition for a flat plate. The Reynolds number varied from 0.8 x 10(exp 6) to 1.8 x 10(exp 6) per inch; the settling-chamber turbulence level varied from 0.7 percent to 35 percent; and the heat transfer between the plate and the stream was negligible. Transition locations were determined by an optical method. This method was indicative of a permanent change in the boundary-layer density distribution rather than the onset of turbulent bursts. Results showed that, when transition was influenced by roughness, it moved in a way similar to its movement on a smooth plate. That is, it gradually approached the roughness location with either an increase in unit Reynolds number or an increase in turbulence level. For roughness submerged in the linear portion of the boundary-layer velocity profile, the square root of the roughness Reynolds number and the ratio of roughness height to boundary-layer displacement thickness gave similar results as parameters for predicting the effects of roughness. A range of each of these parameters which moved transition less than 10 percent was found and this range was a function of turbulence level.

Author

Boundary Layers; Turbulence; Flat Plates; Surface Roughness; Heat Transfer; Density Distribution

19980231982 Nanjing Univ. of Aeronautics and Astronautics, Nanjing, Jiangsu, China

On the Research of Interactions of Drag Plate the Tail

Mengbu, Qi, Nanjing Univ. of Aeronautics and Astronautics, Nanjing, China; Mingyan, Chen, Nanjing Univ. of Aeronautics and Astronautics, Nanjing, China; Journal of Nanjing University of Aeronautics and Astronautics; Jun. 1997; ISSN 1005-2615; Volume 29, No. 3, pp. 317-320; In Chinese; No Copyright; Avail: Issuing Activity, Hardcopy, Microfiche

The principle of the increase of the nose-up pitching moment by opening the drag plate for an airplane has been studied in NH-2 wind tunnel. The measurements of the separated vortex field and forces for drag plate show that the nose-up pitching moment is produced due to the large induced downwash angle from the separated vortices of drag plate at the tail. The induced downwash angle is reduced asymptotically along the axis of fuselage. The nose-up pitching can be eliminated by moving drag plate foreword to suitable location. If the location of drag plate is kept still, making use of the porous drag plate can reduce the vorticity and the nose-up pitching moment. The nose-up pitching moment can be reduced to acceptable level and enough drag can be kept by the selection of suitable porosity and location of drag plate.

Author

Wind Tunnels; Porous Plates; Drag Reduction; Aerodynamics; Aerodynamic Drag

19980231988 NASA Langley Research Center, Hampton, VA USA

Aerodynamic Characteristics in Sideslip of a Large-Scale 49 deg Sweptback Wing-Body-Tail Configuration with Blowing Applied Over the Flaps and Wing Leading Edge

McLemore, H. Clyde, NASA Langley Research Center, USA; Oct. 1958; 50p; In English

Report No.(s): NASA-MEMO-10-11-58L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been conducted in the Langley full-scale tunnel to determine the aerodynamic characteristics in sideslip of a large-scale 490 sweptback wing-body-tail configuration having wing leading- edge and flap-blowing boundary-layer control. The wing and tails had an aspect ratio of 3.5, a taper ratio of 0.3, and NACA 65AO06 airfoil sections parallel to the plane of symmetry. The tests were conducted over a range of angles of attack of about -5 deg to 28 deg for sideslip angles of 0 deg, -5.06 deg, -10.15 deg, and -15.18 deg. Lateral and longitudinal stability and control characteristics were obtained for6a minimized blowing rate. The Reynolds number of the tests was 5.2 x 10(exp 6), corresponding to a Mach number of 0.08. The results of the investigation showed that sideslip to angles of about -15 deg did not require, from a consideration of the longitudinal characteristics, blowing rates over the wing leading edge or flap greater than that established as minimum at zero sideslip. The optimum configuration was laterally and directionally stable through the complete lift-coefficient range including the stall; however, maximum lift for sideslip angles greater than about 50 was seriously limited by a deficiency of lateral control. Blowing over the leading edge of the retreating wing in sideslip at a rate greater than that established as minimum at zero sideslip was ineffective in improving the

lateral control characteristics. The optimum configuration at zero sideslip had no hysteresis of the aerodynamic parameters upon recovery from stall.
Author
Aerodynamic Characteristics; Sideslip; Body-Wing Configurations; Wind Tunnel Tests; Airfoil Profiles; Leading Edges; Swept-back Wings

19980231992 NASA Langley Research Center, Hampton, VA USA
Low-Speed Aerodynamic and Hydrodynamic Characteristics of a Proposed Supersonic Multijet Water-Based Hydro-Ski Aircraft with Upward-Rotating Engines
Petynia, William W., NASA Langley Research Center, USA; Croom, Delwin R., NASA Langley Research Center, USA; Davenport, Edwin E., NASA Langley Research Center, USA; Oct. 1958; 62p; In English
Report No.(s): NASA-MEMO-10-13-58L; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

The low-speed aerodynamic and hydrodynamic characteristics of a proposed multijet water-based aircraft configuration for supersonic operation have been investigated. The design features include upward-rotating engines, body indentation, a single hydro-ski, and a wing with an aspect ratio of 3.0, a taper ratio of 0.143, 36.90 sweepback of the quarter-chord line, and NACA 65A004 airfoil sections. For the aerodynamic investigation, with the flaps retracted, the model was longitudinally and directionally stable up to the stall. The all-movable horizontal tail was capable of trimming the model up to a lift coefficient of approximately 0.87. All flap configurations investigated had a tendency to become longitudinally unstable at stall. The effectiveness of the all-movable horizontal tail increased with increasing lift coefficient for all flap configurations investigated; however, with the large static margin of the configuration with the center of gravity at 0.25 mean aerodynamic chord, the all-movable horizontal tail was not powerful enough to trim all the various flapped configurations investigated throughout the angle-of-attack range. For the hydrodynamic investigation, longitudinal stability during take-offs and landings was satisfactory. Decreasing the area of the hydro-ski 60 percent increased the maximum resistance and emergence speed 40 and 70 percent, respectively. Without the jet exhaust, the resistance was reduced by simulating the vertical-lift component of the forward engines rotated upward. However, the jet exhaust of the forward engines increased the maximum resistance approximately 60 percent. The engine inlets and horizontal tail were free from spray for all loads investigated and for both hydro-ski sizes.
Author
Aerodynamic Coefficients; Aircraft Configurations; Longitudinal Stability; Rotating Bodies; Loads (Forces)

19980231994 NASA Langley Research Center, Hampton, VA USA
The Total-Pressure Recovery and Drag Characteristics of Several Auxiliary Inlets at Transonic Speeds
Dennard, John S., NASA Langley Research Center, USA; Mar. 1959; 62p; In English
Report No.(s): NASA-MEMO-12-21-58L; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

Several flush and scoop-type auxiliary inlets have been tested for a range of Mach numbers from 0.55 to 1.3 to determine their transonic total-pressure recovery and drag characteristics. The inlet dimensions were comparable with the thickness of the boundary layer in which they were tested. Results indicate that flush inlets should be inclined at very shallow angles with respect to the surface for optimum total-pressure recovery and drag characteristics. Deep, narrow inlets have lower drag than wide shallow ones at Mach numbers greater than 0.9 but at lower Mach numbers the wider inlets proved superior. Inlets with a shallow approach ramp, 7 deg, and diverging ramp walls which incorporated boundary-layer bypass had lower drag than any other inlet tested for Mach numbers up to 1.2 and had the highest pressure recovery of all of the flush inlets. The scoop inlets, which operated in a higher velocity flow than the flush inlets, had higher drag coefficients. Several of these auxiliary inlets projected multiple, periodic shock waves into the stream when they were operated at low mass-flow ratios.
Author
Aerodynamic Coefficients; Transonic Speed; Pressure Recovery; Aerodynamic Drag; Boundary Layers; Intake Systems

19980231995 NASA Lewis Research Center, Cleveland, OH USA
Analysis and Evaluation of Supersonic Underwing Heat Addition
Luidens, Roger W., NASA Lewis Research Center, USA; Flaherty, Richard J., NASA Lewis Research Center, USA; Apr. 1959; 58p; In English
Report No.(s): NASA-MEMO-3-17-59E; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

The linearized theory for heat addition under a wing has been developed to optimize wing geometry, heat addition, and angle of attack. The optimum wing has all of the thickness on the underside of the airfoil, with maximum-thickness point well downstream, has a moderate thickness ratio, and operates at an optimum angle of attack. The heat addition is confined between the fore Mach waves from under the trailing surface of the wing. by linearized theory, a wing at optimum angle of attack may have a range

efficiency about twice that of a wing at zero angle of attack. More rigorous calculations using the method of characteristics for particular flow models were made for heating under a flat-plate wing and for several wings with thickness, both with heat additions concentrated near the wing. The more rigorous calculations yield in practical cases efficiencies about half those estimated by linear theory. An analysis indicates that distributing the heat addition between the fore waves from the undertrailing portion of the wing is a way of improving the performance, and further calculations appear desirable. A comparison of the conventional ramjet-plus wing with underwing heat addition when the heat addition is concentrated near the wing shows the ramjet to be superior on a range basis up to Mach number of about B. The heat distribution under the wing and the assumed ramjet and airframe performance may have a marked effect on this conclusion. Underwing heat addition can be useful in providing high-altitude maneuver capability at high flight Mach numbers for an airplane powered by conventional ramjets during cruise.
Author
Heating; Thickness Ratio; Wings; Fluid Flow; Airframes; Airfoils; Angle of Attack; Mathematical Models

19980231999 NASA Ames Research Center, Moffett Field, CA USA
The Effect of Moment of Area Rule Modifications on the Drag, Lift and Pitching Moment Characteristics of an Unswept Aspect Ratio 6 Wing and Body Combination
Dickey, Robert R., NASA Ames Research Center, USA; Mar. 1959; 32p; In English
Report No.(s): NASA-MEMO-2-24-59A; A-145; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An experimental investigation was conducted to determine the effect of moment-of-area-rule modifications on the drag, lift, and pitching-moment characteristics of a wing-body combination with a relatively high aspect-ratio unswept wing. The basic configuration consisted of an aspect-ratio-6 wing with a sharp leading edge and a thickness ratio of 0.06 mounted on a cut-off Sears-Haack body. The model with full moment-of-area-rule modifications had four contoured pods mounted on the wing and indentations in the body to improve the longitudinal distributions of area and moments of area. Also investigated were modifications employing pods and indentations that were only half the size of the full modifications and modifications with partial body indentations. The models were tested at angles of attack from -2 deg to +12 deg at Mach numbers from 0.6 to 1.4. In general, the moment-of-area-rule modifications had a large effect on the drag characteristics of the models but only a small effect on their lift and pitching-moment characteristics. The modifications provided substantial reductions in the zero-lift drag at transonic and low supersonic speeds, but at subsonic speeds the drag was increased. Near Mach number 1.0, the model with full modification provided the greatest reduction in drag, but at the highest test Mach numbers the half modification gave the largest drag reduction. In general, the percent reductions of zero- lift drag obtained with the aspect-ratio-6 wing were as great or greater than those previously obtained with aspect-ratio-3 wings. The effect of the modifications on the drag due to lift was small except at Mach numbers below 0.9 where the modified models had higher drag-rise factors. Above Mach number 0.9, the modified models had higher lift-drag ratios than the basic model. The modified models also had higher lift curve slopes and generally were slightly more stable than the basic configuration.
Author
Body-Wing Configurations; Unswept Wings; Lift Drag Ratio; Drag Reduction; Pitching Moments; Zero Lift

19980232004 NASA Langley Research Center, Hampton, VA USA
Free-Flight Test of a Technique for Inflating an NASA 12-Foot-Diameter Sphere at High Altitudes
Kehlet, Alan B., NASA Langley Research Center, USA; Patterson, Herbert G., NASA Langley Research Center, USA; Jan. 1959; 20p; In English
Report No.(s): NASA-MEMO-2-5-59L; L-214; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A free-flight test has been conducted to check a technique for inflating an NASA 12-foot-diameter inflatable sphere at high altitudes. Flight records indicated that the nose section was successfully separated from the booster rocket, that the sphere was ejected, and that the nose section was jettisoned from the fully inflated sphere. On the basis of preflight and flight records, it is believed that the sphere was fully inflated by the time of peak altitude (239,000 feet). Calculations showed that during descent, jettison of the nose section occurred above an altitude of 150,000 feet. The inflatable sphere was estimated to start to deform during descent at an altitude of about 120,000 feet.
Author
Inflatable Space Structures; Spheres; High Altitude; Space Erectable Structures

19980232008 NASA Langley Research Center, Hampton, VA USA
Wind-Tunnel Investigation of Some Effects of Wing Sweep and Horizontal-Tail Height on the Static Stability of an Airplane Model at Transonic Speeds
Fisher, Lewis R., NASA Langley Research Center, USA; Williams, James L., NASA Langley Research Center, USA; Oct. 1958; 46p; In English
Report No.(s): NASA-MEMO-10-3-58L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A research model of an airplane with a configuration suitable for supersonic flight was tested at transonic speeds in order to establish the effects on longitudinal and lateral stability of certain changes in both wing sweep and height of the horizontal tail. Two wings of aspect ratio 3 and taper ratio 0.15, one having the quarter-chord line swept back 30 deg and the other 45 deg, were each tested with the horizontal tail of the model in a low and in a high position. One configuration was also tested with fuselage strakes. The tests were made at Mach numbers from 0.60 to 1.17 and Reynolds numbers from 1.9 x 10(exp 6) to 2.6 x 10(exp 6). The results indicated that a low horizontal-tail position (below the wing-chord plane) gave positive longitudinal stability for the model for all angles of attack used (angles of attack up to 24 deg); whereas, a higher tail position (above the wing-chord plane) resulted in a large reduction in stability at moderate angles of attack. With the higher horizontal tail, the 30 deg-swept-wing model had somewhat more stability than the 45 deg-swept-wing model at subsonic Mach numbers. With the lower tail, the 45 deg-swept-wing model had slightly more stability at all Mach numbers. The model with the 30 deg swept wing had greater directional stability with the tail in the higher rather than the lower position, but the opposite was true for the 45 deg-swept-wing model. The directional stability decreased sharply at high angles of attack; this characteristic was alleviated by the use of fuselage strakes which, however, proved to be detrimental to the longitudinal stability of the model tested.
Author
Longitudinal Stability; Aircraft Models; Transonic Speed; Angle of Attack; Aspect Ratio; Directional Stability

19980232010 NASA Langley Research Center, Hampton, VA USA
Effects of Body Shape on the Drag of a 45 degree Sweptback-Wing-Body Configuration at Mach Numbers from 0.90 to 1.43
Olstad, Walter B., NASA Langley Research Center, USA; Fischetti, Thomas L., NASA Langley Research Center, USA; Nov. 1958; 66p; In English
Report No.(s): NASA-MEMO-10-23-58L; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

An investigation was made of the effects of body shape on the drag of a 45 deg sweptback-wing-body combination at Mach numbers from 0.90 to 1.43. Both the expansion and compression fields induced by body indentation were swept back as the stream Mach number increased from 0.94. The line of zero pressure change was generally tangent to the Mach lines associated with the local velocities over the wing and body. The strength of the induced pressure fields over the wing were attenuated with spanwise distance and the major effects were limited to the inboard 60 percent of the wing semispan. Asymmetrical body indentation tended to increase the lift on the forward portion of the wing and reduce the lift on the rearward portion. This redistribution of lift had a favorable effect on the wave drag due to lift. Symmetrical body indentation reduced the drag loading near the wing-body juncture at all Mach numbers. The reduction in drag loading increased in spanwise extent as the Mach number increased and the line of zero induced pressure became more nearly aligned with the line of maximum wing thickness. Calculations of the wave drag due to thickness, the wave drag due to lift, and the vortex drag of the basic and symmetrical M = 1.2 body and wing combinations at an angle of attack of 0 deg predicted the effects of indentation within 11 percent of the wing-basic-body drag throughout the Mach number range from 1.0 to 1.43. Calculations of the wave drag due to thickness, the wave drag due to lift, and the vortex drag for the basic, symmetrical M = 1.2, and asymmetrical M = 1.4 body and wing combinations predicted the total pressure drag to within 8 percent of the experimental value at M = 1.43.
Author
Sweptback Wings; Drag Reduction; Body-Wing Configurations; Pressure Drag; Wave Drag

19980232017 NASA Langley Research Center, Hampton, VA USA
Overview of Laminar Flow Control
Joslin, Ronald D., NASA Langley Research Center, USA; Oct. 1998; 142p; In English
Contract(s)/Grant(s): RTOP 538-05-15-01
Report No.(s): NASA/TP-1998-208705; L-17631; NAS 1.60:208705; No Copyright; Avail: CASI; A07, Hardcopy; A02, Microfiche

The history of Laminar Flow Control (LFC) from the 1930s through the 1990s is reviewed and the current status of the technology is assessed. Early studies related to the natural laminar boundary-layer flow physics, manufacturing tolerances for laminar flow, and insect-contamination avoidance are discussed. Although most of this publication is about slot-, porous-, and

perforated-suction LFC concept studies in wind tunnel and flight experiments, some mention is made of thermal LFC. Theoretical and computational tools to describe the LFC aerodynamics are included for completeness.
Author
Laminar Flow; Boundary Layer Control; Laminar Boundary Layer; Flow Characteristics; Histories; General Overviews

19980232050 NASA Langley Research Center, Hampton, VA USA
Basic Pressure Measurements at Transonic Speeds on a Thin 45 deg Sweptback Highly Tapered Wing with Systematic Spanwise Twist Variations
Mugler, John P., Jr., NASA Langley Research Center, USA; Dec. 1958; 98p; In English
Report No.(s): NASA-MEMO-10-20-58L; No Copyright; Avail: CASI; A05, Hardcopy; A02, Microfiche
Pressure distributions are presented for a thin highly tapered untwisted 45 deg sweptback wing in combination with a body. These tests were made in the Langley 8-foot transonic pressure tunnel at both 1.0 and 0.5 atmosphere stagnation pressures at Mach numbers from 0.800 to 1.200 through an angle-of-attack range of -4 deg to 12 deg.
Author
Pressure Measurement; Wind Tunnel Tests; Pressure Distribution; Sweptback Wings

19980232076 NASA Ames Research Center, Moffett Field, CA USA
Supersonic and Moment-of-Area Rules Combined for Rapid Zero-Lift Wave-Drag Calculations
Levy, Lionel L., Jr., NASA Ames Research Center, USA; Jun. 1959; 72p; In English
Report No.(s): NASA-MEMO-4-19-59A; A-158; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche
The concepts of the supersonic area rule and the moment-of-area rule are combined to develop a new method for calculating zero-lift wave drag which is amenable to the use of ordinary desk calculators. The total zero-lift wave drag of a configuration is calculated by the new method as the sum of the wave drag of each component alone plus the interference between components. In calculating the separate contributions each component or pair of components is analyzed over the smallest allowable length in order to improve the convergence of the series expression for the wave drag. The accuracy of the present method is evaluated by comparing the total zero-lift wave-drag solutions for several simplified configurations obtained by the present method with solutions given by slender-body and linearized theory. The accuracy and computational time required by the present method are also evaluated relative to the supersonic area rule and the moment-of-area rule. The results of the evaluation indicate that total zero-lift wave-drag solutions for simplified configurations can be obtained by the present method which differ from solutions given by slender-body and linearized theory by less than 6 percent. This accuracy for simplified configurations was obtained from only nine terms of the series expression for the wave drag as a result of calculating the total zero-lift wave drag by parts. For the same number of terms these results represent an accuracy greater than that for solutions obtained by either of the two methods upon which the present method is based, except in a few isolated cases. For the excepted cases, solutions by the present method and the supersonic area rule are identical. Solutions by the present method are obtained in one fifth the computing time required by the supersonic area rule. This difference in computing time of course would be substantially reduced if the complete procedures for both methods were programmed on electronic computing machines.
Author
Wave Drag; Zero Lift; Drag Measurement; Moments

19980232082 NASA Langley Research Center, Hampton, VA USA
Semiempirical Procedure for Estimating Lift and Drag Characteristics of Propeller-Wing-Flap Configurations for Vertical-and Short-Take-Off-and-Landing Airplanes
Kuhn, Richard E., NASA Langley Research Center, USA; Feb. 1959; 38p; In English
Report No.(s): NASA-MEMO-1-16-59L; L-144; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche
The analysis presented uses the momentum theory as a starting point in developing semiempirical expressions for calculating the effect of propeller thrust and slipstream on the lift and drag characteristics of wing-flap configurations that would be suitable for vertical-take-off-and-landing (VTOL) and short-take-off-and-landing (STOL) airplanes. The method uses power-off forward-speed information and measured slipstream deflection data at zero forward speed to provide a basis for estimating the lift and drag at combined forward speed and power-on conditions. A correlation of slipstream deflection data is also included. The procedure is applicable only in the unstalled flight regime; nevertheless, it should be useful in preliminary design estimates of the performance that may be expected of VTOL and STOL airplanes.
Author
Lift; Drag; Vertical Takeoff Aircraft; Short Takeoff Aircraft; Propellers; Wing Flaps

19980232085 NASA Langley Research Center, Hampton, VA USA

Transonic Aerodynamic Characteristics of a 45 deg Swept Wing Fuselage Model with a Finned and Unfinned Body Pylon Mounted Beneath the Fuselage or Wing, Including Measurements of Body Loads

Wornom, Dewey E., NASA Langley Research Center, USA; May 1959; 60p; In English

Report No.(s): NASA-MEMO-4-20-59L; L-206; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

An investigation of a model of a standard size body in combination with a representative 45 deg swept-wing-fuselage model has been conducted in the Langley 8-foot transonic pressure tunnel over a Mach number range from 0.80 to 1.43. The body, with a fineness ratio of 8.5, was tested with and without fins, and was pylon-mounted beneath the fuselage or wing. Force measurements were obtained on the wing-fuselage model with and without the body, for an angle-of-attack range from -2 deg to approximately 12 deg and an angle-of-sideslip range from -8 deg to 8 deg. In addition, body loads were measured over the same angle-of-attack and angle-of-sideslip range. The Reynolds number for the investigation, based on the wing mean aerodynamic chord, varied from 1.85 x 10(exp 6) to 2.85 x 10(exp 6). The addition of the body beneath the fuselage or the wing increased the drag coefficient of the complete model over the Mach number range tested. On the basis of the drag increase per body, the under-fuselage position was the more favorable. Furthermore, the bodies tended to increase the lateral stability of the complete model. The variation of body loads with angle of attack for the unfinned bodies was generally small and linear over the Mach number range tested with the addition of fins causing large increases in the rates of change of normal-force coefficient and nose-down pitching-moment coefficient. The variation of body side-force coefficient with sideslip for the unfinned body beneath the fuselage was at least twice as large as the variation of this load for the unfinned body beneath the wing. The addition of fins to the body beneath either the fuselage or the wing approximately doubled the rate of change of body side-force coefficient with sideslip. Furthermore, the variation of body side-force coefficient with sideslip for the body beneath the wing was at least twice as large as the variation of this load with angle of attack.

Author

Aerodynamic Characteristics; Swept Wings; Fuselages; Finned Bodies; Pylons; Loads (Forces); Aerodynamic Drag; Aerodynamic Coefficients

19980232100 Nanjing Univ. of Aeronautics and Astronautics, Nanjing, Jiangsu, China

Effect of Blowing on Strake-Wing Vortices During Dynamic Pitching

Da, Huang, Nanjing Univ. of Aeronautics and Astronautics, Nanjing, China; Genxing, Wu, Nanjing Univ. of Aeronautics and Astronautics, Nanjing, China; Journal of Nanjing University of Aeronautics and Astronautics; Jun. 1997; ISSN 1005-2615; Volume 29, No. 3, pp. 321-325; In Chinese; No Copyright; Avail: Issuing Activity, Hardcopy, Microfiche

A flow-visualization investigation on leading-edge breakdown is conducted in a wind tunnel to identify the effect of blowing on a strake-wing model undergoing large amplitude pitching motions. During these experiments, the visualization of the leading-edge vortices is obtained by injecting smoke. The location of vortex breakdown is recorded by use of the phase-locked photography technique. Results indicate that blowing delays the burst of leading-edge vortices during pitching up and is favorable for producing and developing the leading-edge vortices during pitching down.

Author

Blowing; Strakes; Wings; Vortices; Wind Tunnels; Unsteady Aerodynamics

19980232225 NASA Ames Research Center, Moffett Field, CA USA

Effects of Sting-Support Diameter on the Base Pressures of an Elliptic Cone at Mach Numbers from 0.60 to 1.40

Stivers, Louis S., Jr., NASA Ames Research Center, USA; Levy, Lionel L., Jr., NASA Ames Research Center, USA; Feb. 1961; 34p; In English

Report No.(s): NASA-TN-D-354; A-432; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Measurements were made to determine the effects of sting-support diameter on the base pressures of an elliptic cone with ratio of cross-section thickness to width of 1/3 and a plan-form, semi-apex angle of 15 deg. The investigation was made for model angles of attack from -2 deg to +20 deg at Mach numbers from 0.60 to 1.40, and for a constant Reynolds number of 1.4 million, based on the length of the model. The results indicated that the sting interference decreased the base axial-force coefficients by substantial amounts up to a maximum of about one-third the value of the coefficient for no sting interference. There was no practical diameter of the sting for which the effects of the sting on the base pressures would be negligible throughout the Mach number and angle-of-attack ranges of the investigation.

Author

Cones; Base Pressure; Angle of Attack; Thickness

19980232228 NASA Langley Research Center, Hampton, VA USA

Distribution of Heat Transfer on a 10 deg Cone at Angles of Attack from 0 deg to 15 deg for Mach Numbers of 2.49 to 4.65 and a Solution to the Heat-Transfer Equation that Permits Complete Machine Calculations

Burbank, Paige B., NASA Langley Research Center, USA; Hodge, B. Leon, NASA Langley Research Center, USA; Jun. 1959; 50p; In English

Report No.(s): NASA-MEMO-6-4-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The pressure and heat-transfer distribution were measured on the surface of a thin-walled 10 deg cone for a Mach number range from 2.49 to 4.65 at angles of attack from 0 deg to 15 deg in the Langley Unitary Plan wind tunnel.The results indicate that Kopal's theory adequately predicts the surface Mach number for heat-transfer calculations. The measured laminar heat-transfer coefficients at at an angle of attack of 0 deg are in good agreement with Van Driest theory having a Mangler transformation. At an angle of attack the heat-transfer coefficient along the stagnation line is 1.9 to 4 times greater than at an angle of attack of 0 deg depending upon the the distance from the tip of the nose, Reynolds number and Mach number. Boundary-layer transition and body vortices caused minimum heat-transfer coefficients to occur at the 90 deg to 120 deg meridian angles and increased aerodynamic heating along the 180 deg meridian that in some cases is of the same magnitude as that along the zero meridian (stagnation line). A method was developed for complete machine calculation of the heat-transfer coefficient from transient temperature measurements.

Author

Heat Transfer Coefficients; Aerodynamic Heating; Boundary Layer Transition; Heat Transfer; Pressure Distribution

19980232231 NASA Langley Research Center, Hampton, VA USA

Tables for the Rapid Estimation of Downwash and Sidewash Behind Wings Performing Various Motions at Supersonic Speeds

Bobbitt, Percy J., NASA Langley Research Center, USA; May 1959; 180p; In English

Report No.(s): NASA-MEMO-2-20-59L; No Copyright; Avail: CASI; A09, Hardcopy; A02, Microfiche

Equations for the downwash and sidewash due to supersonic yawed and unswept horseshoe vortices have been utilized in formulating tables and charts to permit a rapid estimation of the flow velocities behind wings performing various steady motions. Tabulations are presented of the downwash and sidewash in the wing vertical plane of symmetry due to a unit-strength yawed horseshoe vortex located at 20 equally spaced spanwise positions along lifting lines of various sweeps. (The bound portion of the yawed vortex is coincident with the lifting line.) Charts are presented for the purpose of estimating the spanwise variations of the flow-field velocities and give longitudinal variations of the downwash and sidewash at a nuMber of vertical and spanwise locations due to a unit-strength unswept horseshoe vortex. Use of the tables and charts to calculate wing downwash or sidewash requires a knowledge of the wing spanwise distribution of circulation. Sample computations for the rolling sidewash and angle-of-attack downwash behind a typical swept wing are presented to demonstrate the use of the tables and charts.

Author

Downwash; Swept Wings; Supersonic Speed; Flow Velocity; Estimating; Backwash

19980232232 NASA Langley Research Center, Hampton, VA USA

Free-Flight Investigation of a Rocket-Propelled Model to Determine the Aerodynamic Heating on a Thin, Unswept, Untapered, Multispar, Aluminum-Alloy Wing at Mach Numbers up to 2.22

Stephens, Emily W., NASA Langley Research Center, USA; Jan. 1959; 42p; In English

Report No.(s): NASA-MEMO-12-15-58L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A free-flight investigation has been made to determine some effects of aerodynamic heating on the structural behavior of a wing at supersonic speeds. The test wing was a thin, unswept, untapered, multispar, aluminum-alloy wing having a 20-inch chord, a 20-inch exposed semispan, and a circular-arc airfoil section with a thickness ratio of 5 percent. The wing was tested on a model propelled by a two-stage rocket-propulsion system to a Mach number of 2.22 and a corresponding Reynolds number per foot of 13.2 x 10(6) Reasonably good agreement was obtained between Stanton numbers obtained from measured temperature-time data and values obtained by the theory of Van Driest for flat plates having turbulent boundary layers. Temperature measurements made in the skin of the wing and in the internal structures agreed well with calculated values. The wing was instrumented to detect any apparent fluttering motion in the wing, but no evidence of flutter was observed throughout the flight.

Author

Aerodynamic Heating; Unswept Wings; Free Flight; Turbulent Boundary Layer; Airfoil Profiles; Aluminum Alloys; Rocket Engines

19980232234 NASA Langley Research Center, Hampton, VA USA

Some Effects of Horizontal-Tail Position on the Vertical-Tail Pressure Distributions of a Complete Model in Sideslip at High Subsonic Speeds

Alford, William J., Jr., NASA Langley Research Center, USA; Oct. 1958; 156p; In English

Report No.(s): NASA-MEMO-10-5-58L; No Copyright; Avail: CASI; A08, Hardcopy; A02, Microfiche

An investigation has been made in the Langley high-speed 7- by 10-foot tunnel of some effects of horizontal-tail position on the vertical-tail pressure distributions of a complete model in sideslip at high subsonic speeds. The wing of the model was swept back 28.82 deg at the quarter-chord line and had an aspect ratio of 3.50, a taper ratio of 0.067, and NACA 65A004 airfoil sections parallel to the model plane of symmetry. Tests were made with the horizontal tail off, on the wing-chord plane extended, and in T-tail arrangements in forward and rearward locations. The test Mach numbers ranged from 0.60 to 0.92, which corresponds to a Reynolds number range from approximately 2.93 x 10(exp 6) to 3.69 x 10(exp 6), based on the wing mean aerodynamic chord. The sideslip angles varied from -3.9 deg to 12.7 deg at several selected angles of attack. The results indicated that, for a given angle of sideslip, increases in angle of attack caused reductions in the vertical-tail loads in the vicinity of the root chord and increases at the midspan and tip locations, with rearward movements in the local chordwise centers of pressure for the midspan locations and forward movements near the tip of the vertical tail. At the higher angles of attack all configurations investigated experienced outboard and rearward shifts in the center of pressure of the total vertical-tail load. Location of the horizontal tail on the wing- chord plane extended produced only small effects on the vertical-tail loads and centers of pressure. Locating the horizontal tail at the tip of the vertical tail in the forward position caused increases in the vertical-tail loads; this configuration, however, experienced considerable reduction in loads with increasing Mach number. Location of the horizontal tail at the tip of the vertical tail in the rearward position produced the largest increases in vertical-tail loads per degree sideslip angle; this configuration experienced the smallest variations of loads with Mach number of any of the configurations investigated.

Author

Tail Assemblies; Sideslip; Airfoil Profiles; Center of Pressure; Pressure Distribution

19980232606 NASA Lewis Research Center, Cleveland, OH USA

Excitation of Continuous and Discrete Modes in Incompressible Boundary Layers

Ashpis, David E., NASA Lewis Research Center, USA; Reshotko, Eli, Case Western Reserve Univ., USA; Jul. 1998; 56p; In English

Contract(s)/Grant(s): RTOP 522-31-23

Report No.(s): NASA/TM-1998-208490; E-11271; NAS 1.15:208490; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

This report documents the full details of the condensed journal article by Ashpis & Reshotko (JFM, 1990) entitled "The Vibrating Ribbon Problem Revisited." A revised formal solution of the vibrating ribbon problem of hydrodynamic stability is presented. The initial formulation of Gaster (JFM, 1965) is modified by application of the Briggs method and a careful treatment of the complex double Fourier transform inversions. Expressions are obtained in a natural way for the discrete spectrum as well as for the four branches of the continuous spectra. These correspond to discrete and branch-cut singularities in the complex wave-number plane. The solutions from the continuous spectra decay both upstream and downstream of the ribbon, with the decay in the upstream direction being much more rapid than that in the downstream direction. Comments and clarification of related prior work are made.

Author

Incompressible Boundary Layer; Fourier Transformation; Flow Stability; Hydrodynamic Coefficients; Vibration; Ribbons

19980232906 NASA Ames Research Center, Moffett Field, CA USA

Investigation at Mach Numbers of 0.20 to 3.50 of a Blended Diamond Wing and Body Combination of Sonic Design but with Low Wave-Drag Increase with Increasing Mach Number

Holdaway, George H., NASA Ames Research Center, USA; Mellenthin, Jack A., NASA Ames Research Center, USA; Hatfield, Elaine W., NASA Ames Research Center, USA; Oct. 1959; 56p; In English

Report No.(s): NASA-TM-X-105; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

A diamond wing and body combination was designed to have an area distribution which would result in near optimum zero-lift wave-drag coefficients at a Mach number of 1.00, and decreasing wave-drag coefficient with increasing Mach number up to near sonic leading-edge conditions for the wing. The airfoil section were computed by varying their shape along with the body radii (blending process) to match the selected area distribution and the given plan form. The exposed wing section had an average maximum thickness of about 3 percent of the local chords, and the maximum thickness of the center-line chord was 5.49 percent. The wing had an aspect ratio of 2 and a leading-edge sweep of 45 deg. Test data were obtained throughout the Mach number range

from 0.20 to 3.50 at Reynolds numbers based on the mean aerodynamic chord of roughly 6,000,000 to 9,000,000. The zero-lift wave-drag coefficients of the diamond model satisfied the design objectives and were equal to the low values for the Mach number 1.00 equivalent body up to the limit of the transonic tests. From the peak drag coefficient near M = 1.00 there was a gradual decrease in wave-drag coefficient up to M = 1.20. Above sonic leading-edge conditions of the wing there was a rise in the wave-drag coefficient which was attributed in part to the body contouring as well as to the wing geometry. The diamond model had good lift characteristics, in spite of the prediction from low-aspect-ratio theory that the rear half of the diamond wing would carry little lift. The experimental lift-curve slope obtained at supersonic speeds were equal to or greater than the values predicted by linear theory. Similarly the other basic aerodynamic parameters, aerodynamic center position, and maximum lift-drag ratios were satisfactorily predicted at supersonic speeds.
Derived from text
Airfoil Profiles; Aerodynamic Drag; Wings; Mach Number; Wave Drag; Swept Wings; Aspect Ratio

19980233218 NASA Lewis Research Center, Cleveland, OH USA
Wave Turbine Analysis Tool Development
Welch, Gerard E., Army Research Lab., USA; Paxson, Daniel E., NASA Lewis Research Center, USA; Jul. 1998; 16p; In English; 34th; Joint Propulsion Conference, 12-15 Jul. 1998, Cleveland, OH, USA; Sponsored by American Inst. of Aeronautics and Astronautics, USA
Contract(s)/Grant(s): RTOP 523-26-33; DA Proj. 1L1-61102-AH-45
Report No.(s): NASA/TM-1998-208485; NAs 1.15:208485; E-11261; ARL-TR-1740; AIAA Paper 98-3402; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A quasi-one-dimensional (Q-1-D) computational fluid dynamic solver, previously developed and validated for pressure-exchanger wave rotors, is extended in the present work to include the blade forces of power producing wave rotors (i.e., wave turbines). The accuracy of the single-passage Q-1-D solver is assessed relative to two two-dimensional solvers: a single-passage code and a multi-block stator/rotor/stator code. Comparisons of computed results for inviscid, steady and unsteady flows in passage geometries typical of wave rotors reveal that the blade force model is accurate and that the correlation (effective stress and heat flux) terms of the Q-1-D passage-averaged formulation can be neglected. The ends of the rotor passages pose particular challenges to Q-1-D formulations because the flow there must at times deviate significantly from the mean camber line angle to match the port flow fields. This problem is most acute during the opening and closing of the rotor passages. An example sub-model is developed to account for the deviation between the flow departure angle and the mean camber line exit angle that occurs as an inviscid flow decelerates to meet a uniform pressure boundary. Comparisons of results from four-port wave turbine simulations reveal that the Q-1-D solver currently overpredicts wave turbine performance levels and highlight the need to devote future effort to the boundary conditions and sub-models of the Q-1-D solver.
Author
Computational Fluid Dynamics; Wave Rotors; Turbines; Stators; Rotors; Elastic Waves

19980233249 United Technologies Research Center, Aeromechanical, Chemical and Fluid Systems, East Hartford, CT USA
A Numerical Simulator for Three-Dimensional Flows Through Vibrating Blade Rows *Final Report*
Chuang, H. Andrew, United Technologies Research Center, USA; Verdon, Joseph M., United Technologies Research Center, U-SA; Aug. 1998; 78p; In English
Contract(s)/Grant(s): NAS3-26618; RTOP 538-03-11
Report No.(s): NASA/CR-1998-208511; E-11283; NAS 1.26:208511; R98-4.101.0238; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

The three-dimensional, multi-stage, unsteady, turbomachinery analysis, TURBO, has been extended to predict the aeroelastic and aeroacoustic response behaviors of a single blade row operating within a cylindrical annular duct. In particular, a blade vibration capability has been incorporated so that the TURBO analysis can be applied over a solution domain that deforms with a vibratory blade motion. Also, unsteady far-field conditions have been implemented to render the computational boundaries at inlet and exit transparent to outgoing unsteady disturbances. The modified TURBO analysis is applied herein to predict unsteady subsonic and transonic flows. The intent is to partially validate this nonlinear analysis for blade flutter applications, via numerical results for benchmark unsteady flows, and to demonstrate the analysis for a realistic fan rotor. For these purposes, we have considered unsteady subsonic flows through a 3D version of the 10th Standard Cascade, and unsteady transonic flows through the first stage rotor of the NASA Lewis, Rotor 67, two-stage fan.
Author
Aeroacoustics; Aeroelasticity; Annular Ducts; Rotors; Subsonic Flow; Three Dimensional Flow; Transonic Flow; Turbomachinery

19980235513 NASA Langley Research Center, Hampton, VA USA
Measurements of Aerodynamic Heat Transfer and Boundary-Layer Transition on a 15 deg. Cone in Free Flight at Supersonic Mach Numbers up to 5.2
Rumsey, Charles B., NASA Langley Research Center, USA; Lee, Dorothy B., NASA Langley Research Center, USA; Aug. 1961; 48p; In English
Report No.(s): NASA-TN-D-888; L-1640; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Measurements of aerodynamic heat transfer have been made at several stations on the 15 deg total-angle conical nose of a rocket-propelled model in free flight at Mach numbers up to 5.2. Data are presented for a range of local Mach number just outside the boundary layer from 1.40 to 4.65 and a range of local Reynolds number from 3.8 x 10(exp 6) to 46.5 x 10(exp 6), based on length from the nose tip to a measurement station. Laminar, transitional, and turbulent heat-transfer coefficients were measured. The laminar data were in agreement with laminar theory for cones, and the turbulent data agreed well with turbulent theory for cones using Reynolds number based on length from the nose tip. At a nearly constant ratio of wall to local static temperature of 1.2 the Reynolds number of transition increased from 14 x 10(exp 6) to 30 x 10(exp 6) as Mach number increased from 1.4 to 2.9 and then decreased to 17 x 10(exp 6) as Mach number increased to 3.7. At Mach numbers near 3.5, transition Reynolds numbers appeared to be independent of skin temperature at skin temperatures very cold with respect to adiabatic wall temperature. The transition Reynolds number was 17.7 x 10(exp 6) at a condition of Mach number and ratio of wall to local static temperature near that for which three-dimensional disturbance theory has been evaluated and has predicted laminar boundary-layer stability to very high Reynolds numbers (approximately 10(exp 12)).
Author
Aerodynamic Heat Transfer; Heat Transfer Coefficients; Perturbation Theory; Laminar Boundary Layer; Boundary Layer Transition; Boundary Layer Stability

19980235518 NASA Ames Research Center, Moffett Field, CA USA
Investigation at Mach Numbers of 0.20 to 3.50 of Blended Wing-Body Combinations of Sonic Design with Diamond, Delta, and Arrow Plan Forms
Holdaway, George H., NASA Ames Research Center, USA; Mellenthin, Jack A., NASA Ames Research Center, USA; Aug. 1960; 80p; In English
Report No.(s): NASA-TM-X-372; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

The models had aspect-ratio-2 diamond, delta, and arrow wings with the leading edges swept 45.00 deg, 59.04 deg, and 70.82 deg, respectively. The wing sections were computed by varying the section shape along with the body radii (blending process) to match the prescribed area distribution and wing plan form. The wing sections had an average value of maximum thickness ratio of about 4 percent of the local chords in a streamwise direction. The models were tested with transition fixed at Reynolds numbers of about 4,000,000 to 9,000,0000, based on the mean aerodynamic chord of the wings. The effect of varying Reynolds number was checked at both subsonic and supersonic speeds. The diamond model was superior to the other plan forms at transonic speeds ((L/D)max = 11.00 to 9.52) because of its higher lift-curve slope and near optimum wave drag due to the blending process. For the wing thickness tested with the diamond model, the marked body and wing contouring required for transonic conditions resulted in a large wave-drag penalty at the higher supersonic Mach numbers where the leading and trailing edges of the wing were supersonic. Because of the low sweep of the trailing edge of the delta model, this configuration was less adaptable to the blending process. Removing a body bump prescribed by the Mach number 1.00 design resulted in a good supersonic design. This delta model with 10 percent less volume was superior to the other plan forms at Mach numbers of 1.55 to 2.35 ((L/D)max = 8.65 to 7.24), but it and the arrow model were equally good at Mach numbers of 2.50 to 3.50 ((L/D)max - 6.85 to O.39). At transonic speeds the arrow model was inferior because of the reduced lift-curve slope associated with its increased sweep and also because of the wing base drag. The wing base-drag coefficients of the arrow model based on the wing planform area decreased from a peak value of 0.0029 at Mach number 1.55 to 0.0003 at Mach number 3.50. Linear supersonic theory was satisfactory for predicting the aerodynamic trends at Mach numbers from 1.55 to 3.50 of lift-curve slope, wave drag, drag due to lift, aerodynamic-center location, and maximum lift-drag ratios for each of the models.
Author
Body-Wing Configurations; Leading Edges; Aerodynamic Drag; Airfoil Profiles; Mach Number; Delta Wings; Low Aspect Ratio Wings; Lift Drag Ratio

19980235519 NASA Langley Research Center, Hampton, VA USA
Boundary-Induced Downwash Due to Lift in a Two-Dimensional Slotted Wind Tunnel
Katzoff, S., NASA Langley Research Center, USA; Barger, Raymond L., NASA Langley Research Center, USA; 1959; 24p; In English

Report No.(s): NASA-TR-R-25; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A solution has been obtained for the complete tunnel-interference flow for a lifting vortex in a two-dimensional slotted tunnel. Curves are presented for the longitudinal distribution of tunnel-induced downwash angle for various values of the boundary openness parameter and for various heights of the vortex above the tunnel center line. Some quantitative discussion is given of the use of these results in calculating the tunnel interference for three-dimensional wings in rectangular tunnels with closed side walls and slotted top and bottom.

Author

Downwash; Slotted Wind Tunnels; Boundaries

19980235539 NASA Dryden Flight Research Center, Edwards, CA USA

Determination of Sun Angles for Observations of Shock Waves on a Transport Aircraft

Fisher, David F., NASA Dryden Flight Research Center, USA; Haering, Edward A., Jr., NASA Dryden Flight Research Center, USA; Noffz, Gregory K., NASA Dryden Flight Research Center, USA; Aguilar, Juan I., NASA Dryden Flight Research Center, USA; Sep. 1998; 18p; In English; 8th; Flow Visualization, 1-4 Sep. 1998, Sorrento, Italy

Contract(s)/Grant(s): RTOP 529-59-04

Report No.(s): NASA/TM-1998-206551; H-2251; NAS 1.15:206551; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Wing compression shock shadowgraphs were observed on two flights during banked turns of an L-1011 aircraft at a Mach number of 0.85 and an altitude of 35,000 ft (10,700 m). Photos and video recording of the shadowgraphs were taken during the flights to document the shadowgraphs. Bright sunlight on the aircraft was required. The time of day, aircraft position, speed and attitudes were recorded to determine the sun azimuth and elevation relative to the wing quarter chord-line when the shadowgraphs were visible. Sun elevation and azimuth angles were documented for which the wing compression shock shadowgraphs were visible. The shadowgraph was observed for high to low elevation angles relative to the wing, but for best results high sun angles relative to the wing are desired. The procedures and equations to determine the sun azimuth and elevation angle with respect to the quarter chord-line is included in the Appendix.

Author

Wings; Shadowgraph Photography; L-1011 Aircraft; Elevation Angle; Mach Number

19980236156 Defence Science and Technology Organisation, Information Technology Div., Canberra Australia

Maneuver Controller Design for an F-111C Flight Dynamics Model

Gibbens, Peter W., Defence Science and Technology Organisation, Australia; May 1998; 129p; In English

Report No.(s): AD-A352580; DSTO-RR-0129; DODA-AR-010-504; No Copyright; Avail: CASI; A07, Hardcopy; A02, Microfiche

A maneuver controller program has been developed to fly an F-111C dynamic flight model through any number of prescribed maneuvers. A selection of discrete maneuvers is available which can be used as building blocks to represent most of those likely to be encountered in flight. Generalized maneuvers can also be flown by providing reference flight trajectories generated by an external source. The dynamic model and maneuver controller have been developed to allow the realistic modeling of maneuvers required by mission analyses, weapons delivery studies and systems assessments.

DTIC

Dynamic Models; Controllers; Maneuvers; Automatic Pilots; Control Theory; F-111 Aircraft; Weapons Delivery

19980236568 Advisory Group for Aerospace Research and Development, Neuilly-Sur-Seine, France

Wind Tunnel Wall Corrections ***La Correction des Effets de Paroi en Soufflerie***

Ewald, B. F. R., Editor, Technische Univ., Germany; Oct. 1998; 560p; In English

Report No.(s): AGARD-AG-336; ISBN 92-836-1076-8; Copyright Waived; Avail: CASI; A24, Hardcopy; A04, Microfiche

This AGARDograph has been compiled by an international team of wind tunnel wall correction experts. The state of the art in wall corrections is presented with special emphasis given to the description of modern methods based on Computational Fluid Dynamics (CFD). Topics covered include: Open Test Sections, Closed Test Sections, Ventilated Test Sections, Boundary Measurement Methods, Transonic Wall Interference, Bluff Body Corrections, Adaptive Walls, Panel Methods, and CFD Methods.

Author

Wall Flow; Boundary Layer Flow; Computational Fluid Dynamics; Boundary Conditions; Aerodynamic Interference; Wind Tunnel Walls; Correction

19980236752 NASA Langley Research Center, Hampton, VA USA

An Operational Wake Vortex Sensor Using Pulsed Coherent Lidar

Barker, Ben C., Jr., NASA Langley Research Center, USA; Koch, Grady J., NASA Langley Research Center, USA; Nguyen, D. Chi, Research Triangle Inst., USA; Nineteenth International Laser Radar Conference; Jul. 1998, Part 2, pp. 681-684; In English; Also announced as 19980236718; No Copyright; Avail: CASI; A01, Hardcopy; A04, Microfiche

NASA and FAA initiated a program in 1994 to develop methods of setting spacings for landing aircraft by incorporating information on the real-time behavior of aircraft wake vortices. The current wake separation standards were developed in the 1970's when there was relatively light airport traffic and a logical break point by which to categorize aircraft. Today's continuum of aircraft sizes and increased airport packing densities have created a need for re-evaluation of wake separation standards. The goals of this effort are to ensure that separation standards are adequate for safety and to reduce aircraft spacing for higher airport capacity. of particular interest are the different requirements for landing under visual flight conditions and instrument flight conditions. Over the years, greater spacings have been established for instrument flight than are allowed for visual flight conditions. Preliminary studies indicate that the airline industry would save considerable money and incur fewer passenger delays if a dynamic spacing system could reduce separations at major hubs during inclement weather to the levels routinely achieved under visual flight conditions. The sensor described herein may become part of this dynamic spacing system known as the "Aircraft VOrtex Spacing System" (AVOSS) that will interface with a future air traffic control system. AVOSS will use vortex behavioral models and short-term weather prediction models in order to predict vortex behavior sufficiently into the future to allow dynamic separation standards to be generated. The wake vortex sensor will periodically provide data to validate AVOSS predictions. Feasibility of measuring wake vortices using a lidar was first demonstrated using a continuous wave (CW) system from NASA Marshall Space Flight Sensor and tested at the Volpe National Transportation Systems Center's wake vortex test site at JFK International Airport. Other applications of CW lidar for wake vortex measurement have been made more recently, including a system developed by the MIT Lincoln Laboratory. This lidar has been used for detailed measurements of wake vortex velocities in support of wake vortex model validation. The first measurements of wake vortices using a pulsed, lidar were made by Coherent Technologies, Inc. (CTI) using a 2 micron solid-state, flashlamp-pumped system operating at 5 Hz. This system was first deployed at Denver's Stapleton Airport. Pulsed lidar has been selected as the baseline technology for an operational sensor due to its longer range capability.

Derived from text

Aircraft Wakes; Vortices; Air Traffic Control; Optical Radar; Aircraft Approach Spacing; Radar Measurement; Remote Sensors

19980236837 NASA Langley Research Center, Hampton, VA USA

Aerodynamic Load Measurements and Opening Characteristics of Automatic Leading Edge Slats on a 45 deg Sweptback Wing at Transonic Speeds

Arabian, Donald D., NASA Langley Research Center, USA; Runckel, Jack F., NASA Langley Research Center, USA; Reid, Charles F, Jr., NASA Langley Research Center, USA; Sep. 1961; 50p; In English

Report No.(s): NASA-TN-D-900; L-1609; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Measurements of the normal force and chord force were made on the slats of a sting-mounted wing-fuselage model through a Mach number range of 0.60 to 1.03 and at angles of attack from 0 to 20 deg at subsonic speeds and from 0 to 8 deg at Mach number 1.03. The 20-percent-chord tapered leading-edge slats extended from 25 to 95 percent of the semispan and consisted of five segments. The model wing had 45 deg sweep, an aspect ratio of 3.56, a taper ratio of 0.3, and NACA 64(06)AO07 airfoil sections. Slat forces and moments were determined for the slats in the almost-closed and open positions for spanwise extents of 35 to 95 percent and 46 to 95 percent of the semispan. The results of the investigation showed little change in the slat maximum force and moment coefficients with Mach number. The coefficients for the open and almost-closed slat positions had similar variations with angle of attack. The loads on the individual slat segments were found to increase toward the tip for moderate angles of attack and decrease toward the tip for high angles of attack. An analysis of the opening and closing characteristics of aerodynamically operated slats opening on a circular-arc path is included.

Author

Aerodynamic Loads; Airfoil Profiles; Force Distribution; Sweptback Wings; Transonic Speed; Wind Tunnel Tests; Wind Tunnel Models

19980236887 NASA Ames Research Center, Moffett Field, CA USA

Experimental Determination of the Pressure Distribution on a Rectangular Wing Oscillating in the First Bending Mode for Mach Numbers from 0.24 to 1.30

Lessing, Henry C., NASA Ames Research Center, USA; Troutman, John L., NASA Ames Research Center, USA; Menees, Gene P., NASA Ames Research Center, USA; Dec. 1960; 94p; In English

Report No.(s): NASA-TN-D-344; A-354; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

The results of an experimental investigation in a wind tunnel to obtain the aerodynamic pressure distribution on an unswept rectangular wing oscillating in its first symmetrical bending mode are presented. The wing was of aspect ratio 3 with 5-percent-thick biconvex airfoil sections. Data were obtained at 0 deg and 5 deg angle of attack in the Mach number range from 0.24 to 1.30 at Reynolds numbers, depending on the mach number ranging from 2.2 to 4.6 million per foot. The reduced frequencies also a function of Mach number, ranged from 0.46 at M = 0.24 to 0.10 at M = 1.30. The most important results presented are the chordwise distributions of the surface pressures generated by the bending oscillations. Similar data obtained under static conditions are also presented. The results show that the phenomena causing irregularities in the static pressure such as three-dimensional tip effects, local shock waves, and separation will also produce significant changes in the oscillatory pressures. The experimental data are also compared with the oscillatory pressure distributions computed by means of the most complete linearized theories available. The comparison shows that subsonic linearized theory is adequate for predicting the pressures and associated phase angles at low subsonic speeds and low angles of attack for this wing. However, the appearance of local shock waves and flow separation as the Mach number and angle of attack are increased causes significant changes in the experimental data from that predicted by the theory. At the low supersonic speeds covered in the experimental investigation, linearized theory is completely inadequate, principally because of the detached bow wave caused by the wing thickness. Some indication of wind-tunnel resonance was noted; however, the effects on the experimental data appear to be confined to the M = 0.70 results.
Author
Pressure Distribution; Rectangular Wings; Wind Tunnel Tests; Wing Oscillations; Transonic Speed; Unswept Wings; Static Pressure; Separated Flow; Bow Waves; Bending

03
AIR TRANSPORTATION AND SAFETY

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

19980231963 Naval Air Warfare Center, Weapons Div., China Lake, CA USA
Mishap Data Evaluation of Current Naval Aircraft *Final Report, 1987-1996*
Hennings, Elsa J., Naval Air Warfare Center, USA; Aug. 1998; 28p; In English
Report No.(s): AD-A351582; NAWCWPSN-TP-8332; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This report documents a study of U.S. Naval mishap data from January 1987 to September 1996 involving F/A-18A,B,C,D; AV-8B/TAV-8B; F-14A,B,C,D; EA-6B; T-45; T-2; S-3; and TA-4 aircraft. The study was conducted to determine which technological improvements in aircraft and aircrew safety systems might reduce the mishap/injury/lethality rates of the subject aircraft and crewmembers. Data were first sorted into aircraft categories, including aircraft lost per year by platform and mishaps per 100,000 flight hours by year and by platform. The crewmembers involved were then sorted into injury categories. For those crewmembers in the categories of fatality, permanent total disability, permanent partial disability, and major injury, the data were sorted into eject/no eject categories and then into technology categories that may have prevented the fatality or injury, first for the entire group and then for only those crewmembers who ejected. The results of this study indicate that 268 of the aircraft listed above have been lost in the 10 years studied. These mishaps involved 416 crewmembers, 192 (46.2%) of which were killed, disabled, or received a major injury. of these 192 crewmembers, the largest number of fatalities and injuries (83 crewmembers (43.2%)) may have been eliminated if an automatic ejection system were available. One hundred twenty-three (64%) of these 192 crewmembers were known to have ejected, and of these, 41 (33.3%) may have been helped if an improved propulsion system were available. Thirty-three (26.8%) of the 123 would have been helped with an improved restraint system, while 30 (24.4%) would have been helped with an optimized parachute system. It should be noted that each crewmember may have been helped by improvements in more than one technology category; therefore the numbers do not add up to 100%.
DTIC
Accidents; Data Processing; Technology Assessment; Attack Aircraft

19980232108 Naval Air Warfare Center, Aircraft Div., Patuxent River, MD USA
A Cross-Platform Multi-Simulation Software Executive
Nichols, James; Magyar, Thomas J.; Schug, Eric C.; Jan. 1998; 9p; In English; Prepared in collaboration with Science Applications International Corp., Simulation and Research Services Div., MD.
Report No.(s): AD-A350847; No Copyright; Avail: CASI; A02, Hardcopy; A01, Microfiche

The Manned Flight Simulator (MFS) was created to provide high fidelity simulation capability and support for the Navy's fleet of aircraft. The facility contains five simulation stations, which share a common interface to the computer networks, visual image generators and actual aircraft flight hardware. Any cockpit at MFS can be used in these simulation stations using a "roll-in,

roll-out" concept to easily transfer a cockpit from one simulation station to the next. With this interface, a simulation executive was required to run the multiple simulations at any simulation station. The Controls Analysis and Simulation Test Loop Environment (CASTLE) was developed to meet this requirement. Since this initial requirement, CASTLE has greatly expanded and includes many tools for use during a real-time piloted session or for desktop development and analysis use. With new development, popularity and increased performance of computers, CASTLE now operates on a variety of platforms and operating systems using the same source code and graphical user interface (GUI). These operating systems include SGI-UNIX, HP-UX, DEC VMS and Windows95/NT.
DTIC
Computer Networks; Computerized Simulation; Flight Simulators; Graphical User Interface; Simulation

19980232157 NASA Ames Research Center, Moffett Field, CA USA
Rating the Relevance of QUORUM-Selected ASRS Incident Narratives to a "Controlled Flight into Terrain" Accident
McGreevy, Michael W., NASA Ames Research Center, USA; Statler, Irving C., NASA Ames Research Center, USA; Sep. 1998; 96p; In English
Contract(s)/Grant(s): RTOP 548-71-12
Report No.(s): NASA/TM-1998-208749; A-9812680; NAS 1.15:208749; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

An exploratory study was conducted to identify commercial aviation incidents that are relevant to a "controlled flight into terrain" (CFIT) accident using a NASA-developed text processing method. The QUORUM method was used to rate 67820 incident narratives, virtually all of the narratives in the Aviation Safety Reporting System (ASRS) database, according to their relevance to two official reports on the crash of American Airlines Flight 965 near Cali, Colombia in December 1995. For comparison with QUORUM's ratings, three experienced ASRS analysts read the reports of the crash and independently rated the relevance of the 100 narratives that were most highly rated by QUORUM, as well as 100 narratives randomly selected from the database. Eighty-four of 100 QUORUM-selected narratives were rated as relevant to the Cali accident by one or more of the analysts. The relevant incidents involved a variety of factors, including, over-reliance on automation, confusion and changes during descent/approach, terrain avoidance, and operations in foreign airspace. In addition, the QUORUM collection of incidents was found to be significantly more relevant than the random collection.
Author
Aircraft Accidents; Aircraft Accident Investigation; Commercial Aircraft; Ratings; Civil Aviation; Assessments

19980232164 General Accounting Office, Resources, Community and Economic Development Div., Washington, DC USA
Aviation Safety: FAA's Use of Emergency Orders to Revoke or Suspend Operating Certificates
Dillingham, Gerald L.; Aug. 06, 1998; 17p; In English; Testimony Before the Subcommittee on Aviation, Committee on Transportation and Infrastructure, House of Representatives.
Report No.(s): AD-A351181; GAO/T-RCED; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Since the fatal crashes of ValuJet Flight 592 in May 1996 and TWA Flight 800 in July 1996, FAA'S oversight of the aviation community and the agency's enforcement actions in response to violations have come under increased scrutiny. While some have criticized FAA for not responding swiftly or forcefully enough to safety violations, others have questioned its haste in using emergency orders to suspend or revoke the certificates that pilots, airlines, and others need to operate. At the request of Senator James M. Inhofe, we recently completed a review of FAA'S use of emergency orders during fiscal years 1990 through 1997. Our report provided information on (1) the extent to which FAA used emergency orders, (2) the ways in which changes in FAA'S policies might have affected the agency's use of emergency orders, and (3) the time needed for FAA to investigate alleged violations and issue emergency orders.
DTIC
Policies; Aircraft Safety; Flight Safety; Crashes; Commercial Aircraft

19980232165 General Accounting Office, Resources, Community and Economic Development Div., Washington, DC USA
Aviation Safety: FAA's Use of Emergency Orders to Revoke or Suspend Operating Certificates
Jul. 1998; 42p; In English; Report to the Honorable James M. Inhove, U.S. Senate.
Report No.(s): AD-A351187; GAO/RCED-98-199; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

At your request, we reviewed FAA'S use of emergency orders during fiscal years 1990 through 1997. This report provides information on: (1) the extent to which FAA used emergency orders, including data on regional variation in their use, the types of certificate holders affected, and the final outcomes of cases initiated using emergency orders; (2) the ways in which changes

in FAA'S policies might have affected the agency's use of emergency orders; and (3) the time needed for FAA to investigate alleged violations and issue emergency orders.
DTIC
Emergencies; Revenue; Civil Aviation; Flight Safety

19980233234 Lockheed Martin Engineering and Sciences Co., Langley Program Office, Hampton, VA USA
Integrated Display System for Low Visibility Landing and Surface Operations
Beskenis, Sharon Otero, Lockheed Martin Engineering and Sciences Co., USA; Green, David F., Jr., Lockheed Martin Engineering and Sciences Co., USA; Hyer, Paul V., Lockheed Martin Engineering and Sciences Co., USA; Johnson, Edward J., Jr., Lockheed Martin Engineering and Sciences Co., USA; Jul. 1998; 64p; In English
Contract(s)/Grant(s): NAS1-96014; RTOP 538-04-13-02
Report No.(s): NASA/CR-1998-208446; NAS 1.26:208446; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche
This report summarizes the software products and system architectures developed by Lockheed Martin in support of the Low Visibility Landing and Surface Operations (LVLASO) program at NASA Langley Research Center. It presents an overview of the technical aspects, capabilities, and system integration issues associated with an integrated display system (IDS) that collects, processes and presents information to an aircraft flight crew during all phases of landing, roll-out, turn-off, inbound taxi, outbound taxi and takeoff. Communications hardware, drivers, and software provide continuous real-time data at varying rates and from many different sources to the display programs for presentation on a head-down display (HDD) and/or a head-up display (HUD). An electronic moving map of the airport surface is implemented on the HDD which includes the taxi route assigned by air traffic control, a text messaging system, and surface traffic and runway status information. Typical HUD symbology for navigation and control of the aircraft is augmented to provide aircraft deceleration guidance after touchdown to a pilot selected exit and taxi guidance along the route assigned by ATC. HUD displays include scene-linked symbolic runways, runway exits and taxiways that are conformal with the actual locations on the airport surface. Display formats, system architectures, and the various IDS programs are discussed.
Author
Aircraft Guidance; Real Time Operation; Display Devices; Head-Up Displays; Systems Integration; Communication Equipment; Low Visibility; Aircraft Landing; Airfield Surface Movements; All-Weather Landing Systems; Landing Instruments

19980234599 Kurtz Labs., Yellow Springs, OH USA
Helicopter Neutralization Performance Testing Protective Forces
Jun. 1998; 17p; In English
Report No.(s): AD-A351531; No Copyright; Avail: Issuing Activity (Defense Technical Information Center (DTIC)), Hardcopy, Microfiche
This paper describes the development of performance testing which examined the ability of security forces to detect and neutralize helicopter threats. It describes the application of laser engagement simulation technology to provide data regarding protective force effectiveness. Guidelines and lessons-learned are presented which offer a means of replicating such testing at other facilities which may face helicopter threats.
DTIC
Aircraft Detection; Helicopters; Helicopter Performance; Laser Applications; Systems Simulation

19980236502 National Transportation Safety Board, Washington, DC USA
Aircraft Accident Report; Uncontrolled Impact with Terrain, Fine Airlines Flight 101, Douglas DC-8-61, N27UA, Miami, Florida, August 7, 1997
Jun. 16, 1998; 155p; In English
Report No.(s): AD-A352539; NTSB/AAR-98/02; PB98-910402; No Copyright; Avail: CASI; A08, Hardcopy; A02, Microfiche
This report explains the accident involving Fine Airlines flight 101, a Douglas DC-8-61, which crashed after takeoff from runway 27R at Miami International Airport, Miami, Florida, on August 7, 1997. Safety issues in the report include the effects of improper cargo loading on airplane performance and handling, operator oversight of cargo loading and training of cargo loading personnel, the loss of critical flight data recorder information, and FAA surveillance of cargo carrier operations.
DTIC
Aircraft Accident Investigation; Commercial Aircraft; Aircraft Accidents; Aircraft Safety; Douglas Aircraft

19980236937 General Accounting Office, Resources, Community and Economic Development Div., Washington, DC USA
Aviation Safety: FAA's Use of Emergency Orders to Revoke or Suspend Operating Certificates
Jul. 1998; 48p; In English
Report No.(s): PB98-169667; GAO/RCED-98-199; B-279496; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Since the fatal crashes of ValueJet Flight 592 in May 1996 and TWA Flight 800 in July 1996, FAA's oversight of the aviation community and the agency's enforcement actions in response to violations have come under increases scrutiny. While some have criticized FAA for not responding swiftly or forcefully enough to safety violations, others have questioned its haste in using emergency orders to suspend or revoke the certificates that pilots, airlines, and others need to operate. At your request, we reviewed FAA's use of emergency orders during fiscal years 1990 through 1997. This report provides information on (1) the extent to which FAA used emergency orders, including data on regional variation in their use, the types of certificates holders affected, and the final outcomes of case initiated using emergency orders; (2) the ways in which changes in FAA's policies might have affected the agency's use for emergency orders; and (3) the time needed for FAA to investigate alleged violations and issue emergency orders.
NTIS
Civil Aviation; Emergencies; Aircraft Safety; Flight Safety

04
AIRCRAFT COMMUNICATIONS AND NAVIGATION

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

19980231957 Naval Postgraduate School, Monterey, CA USA
An Integrated INS/GPS Navigation System for Small AUVS Using an Asynchronous Kalman Filter
Hernadez, Glenn C., Naval Postgraduate School, USA; Jun. 1998; 56p; In English
Report No.(s): AD-A351512; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

A Small AUV Navigation System (SANS) is being developed at the Naval Postgraduate School. The SANS is an integrated INS/GPS navigation system composed of low-cost, small-size components. It is designed to demonstrate the feasibility of using a low-cost Inertial Measurement Unit (IMU) to navigate between intermittent GPS fixes. This thesis presents recent improvements to the SANS hardware and software. The 486-based ESP computer used in the previous version of SANS is now replaced by an AMD 586DX133 based PC/104 computer to provide more computing power, reliability and compatibility with PC/104 industrial standards. The previous SANS navigation filter consisting of a complementary constant gain filter is now aided by an asynchronous Kalman filter. This navigation filter has six states for orientation estimation (constant gain) and eight states for position estimation (Kalman filtered). Low-frequency DGPS noise is explicitly modeled based on an experimentally obtained a autocorrelation function. Ocean currents are also modeled as a low-frequency random process. The asynchronous nature of GPS measurements resulting from AUV submergence or wave splash on the DGPS antennas is also taken into account by adopting an asynchronous Kalman filter as the basis for the SANS software. Matlab simulation studies of the asynchronous filter have been conducted and results documented in this thesis.
DTIC
Kalman Filters; Global Positioning System; Computer Programs; Inertial Navigation

19980231986 Nanjing Univ. of Aeronautics and Astronautics, Nanjing, Jiangsu, China
Research on Terrain-Aided Navigation System
Xin, Ma, Nanjing Univ. of Aeronautics and Astronautics, Nanjing, China; Xin, Yuan, Nanjing Univ. of Aeronautics and Astronautics, Nanjing, China; Journal of Nanjing University of Aeronautics and Astronautics; Jun. 1997; ISSN 1005-2615; Volume 29, No. 3, pp. 289-294; In Chinese; No Copyright; Avail: Issuing Activity, Hardcopy, Microfiche

Higher demands have been required for the navigation accuracy of various strategic and tactical weapons. In addition to the high accuracy, the navigation systems should have other features such as all-weather, independence, high activeness, high electronic- environment and concealment in modern wars. Only the integrated navigation systems, in which the inertial navigation system and other navigation devices are optimized can satisfy these demands. The terrain-aided navigation system is of bright prospects. The paper describes one of the Terrain-Aided Navigation-SITAN and produces the simulated digital maps by computer on the basis of the statistical model of the terrain. The simulation results prove that the navigation accuracy can be improved by using the digital maps to aid the inertial navigation, and influenced by the roughness of the terrain.
Author
Terrain; Terrain Analysis; Navigation Aids; Inertial Navigation; Air Navigation

19980233518 Beijing Univ. of Aeronautics and Astronautics, Beijing, China
Pilot-Navigation Expert System
Wenru, Ning, Beijing Univ. of Aeronautics and Astronautics, China; Haijun, Shen, Beijing Univ. of Aeronautics and Astronautics, China; Journal of Beijing University of Aeronautics and Astronautics; Dec. 1997; ISSN 1001-5965; Volume 23, No. 6, pp. 763-768; In Chinese; No Copyright; Avail: Issuing Activity, Hardcopy, Microfiche

The paper addresses a pilot navigation expert system scheme which consists of two channels: Pilot Assistant (PA) channel and Pilot Assistant Learning (PAL) channel. The PA channel performs the navigation calculation, the situation estimation, the decision making and the display of current situation and the suggestions to the pilot, while the PAL performs the update of Knowledge Base(KB) which includes such tasks as the evaluation of the maneuvers performed by the pilot, the induction and generalization of the rules and knowledge and the correction of the knowledge base. Some principles, methods and algorithms concerning the system were discussed, and the learning channel was intensively considered in this paper.
Author
Navigation; Expert Systems; Aircraft Pilots

19980236139 Army Research Lab., Sensors Directorate, Adelphi, MD USA
Performance of the Miniature Airborne GPS Receiver, *Oct. 1997 - Jun. 1998*
Van Flandern, Tom; Bahder, Thomas B.; Jul. 1998; 21p; In English
Report No.(s): AD-A351939; ARL-TR-1739; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

At a fixed, well-surveyed location, position determinations from a MAGR (Miniaturized Airborne Global Positioning System Receiver) averaged over a six-week period were correct to within 0.5 m. However, the standard deviation of an individual position determination was 56 m. Almost 20 percent of the individual position determinations had errors exceeding 20 percent. One in every 300 position determinations had an error exceeding 0.5 km. This anomalously large error distribution tail raises questions about the MAGR's suitability for some Army-critical functions, such as precision guidance.
DTIC
Global Positioning System; Receivers; Miniature Electronic Equipment; Position (Location)

19980236153 Joint Chiefs of Staff, Washington, DC USA
1998 CJCS Master Positioning, Navigation, and Timing Plan
Feb. 13, 1998; 94p; In English
Report No.(s): AD-A352402; CJCSI-6130.01A; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

This instruction implements DOD positioning, navigation, and timing (PNT) policy and provides consolidated CJCS instructions to PNT developers and users. It identifies the roles and responsibilities of PNT users, developers, and supporting DOD elements. In recognition of the four emerging operational concepts presented in Joint Vision 2010, this CJCS Master Positioning, Navigation, and Timing Plan (MPNTP) provides the backbone of PNT information management for the US warfighter. This plan also informs the Services about major DOD PNT R&D activities. As such, it provides a broad, consolidated PNT information base to ensure consistent, informed management decisions and better allocation of Service resources.
DTIC
Navigation; Information Systems; Time Measurement; Positioning; Information Management

19980236829 NASA Goddard Space Flight Center, Greenbelt, MD USA
Research Lasers and Air Traffic Safety: Issues, Concerns and Responsibilities of the Research Community
Nessler, Phillip J., Jr., NASA Goddard Space Flight Center, USA; Nineteenth International Laser Radar Conference; Jul. 1998, Part 2, pp. 975-977; In English; Also announced as 19980236718; No Copyright; Avail: CASI; A01, Hardcopy; A04, Microfiche

The subject of outdoor use of lasers relative to air traffic has become a diverse and dynamic topic. During the past several decades, the use of lasers in outdoor research activities have increased significantly. Increases in the outdoor use of lasers and increases in air traffic densities have changed the levels of risk involved. to date there have been no documented incidents of air traffic interference from research lasers; however, incidents involving display lasers have shown a marked increase. As a result of the national response to these incidents, new concerns over lasers have arisen. Through the efforts of the SAE G-10T Laser Safety Hazards Subcommittee and the ANSI Z136.6 development committee, potential detrimental effects to air traffic beyond the traditional eye damage concerns have been identified. An increased emphasis from the Federal Aviation Administration (FAA), the Center for Devices and Radiological Hazards (CDRH), and the National Transportation Safety Board (NTSB) along with increased concern by the public have resulted in focused scrutiny of potential hazards presented by lasers. The research community needs to rethink the traditional methods of risk evaluation and application of protective measures. The best current approach to assure adequate protection of air traffic is the application of viable hazard and risk analysis and the use of validated

protective measures. Standards making efforts and regulatory development must be supported by the research community to assure that reasonable measures are developed. Without input, standards and regulations can be developed that are not compatible with the needs of the research community. Finally, support is needed for the continued development and validation of protective measures.
Derived from text
Laser Applications; Air Traffic; Radiation Hazards; Laser Damage; Safety Management

05
AIRCRAFT DESIGN, TESTING AND PERFORMANCE

Includes aircraft simulation technology.

19980231962 Army Research Lab., Sensors Directorate, Adelphi, MD USA
Doppler Signature Measurements of an Mi-24 Hind-D Helicopter at 92 GHz
Wellman, Ronald J., Army Research Lab., USA; Silvious, Jerry L., Army Research Lab., USA; Jul. 1998; 20p; In English
Report No.(s): AD-A351581; ARL-TR-1637; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This report describes millimeter wave Doppler signature measurements made on a hovering Mi-24 Hind-D helicopter. The measurement system, a CW sensor operating at 92 GHz, is described in detail. The recording system consisted of a high speed A/D board installed in a dual Pentium computer running Windows NT. The report presents the results of the Doppler analysis of the data. The signatures created by the rotating blades of the jet turbines known as jet engine modulation (JEM) lines and the Doppler signature components created by the many moving parts of the main and tail rotor assemblies are discussed in detail and a comparison to the theoretical values is presented.
DTIC
Doppler Effect; Computer Programs; Helicopters; Signatures; Millimeter Waves

19980231973 NASA Langley Research Center, Hampton, VA USA
An Analysis of Flight-Test Measurements of the Wing Structural Deformations in Rough Air of a Large Flexible Swept-Wing Airplane
Murrow, Harold N., NASA Langley Research Center, USA; Jan. 1959; 50p; In English
Report No.(s): NASA-MEMO-12-3-58L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An analysis is made of wing deflection and streamwise twist measurements in rough-air flight of a large flexible swept-wing bomber. Random-process techniques are employed in analyzing the data in order to describe the magnitude and characteristics of the wing deflection and twist responses to rough air. Power spectra and frequency-response functions for the wing deflection and twist responses at several spanwise stations are presented. The frequency-response functions describe direct and absolute response characteristics to turbulence and provide a convenient basis for assessing analytic calculation techniques. The wing deformations in rough air are compared with the expected deformations for quasi-static loadings of the same magnitude, and the amplifications are determined. The results obtained indicate that generally the deflections are amplified by a small amount, while the streamwise twists are amplified by factors of the order of 2.0. The magnitudes of both the deflection velocities and the twist angles are shown to have significant effects on the local angles of attack at the various stations and provide the main source of aerodynamic loading, particularly at frequencies in the vicinity of the first wing-vibration mode.
Author
Aerodynamic Loads; Swept Wings; Flight Tests; Deformation; Analysis (Mathematics)

19980232006 NASA Langley Research Center, Hampton, VA USA
Results of a Cyclic Load Test of an RB-47E Airplane
Huston, Wilber B., NASA Langley Research Center, USA; 1959; 66p; In English
Report No.(s): NASA-MEMO-3-15-59L; AF-AM-171; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

Results of a cyclic load test made by NASA on an EB-47E airplane are given. The test reported on is for one of three B-47 airplanes in a test program set up by the U. S. Air Force to evaluate the effect of wing structural reinforcements on fatigue life. As a result of crack development in the upper fuselage longerons of the other two airplanes in the program, a longeron and fuselage skin modification was incorporated early in the test. Fuselage strain-gage measurements made before and after the longeron modification and wing strain-gage measurements made only after wing reinforcement are summarized. The history of crack develop-

ment and repair is given in detail. Testing was terminated one sequence short of the planned end of the program with the occurrence of a major crack in the lower right wing skin.
Author
B-47 Aircraft; Cyclic Loads; Load Tests; Strain Gages; Fatigue Life

19980232014 NASA Ames Research Center, Moffett Field, CA USA
Subsonic Wing Loadings on a 45 deg Sweptback Wing and Body Combination at High Angles of Attack
Axelson, John A, NASA Ames Research Center, USA; Haacker, Jack F., NASA Ames Research Center, USA; Feb. 1959; 58p; In English
Report No.(s): NASA-MEMO-1-18-59A; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

A study has been made of the subsonic pressure distributions and loadings for a 45 deg sweptback-wing and body combination at angles of attack up to 36 deg. The wing had an aspect ratio of 5.5, a taper ratio of 0.53, and NACA 64A010 sections normal to the quarter-chord line and was mounted on a slender body of fineness ratio 12.5. Test results are presented for Mach numbers of 0.30 and 0.50 with corresponding Reynolds numbers of 1.5 and 2.0 million, respectively. The stall patterns and spanwise loadings at high angles of attack for the present model are correlated with those for other 45 deg sweptback wing and body combinations having aspect ratios between 4.0 and 8.0. A tentative approach is presented for extrapolating the Weissinger span-loading method to higher angles of attack, and for deriving the spanwise-load distributions for 45 deg sweptback wings at angles of attack above 20 deg. The investigation also included tests of the body in combination with only one panel of the swept wing. The problem of estimating the normal-force coefficient for the single panel at high angles of attack is considered.
Author
Angle of Attack; Pressure Distribution; Reynolds Number; Estimating; Wing Loading

19980232056 National Academy of Sciences - National Research Council, Washington, DC USA
Review and Evaluation of the Air Force Hypersonic Technology Program ***Final Report, 15 Apr. 1997 - 14 Jul. 1998***
Jul. 14, 1998; 70p; In English
Contract(s)/Grant(s): F49620-97-1-0281
Report No.(s): AD-A351152; AFSB-J-97-01-A; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

The Air Force requested the National Research Council to examine the Hypersonic Technology (HyTech) Program managed by the Air Force Research Laboratory at Wright Patterson Air Force Base in Dayton, Ohio. A committee with the necessary expertise was formed to respond to a statement of task that asked a series of questions to determine if a Mach 8, hydrocarbon-fueled, scramjet-powered, air-launched cruise missile could be developed by the Air Force to reach an initial operational capability by the year 2015. It was the conclusion by the committee that this type of weapons system could be developed by that date, but not as a result of the effort being applied within the HyTech Program. That program was reduced (due to funding limitations) to a technology base program to support the propulsion component of the missile only. Many other enabling technologies such as materials, guidance and control, and warhead development would require similar emphasis under a coordinated development program that includes necessary flight testing. In the report, the committee offers its plan for accomplishing the necessary steps to achieve an initial operational capability for the missile by 2015.
DTIC
Hypersonics; Air Launching

19980232063 Naval Postgraduate School, Monterey, CA USA
A History of the Survivability Design of Military Aircraft
Ball, Robert E.; Atkinson, Dale B.; Jan. 1998; 16p; In English
Report No.(s): AD-A351434; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

In simple words, survivability in combat is achieved by not getting hit by the enemy's weapons or withstanding the effects of any hits suffered. The likelihood an aircraft gets hit while on a mission is referred to as the aircraft's susceptibility, and the likelihood the aircraft is killed by the hit is referred to as the aircraft's vulnerability. Reduction of aircraft susceptibility is achieved by: (1) the selection of the appropriate weapons, tactics, threat suppression, and support jamming for the mission, (2) reducing the aircraft's signatures, and (3) incorporating on-board threat warning equipment and countermeasures in the form of electromagnetic jammers and expendables. Reduction of aircraft vulnerability is achieved by: (1) the use of redundant flight critical components, adequately separated so that a single hit does not kill them all, (2) properly locating the critical components to reduce vulnerability, (3) designing the critical components, or adding equipment, to suppress the effects of any hits, and (4) shielding those components that cannot be protected otherwise. All of these concepts for enhancing survivability impact the design of the aircraft. The importance of survivability in the design of aircraft has varied throughout the 20th century from a total neglect to

the highest priority. This paper presents the evolution of the survivability design of aircraft from the beginning of World War II to the present time.
DTIC
Aircraft Design; Combat

19980232089 NASA Ames Research Center, Moffett Field, CA USA
A Flight Evaluation of the Factors which Influence the Selection of Landing Approach Speeds
Drinkwater, Fred J., III, NASA Ames Research Center, USA; Cooper, George E., NASA Ames Research Center, USA; Dec. 1958; 26p; In English
Report No.(s): NASA-MEMO-10-6-58A; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The factors which influence the selection of landing approach speeds are discussed from the pilot's point of view. Concepts were developed and data were obtained during a landing approach flight investigation of a large number of jet airplane configurations which included straight-wing, swept-wing, and delta-wing airplanes as well as several applications of boundary-layer control. Since the fundamental limitation to further reductions in approach speed on most configurations appeared to be associated with the reduction in the pilot's ability to control flight path angle and airspeed, this problem forms the basis of the report. A simplified equation is presented showing the basic parameters which govern the flight path angle and airspeed changes, and pilot control techniques are discussed in relation to this equation. Attention is given to several independent aerodynamic characteristics which do not affect the flight path angle or airspeed directly but which determine to a large extent the effort and attention required of the pilot in controlling these factors during the approach. These include stall characteristics, stability about all axes, and changes in trim due to thrust adjustments. The report considers the relationship between piloting technique and all of the factors previously mentioned. A piloting technique which was found to be highly desirable for control of high-performance airplanes is described and the pilot's attitudes toward low-speed flight which bear heavily on the selection of landing approach speeds under operational conditions are discussed.
Author
Flight Tests; Landing Speed; Data Acquisition; Control Systems Design; Aerodynamic Characteristics

19980232153 General Accounting Office, National Security and International Affairs Div., Washington, DC USA
Report to the Secretary of Defense. Army Aviation: Apache Longbow Weight and Communication Issues
Sep. 1998; 18p; In English
Report No.(s): GAO/NSIAD-98-203; B-278882; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The Apache Longbow helicopter is designed to conduct precision attacks in adverse weather in battlefields obscured by smoke, automatically engage multiple targets, and provide fire-and-forget missile capability. The Apache Longbow configuration consists of a modified airframe, a fire control radar, and a new Longbow (radio frequency) Hellfire missile. The Army plans to reconfigure many of the current Apache fleet to the Apache Longbow requirements with full combat capabilities, and others will be upgraded with less powerful engines, and not have the capability to communicate target data. The Apache Longbow is being designed to meet certain requirements to enhance survivability in combat missions. It must be able to climb at least 450 feet per minute at 4,000 feet, while carrying a full fuel load and ammunition. The report states that this requirement cannot be achieved, and the systems survivability will be adversely impacted. The Army also requires the Apache Longbow to have a capability to transfer data about targets to other Apache Longbows, via radio. The report states that initially the radio is not available, and that unresolved technical issues have delayed the radio's development.
CASI
Climbing Flight; Combat; Military Helicopters; Helicopter Performance; Aircraft Survivability; Military Technology; Congressional Reports

19980234657 NASA Langley Research Center, Hampton, VA USA
Bi-Level Integrated System Synthesis (BLISS)
Sobieszczanski-Sobieski, Jaroslaw, NASA Langley Research Center, USA; Agte, Jeremy S., George Washington Univ., USA; Sandusky, Robert R., Jr., George Washington Univ., USA; Aug. 1998; 40p; In English
Contract(s)/Grant(s): RTOP 509-10-31-03
Report No.(s): NASA/TM-1998-208715; NAS 1.15:208715; L-17778; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

BLISS is a method for optimization of engineering systems by decomposition. It separates the system level optimization, having a relatively small number of design variables, from the potentially numerous subsystem optimizations that may each have a large number of local design variables. The subsystem optimizations are autonomous and may be conducted concurrently. Sub-

system and system optimizations alternate, linked by sensitivity data, producing a design improvement in each iteration. Starting from a best guess initial design, the method improves that design in iterative cycles, each cycle comprised of two steps. In step one, the system level variables are frozen and the improvement is achieved by separate, concurrent, and autonomous optimizations in the local variable subdomains. In step two, further improvement is sought in the space of the system level variables. Optimum sensitivity data link the second step to the first. The method prototype was implemented using MATLAB and iSIGHT programming software and tested on a simplified, conceptual level supersonic business jet design, and a detailed design of an electronic device. Satisfactory convergence and favorable agreement with the benchmark results were observed. Modularity of the method is intended to fit the human organization and map well on the computing technology of concurrent processing.
Author
Parallel Processing (Computers); Supersonic Jet Flow; Design Analysis; Concurrent Processing; Computer Programming

19980235520 Georgia Inst. of Tech., Atlanta, GA USA
Normal Component of Induced Velocity for Entire Field of a Uniformly Loaded Lifting Rotor with Highly Swept Wake as Determined by Electromagnetic Analog
Castles, Walter, Jr., Georgia Inst. of Tech., USA; Durham, Howard L., Jr., Georgia Inst. of Tech., USA; Kevorkian, Jirair, Georgia Inst. of Tech., USA; 1959; 30p; In English; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Values of the normal component of induced velocity throughout the entire field of a uniformly loaded r(rotor at high high speed are presented in the form of charts and tables. Many points were found by an electromagnetic analog, details of which are given. Comparisons of computed and analog values for the induced velocity indicate that the latter are sufficiently accurate for engineering purposes.
Author
Lifting Rotors; Charts; Rotor Speed; Velocity Distribution

19980235529 NASA Marshall Space Flight Center, Huntsville, AL USA
Aircraft Structural Mass Property Prediction Using Conceptual-Level Structural Analysis
Sexstone, Matthew G., NASA Marshall Space Flight Center, USA; 1998; 18p; In English; 57th, 18-20 May 1998, Wichita, KS, USA; Sponsored by Society of Allied Weight Engineers, Inc., USA
Report No.(s): NASA/TM-1998-208129; NAS 1.15:208129; SAWE Paper 2410; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This paper describes a methodology that extends the use of the Equivalent LAminated Plate Solution (ELAPS) structural analysis code from conceptual-level aircraft structural analysis to conceptual-level aircraft mass property analysis. Mass property analysis in aircraft structures has historically depended upon parametric weight equations at the conceptual design level and Finite Element Analysis (FEA) at the detailed design level ELAPS allows for the modeling of detailed geometry, metallic and composite materials, and non-structural mass coupled with analytical structural sizing to produce high-fidelity mass property analyses representing fully configured vehicles early in the design process. This capability is especially valuable for unusual configuration and advanced concept development where existing parametric weight equations are inapplicable and FEA is too time consuming for conceptual design. This paper contrasts the use of ELAPS relative to empirical weight equations and FEA. ELAPS modeling techniques are described and the ELAPS-based mass property analysis process is detailed Examples of mass property stochastic calculations produced during a recent systems study are provided This study involved the analysis of three remotely piloted aircraft required to carry scientific payloads to very high altitudes at subsonic speeds. Due to the extreme nature of this high-altitude flight regime,few existing vehicle designs are available for use in performance and weight prediction. ELAPS was employed within a concurrent engineering analysis process that simultaneously produces aerodynamic, structural, and static aeroelastic results for input to aircraft performance analyses. The ELAPS models produced for each concept were also used to provide stochastic analyses of wing structural mass properties. The results of this effort indicate that ELAPS is an efficient means to conduct multidisciplinary trade studies at the conceptual design level.
Author
Structural Analysis; Aircraft Design; Performance Prediction; Finite Element Method; Concurrent Engineering; Aircraft Performance; Aeroelasticity

19980235564 NASA Dryden Flight Research Center, Edwards, CA USA
Deterministic Reconfigurable Control Design for the X-33 Vehicle
Wagner, Elaine A., Lockheed Martin Corp., USA; Burken, John J., NASA Dryden Flight Research Center, USA; Hanson, Curtis E., NASA Dryden Flight Research Center, USA; Wohletz, Jerry M., Massachusetts Inst. of Tech., USA; 1998; 10p; In English, 10-12 Aug. 1998, Boston, MA, USA; Sponsored by American Inst. of Aeronautics and Astronautics, USA

Contract(s)/Grant(s): NCC8-115
Report No.(s): AIAA Paper 98-4413; Copyright; Avail: Issuing Activity, Hardcopy, Microfiche

In the event of a control surface failure, the purpose of a reconfigurable control system is to redistribute the control effort among the remaining working surfaces such that satisfactory stability and performance are retained. Four reconfigurable control design methods were investigated for the X-33 vehicle: Redistributed Pseudo-Inverse, General Constrained Optimization, Automated Failure Dependent Gain Schedule, and an Off-line Nonlinear General Constrained Optimization. The Off-line Nonlinear General Constrained Optimization approach was chosen for implementation on the X-33. Two example failures are shown, a right outboard elevon jam at 25 deg. at a Mach 3 entry condition, and a left rudder jam at 30 degrees. Note however, that reconfigurable control laws have been designed for the entire flight envelope. Comparisons between responses with the nominal controller and reconfigurable controllers show the benefits of reconfiguration. Single jam aerosurface failures were considered, and failure detection and identification is considered accomplished in the actuator controller. The X-33 flight control system will incorporate reconfigurable flight control in the baseline system.

Author

X-33 Reusable Launch Vehicle; Supersonic Speed; Flight Envelopes; Flight Control; Control Theory; Control Surfaces

19980235565 Lockheed Martin Corp., Skunk Works, Palmdale, CA USA

Flight Dynamics and Stability and Control Characteristics of the X-33 Vehicle

Lee, H. P., Lockheed Martin Corp., USA; Chang, M., Lockheed Martin Corp., USA; Kaiser, M. K., Lockheed Martin Corp., USA; 1998; 12p; In English; Guidance, Navigation and Control, 10-12 Aug. 1998, Boston, MA, USA; Sponsored by American Inst. of Aeronautics and Astronautics, USA
Contract(s)/Grant(s): NCC8-115
Report No.(s): AIAA Paper 98-4410; No Copyright; Avail: Issuing Activity, Hardcopy, Microfiche

This paper presents the flight dynamics and stability and control characteristics of the X-33 vehicle. The vehicle model is based on aerodynamic data obtained from wind tunnel testing which includes power effects during ascent phase of the flight. Mathematical modeling of the Aerospike propulsion system and the reaction control systems are also included. Stability characteristics of the vehicle are presented for various flight regimes. Flight dynamics characteristics of the vehicle is based on linearized model of the vehicle. They are shown in this paper for the second flight trajectory which is to be flown from Edwards Air Force Base in California to Michael Army Air Field in Utah. This trajectory, designed as Michael-C, is designed to provide s flight profile with sufficient hypersonic Mach numbers to generate an aerothermal environment that produces the maximum catalytic aeroheating for the validation of the X-33 technology development. Analytical results are provided for three phases of the flight: Ascent, Entry and Terminal Area Energy Management (TAEM). Methods for estimating control power requirement to achieve desirable level of static stability are presented.

Author

X-33 Reusable Launch Vehicle; Aerodynamics; Aerospike Engines; Dynamic Control; Flight Paths; Hypersonics; Propulsion; Trajectories

19980235624 NASA Langley Research Center, Hampton, VA USA

Analysis of Acceleration, Airspeed, and Gust-Velocity Data From a Four-Engine Transport Airplane Operating Over a Northwestern USA Alaska Route

Engel, Jerome N., NASA Langley Research Center, USA; Copp, Martin R., NASA Langley Research Center, USA; Feb. 1959; 18p; In English
Report No.(s): NASA-MEMO-1-17-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Acceleration, airspeed, and altitude data obtained with an NACA VGH recorder from a four-engine commercial transport airplane operating over a northwestern USA-Alaska route were evaluated to determine the magnitude and frequency of occurrence of gust and maneuver accelerations., operating airspeeds, and gust velocities. The results obtained were then compared with the results previously reported in NACA Technical Note 3475 for two similar airplanes operating over transcontinental routes in the United States. No large variations in the gust experience for the three operations were noted. The results indicate that the gust-load experience of the present operation closely approximated that of the central transcontinental route in the USA with which it is compared and showed differences of about 4 to 1 when compared with that of the southern transcontinental route in the United States. In general, accelerations due to gusts occurred much more frequently than those due to operational maneuvers. At a measured normal-acceleration increment of 0.5g, accelerations due to gusts occurred roughly 35 times more frequently than those due to operational maneuvers.

Author

Airspeed; Acceleration; Transport Aircraft; Gust Loads; Gusts

19980235626 NASA Dryden Flight Research Center, Edwards, CA USA

Approach and Landing Investigation at Lift-Drag Ratios of 2 to 4 Utilizing a Straight-Wing Fighter Airplane

Matranga, Gene J., NASA Dryden Flight Research Center, USA; Armstrong, Neil A., NASA Dryden Flight Research Center, USA; Aug. 1959; 24p; In English

Report No.(s): NASA-TM-X-31; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A series of landings was performed with a straight-wing airplane to evaluate the effect of low lift-drag ratios on approach and landing characteristics. Landings with a peak lift-drag ratio as low as 3 were performed by altering the airplane configuration (extending speed brakes, flaps, and gear and reducing throttle setting). As lift-drag ratio was reduced, it was necessary either to make the landing pattern tighter or to increase initial altitude, or both. At the lowest lift-drag ratio the pilots believed a 270 deg overhead pattern was advisable because of the greater ease afforded in visually positioning the airplane. The values of the pertinent flare parameters increased with the reduction of lift-drag ratio. These parameters included time required for final flare; speed change during final flare; and altitude, glide slope, indicated airspeed, and vertical velocity at initiation of final flare. The pilots believed that the tolerable limit was reached with this airplane in the present configuration, and that if, because of a further reduction in lift-drag ratio, more severe approaches than those experienced in this program were attempted, additional aids would be required to determine the flare-initiation point.

Author

Aerodynamic Characteristics; Approach; Fighter Aircraft; Rectangular Wings; Lift Drag Ratio; Airspeed

19980235628 NASA Langley Research Center, Hampton, VA USA

Effects of Horizontal-Control Planform and Wing-Leading-Edge Modification on Low-Speed Longitudinal Aerodynamic Characteristics of a Canard Airplane Configuration

Spencer, Bernard, Jr., NASA Langley Research Center, USA; 1981; 541p; In English

Report No.(s): NASA-TM-X-549; L-1372; No Copyright; Avail: CASI; A23, Hardcopy; A04, Microfiche

An investigation at low subsonic speeds has been conducted in the Langley 300-MPH 7- by 10-foot tunnel. The basic wing had a trapezoidal planform, an aspect ratio of 3.0., a taper ratio of 0.143, and an unswept 80-percent-chord line. Modifications to the basic wing included deflectable full-span and partial-span leading-edge chord-extensions. A trapezoidal horizontal control similar in planform to the basic wing and a 60 deg sweptback delta horizontal control were tested in conjunction with the wing. The total planform area of each horizontal control was 16 percent of the total basic-wing area. Modifications to these horizontal controls included addition of a full-span chord-extension to the trapezoidal planform and a fence to the delta planform.

Author

Aerodynamic Characteristics; Canard Configurations; Leading Edges; Planforms

19980236450 NASA Langley Research Center, Hampton, VA USA

Analysis of the Flight Motions of a Small Deployable Glider Configuration

Coe, Paul L., Jr., NASA Langley Research Center, USA; Mar. 1975; 32p; In English

Report No.(s): NASA-TM-SX-3205; L-10031; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation was conducted at the request of the U.S. Air Force Avionics Laboratory to analyze the flight characteristics of a small uncontrolled glider with folding wings. The study consisted of wind-tunnel tests of an actual glider and a theoretical analysis of the performance, stability, and trimmability of the configuration.

Author

Gliders; Flight Characteristics; Avionics

19980236454 NASA Langley Research Center, Hampton, VA USA

Aerodynamic Characteristics of a Variable Sweep Fighter Model at Mach Numbers from 1.60 to 2.36

Blair, A. B., Jr., NASA Langley Research Center, USA; Richardson, Celia S., NASA Langley Research Center, USA; Mar. 1972; 110p; In English

Contract(s)/Grant(s): RTOP 136-63-02-28

Report No.(s): NASA-TM-SX-2489; N-AM-154; L-8048; No Copyright; Avail: CASI; A06, Hardcopy; A02, Microfiche

An investigation has been made in the Langley Unitary Plan wind tunnel to determine the aerodynamic characteristics of a 1/22-scale model of the Grumman F-14A airplane with various combinations of external stores mounted both in the fuselage channel and on wing-glove pylons. In addition, several inlet-exit geometries, as well as a configuration incorporating modifications to simulate a reconnaissance vehicle, were investigated. The model was tested at Mach numbers from 1.60 to 2.36 through an angle-of-attack range from about -4 deg to 20 deg and an angle-of-sideslip range from about -4 deg to 8 deg. A Reynolds number

of 8.20 x 10(exp 6) per meter (2.5 x 10(exp 6) per foot) was used at the lower angles of attack and 4.92 x 10(exp 6) per meter (1.5 x 10(exp 6) per foot) at the higher angles of attack. In order to expedite publication, no analysis of the data obtained has been made.
Author
Aerodynamic Characteristics; F-14 Aircraft; Scale Models; Angle of Attack; Wing Planforms

19980236510 Defence Science and Technology Organisation, Information Technology Div., Canberra Australia
Accelerated Environmental Testing of Composite Materials
Vodicka, Roger, Defence Science and Technology Organisation, Australia; Apr. 1998; 60p; In English
Report No.(s): AD-A352718; DSTO-TR-0657; DODA-AR-010-516; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

Composite materials are found to lose mechanical on exposure to aircraft operating environments. This is mainly due to absorption of moisture from humid air by the matrix material. Composite materials are extensively used by the RAAF for both major structural components on the F/A-18 and for bonded repairs and doublers. The performance of these materials under long-term environmental exposure is an important aspect of both aircraft certification and in the understanding of how the components will age. This report provides a broad overview of environmental effects on composite materials and methods which may be used to predict their long-term behaviour. The use of accelerated testing environments in the laboratory is an attractive proposition as it enables tests to be carried out in reduced time frames. A number of accelerated testing methodologies and their implications are outlined here. Accelerated testing can be carried out with confidence if the exposure conditions are representative and the failure modes of the material during mechanical tests reflect those seen in service.
DTIC
Composite Materials; Mechanical Properties; Aircraft Maintenance; Bonded Joints; Environmental Tests; Accelerated Life Tests; Matrix Materials

19980236806 NASA Langley Research Center, Hampton, VA USA
Differential Absorption Lidar (DIAL) Measurements of Atmospheric Water Vapor Utilizing Robotic Aircraft
Hoang, Ngoc, Aurora Flight Sciences Corp., USA; DeYoung, Russell J., NASA Langley Research Center, USA; Prasad, Coorg R., Science and Engineering Services, Inc., USA; Laufer, Gabriel, Virginia Univ., USA; Nineteenth International Laser Radar Conference; Jul. 1998, Part 2, pp. 891-894; In English; Also announced as 19980236718; No Copyright; Avail: CASI; A01, Hardcopy; A04, Microfiche

A new unpiloted air vehicle (UAV) based water vapor DIAL system will be described. This system is expected to offer lower operating costs, longer test duration and severe weather capabilities. A new high-efficiency, compact, light weight, diode-pumped, tunable Cr:LiSAF laser will be developed to meet the UAV payload weight and size limitations and its constraints in cooling capacity, physical size and payload. Similarly, a new receiver system using a single mirror telescope and an avalanche photo diode (APD) will be developed. Projected UAV parameters are expected to allow operation at altitudes up to 20 km, endurance of 24 hrs and speed of 400 km/hr. At these conditions measurements of water vapor at an uncertainty of 2-10% with a vertical resolution of 200 m and horizontal resolution of 10 km will be possible.
Derived from text
Differential Absorption Lidar; Radar Measurement; Remote Sensors; Tunable Lasers; Atmospheric Moisture; Pilotless Aircraft

19980236840 NASA Dryden Flight Research Center, Edwards, CA USA
Launch, Low-Speed, and Landing Characteristics Determined from the First Flight of the North American X-15 Research Airplane
Finch, Thomas W., NASA Dryden Flight Research Center, USA; Matranga, Gene J., NASA Dryden Flight Research Center, USA; Sep. 1959; 28p; In English
Report No.(s): NASA-TM-X-195; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The first flight of the North American X-15 research airplane was made on June 8, 1959. This was accomplished after completion of a series of captive flights with the X-15 attached to the B-52 carrier airplane to demonstrate the aerodynamic and systems compatibility of the X-15//B-52 combination and the X-15 subsystem operation. This flight was planned as a glide flight so that the pilot need not be concerned with the propulsion system. Discussions of the launch, low-speed maneuvering, and landing characteristics are presented, and the results are compared with predictions from preflight studies. The launch characteristics were generally satisfactory, and the X-15 vertical tail adequately cleared the B-52 wing cutout. The actual landing pattern and landing characteristics compared favorably with predictions, and the recommended landing technique of lowering the flaps and landing gear at a low altitude appears to be a satisfactory method of landing the X-15 airplane. There was a quantitative correlation between flight-measured and predicted lift-drag-ratio characteristics in the clean configuration and a qualitative correlation in the landing configuration. A longitudinal-controllability problem, which became severe in the landing configuration, was evident throughout the flight and, apparently, was aggravated by the sensitivity of the side-located control stick. In the low-to-moderate angle-of-attack range covered, the longitudinal and directional stability were indicated to be adequate.

Author

Aerodynamic Characteristics; Low Speed; Launching; Landing; Directional Stability; Maneuvers; X-15 Aircraft

19980236891 NASA Marshall Space Flight Center, Huntsville, AL USA

Analysis of Linear Aerospike Plume Induced X-33 Base Heating Environment

Wang, Ten-See, NASA Marshall Space Flight Center, USA; 1998; 32p; In English; 7th; Joint Thermophysics and Heat Transfer Conference, 15-18 Jun. 1998, Albuquerque, NM, USA; Sponsored by American Society of Mechanical Engineers, USA; Original contains color illustrations

Contract(s)/Grant(s): NAS8-40582; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Computational analysis is conducted to study the effect of an linear aerospike engine plume on the X-33 base-heating environment during ascent flight. to properly account for the freestream-body interaction and to allow for potential plume-induced flow-separation, the thermo-flowfield of the entire vehicle at several trajectory points is computed. A sequential grid-refinement technique is used in conjunction with solution-adaptive, patched, and embedded grid methods to limit the model to a manageable size. The computational methodology is based on a three-dimensional, finite-difference, viscous flow, chemically reacting, pressure-based computational fluid dynamics formulation, and a three-dimensional, finite-volume, spectral-line based weighted-sum-of-gray-gases absorption, computational radiation heat transfer formulation. The computed forebody and afterbody surface pressure coefficients and base pressure characteristic curves are compared with those of a cold-flow test. The predicted convective and radiative base-heat fluxes, the effect of base-bleed, and the potential of plume-induced flow separation are presented.

Author

X-33 Reusable Launch Vehicle; Aerospike Engines; Base Heating; Free Flow; Radiative Heat Transfer; Three Dimensional Flow; Boundary Layer Separation; Convective Heat Transfer

06
AIRCRAFT INSTRUMENTATION

Includes cockpit and cabin display devices; and flight instruments.

19980233514 Beijing Univ. of Aeronautics and Astronautics, Beijing, China

Computer Aided Design on Location of Fuel Sensor in Aircraft

Li, Guan, Beijing Univ. of Aeronautics and Astronautics, China; Xinglu, Chen, Beijing Univ. of Aeronautics and Astronautics, China; Journal of Beijing University of Aeronautics and Astronautics; Dec. 1997; ISSN 1001-5965; Volume 23, No. 6, pp. 783-787; In Chinese; No Copyright; Avail: Issuing Activity, Hardcopy, Microfiche

The location principle of fuel sensors in digital fuel gaging systems according to the principle in fuel quantity measurement is introduced. From which, it presents the conception of measurable area and sets forth the key techniques in calculating it. In the end, the paper provides some relative conclusions about the location design of fuel sensors in aircraft by calculating measurable area and check the measurability.

Author

Computer Aided Design; Aircraft Fuel Systems; Position (Location)

19980235212 Beijing Univ. of Aeronautics and Astronautics, Beijing, China

Picture, Image and Symbolized Measurements in Aircraft and Spacecraft Instruments

Junqin, Huang, Beijing Univ. of Aeronautics and Astronautics, China; Journal of Beijing University of Aeronautics and Astronautics; Dec. 1997; ISSN 1001-5965; Volume 23, No. 6, pp. 788-793; In Chinese; No Copyright; Avail: Issuing Activity, Hardcopy, Microfiche

Picture, image and symbolized measurements in aircraft and spacecraft instruments are discussed. A design idea of intelligent aircraft and spacecraft instruments is presented in order to solve the troubling and difficult problems of the general design of instrument panels in the last thirty years: the more the number of required display parameters are increased, the heavier the pilot's duties would be.

Author

Images; Aircraft Instruments; Spacecraft Instruments; Symbols

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.

19980231996 NASA Lewis Research Center, Cleveland, OH USA
Application of Gas Analysis to Combustor Research
Hibbard, R. R., NASA Lewis Research Center, USA; Evans, Albert, NASA Lewis Research Center, USA; Feb, 1959; 24p; In English
Report No.(s): NASA-MEMO-1-26-59E; E-245; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The performance of turbine-engine combustors usually is given in terms of operating limits and combustion efficiency. The latter property is determined most often by measuring the increase in enthalpy across the combustor through the use of thermocouples. This investigation was conducted to determine the ability of gas-analytical techniques to provide additional information about combustor performance. Gas samples were taken at the outlet and two upstream stations and their compositions determined. In addition to over-all combustion efficiency, estimates of local fuel-air ratios, local combustion efficiencies, and heat-release rates can be made. Conclusions can be drawn concerning the causes of combustion inefficiency and may permit corrective design changes to be made more intelligently. The purpose of this investigation was not to present data for a given combustor but rather to show the types and value of additional information that can be gained from gas-analytical data.
Author
Fuel Combustion; Gas Analysis; Combustion Efficiency; Heat Transfer; Combustion Chambers

19980232083 NASA Lewis Research Center, Cleveland, OH USA
Structural Design and Preliminary Evaluation of a Lightweight, Brazed, Air-Cooled Turbine Rotor Assembly
Meyer, Andre J., Jr., NASA Lewis Research Center, USA; Morgan, William C., NASA Lewis Research Center, USA; Dec. 1958; 28p; In English
Report No.(s): NASA-MEMO-10-5-58E; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A lightweight turbine rotor assembly was devised, and components were evaluated in a full-scale jet engine. Thin sheet-metal airfoils were brazed to radial fingers that were an integral part of a number of thin disks composing the turbine rotor. Passages were provided between the disks and in the blades for air cooling. The computed weight of the assembly was 50 percent less than that of a similar turbine of normal construction used in a conventional turbojet engine. Two configurations of sheet-metal test blades simulating the manner of attachment were fabricated and tested in a turbojet engine at rated speed and temperature. After 8-1/2 hours of operation pieces broke loose from the tip sections of the better blades. Severe cracking produced by vibration was determined as the cause of failure. Several methods of overcoming the vibration problem are suggested.
Author
Structural Design; Turbines; Air Cooling; Turbojet Engines; Brazing

19980232149 Allison Engine Co., Indianapolis, IN USA
Energy Efficient Engine Low Pressure Subsystem Flow Analysis *Final Report, Mar. 1996 - Sep. 1997*
Hall, Edward J., Allison Engine Co., USA; Lynn, Sean R., Allison Engine Co., USA; Heidegger, Nathan J., Allison Engine Co., USA; Delaney, Robert A., Allison Engine Co., USA; Mar. 1998; 174p; In English; Original contains color illustrations
Contract(s)/Grant(s): NAS3-27394; RTOP 509-10-11
Report No.(s): NASA/CR-1998-206597; NAS 1.26:206597; E-11067; No Copyright; Avail: CASI; A08, Hardcopy; A02, Microfiche

The objective of this project is to provide the capability to analyze the aerodynamic performance of the complete low pressure subsystem (LPS) of the Energy Efficient Engine (EEE). The analyses were performed using three-dimensional Navier-Stokes numerical models employing advanced clustered processor computing platforms. The analysis evaluates the impact of steady aerodynamic interaction effects between the components of the LPS at design and off-design operating conditions. Mechanical coupling is provided by adjusting the rotational speed of common shaft-mounted components until a power balance is achieved. The Navier-Stokes modeling of the complete low pressure subsystem provides critical knowledge of component aero/mechanical interactions that previously were unknown to the designer until after hardware testing.
Author
Turbofans; Mathematical Models; Navier-Stokes Equation; Three Dimensional Models; Computational Fluid Dynamics; Aerodynamic Characteristics; Computerized Simulation; Turbofan Engines; Rotor Aerodynamics; Flow Characteristics

19980232156 General Accounting Office, National Security and International Affairs Div., Washington, DC USA
Report to the Honorable Walter B. Jones, House of Representatives. Defense Depot Maintenance: Weaknesses in the T406 Engine Logistics Support Decision Methodology
Sep. 1998; 26p; In English
Report No.(s): GAO/NSIAD-98-221; B-280621; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The Navy's designation of the V-22 (also called Osprey) aircraft's T406 engine as a commercial item, which allows the logistics support and maintenance to be performed by a contractor, while not unjustifiable was, according to the report, not very well documented and somewhat inconsistent. The report faults the Navy for relying on the contractor's assurance that 90 percent of the parts in the engine were common parts, while this Agency's findings were that 79% of the parts were common to at least one of the other engines that the company makes. The report also finds that the Navy made some errors in the cost savings that would be realized by the use of the commercial maintenance contractor rather than establishing maintenance capability in a military depot.
CASI
Cost Reduction; Navy; V-22 Aircraft; Aircraft Maintenance; Maintainability; Spare Parts; Congressional Reports

19980232222 NASA Lewis Research Center, Cleveland, OH USA
Factors that Affect Operational Reliability of Turbojet Engines, Chapter 1, Objectives
Pinkel, Benjamin, NASA Lewis Research Center, USA; 1960; 216p; In English
Report No.(s): NASA-TR-R-54; No Copyright; Avail: CASI; A10, Hardcopy; A03, Microfiche

The problem of improving operational reliability of turbojet engines is studied. Failure statistics for this engine are presented, the theory and experimental evidence on how engine failures occur are described, and the methods available for avoiding failure in operation are discussed.
Author
Engine Failure; Reliability; Turbojet Engines

19980232227 NASA, Washington, DC USA
Control of Gas-Turbine and Ramjet Engines
Zalmanzon, L. A.; Cherkasov, B. A.; Jul. 1961; 257p; In English; Translated into English by the Government Defense Industry Press, Moscow, 1956
Report No.(s): NASA-TT-F-41; No Copyright; Avail: CASI; A12, Hardcopy; A03, Microfiche

This book deals with the principles of control of aviation gas turbine and ramjet engines. Attention is concentrated on describing the physical bases of the processes of engine control and an exposition of the methods of experimental study and design of control apparatus. A series of examples bring out the close connection between the operation of the elements in the control loop and the fuel supply system of the engines. The characteristics of the several elements of the fuel apparatus are given. The book is written as a textbook for students in Aviation Institutes, but may also be useful for workers specializing in the field of aviation engines.
Author
Engine Control; Gas Turbine Engines; Fuel Systems; Ramjet Engines

19980235627 NASA Lewis Research Center, Cleveland, OH USA
Internal-Performance Evaluation of Two Fixed-Divergent-Shroud Ejectors
Mihaloew, James R., NASA Lewis Research Center, USA; 1960; 34p; In English
Report No.(s): NASA-TM-X-257; E-691; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Ejectors designed for use in a Mach 2.2 aircraft were evaluated over a range of representative primary pressure ratios and ejector corrected weight-flow ratios. Basic thrust and pumping characteristics are discussed in terms of an assumed engine operating schedule to illustrate the variation of performance with Mach number. The two designs differed about 16 percent in the shroud longitudinal spacing ratio. For corrected ejector weight-flow ratios up to 0.10, the performance of the fixed-shroud ejector designs

is comparable with that of a similar continuously variable ejector except at conditions corresponding to acceleration with afterburning from Mach 0.4 to 1.2. In this region, the ejector thrust ratio decreased to a minimum of 0.96.
Author
Performance Tests; Pressure Ratio; Afterburning

19980236452 NASA Lewis Research Center, Cleveland, OH USA
Internal Performance Evaluation of a Two Position Divergent Shroud Ejector
Mihaloew, James R., NASA Lewis Research Center, USA; Stofan, Andrew J., NASA Lewis Research Center, USA; 1960; 24p; In English
Report No.(s): NASA-TM-X-258; E-753; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A two-position divergent shroud ejector was investigated in an unheated quiescent-air facility over a range of operational variables applicable to a Mach 2.5 aircraft. The performance data are shown in terms of hypothetical engine operating conditions to illustrate variations of performance with Mach number. The overall thrust performance was reasonably good, with ejector thrust ratios ranging from 0.97 to 0.98 for all conditions except that corresponding to acceleration with afterburning through the transonic flight Mach number region from 0.9 to 1.1, where the ejector thrust ratio decreased to as low as 0.945 for an ejector corrected weight-flow ratio of 0.105.
Author
Shrouds; Ejectors; Afterburning; Performance Tests; Transonic Flight; Thrust

19980236663 NASA Langley Research Center, Hampton, VA USA
Static Performance of a Wing-Mounted Thrust Reverser Concept
Asbury, Scott C., NASA Langley Research Center, USA; Yetter, Jeffrey A., NASA Langley Research Center, USA; 1998; 19p; In English; 34th; Propulsion, 13-15 Jul. 1998, Cleveland, OH, USA; Sponsored by American Inst. of Aeronautics and Astronautics, USA
Report No.(s): AIAA Paper 98-3256; No Copyright; Avail: Issuing Activity, Hardcopy, Microfiche

An experimental investigation was conducted in the Jet-Exit Test Facility at NASA Langley Research Center to study the static aerodynamic performance of a wing-mounted thrust reverser concept applicable to subsonic transport aircraft. This innovative engine powered thrust reverser system is designed to utilize wing-mounted flow deflectors to produce aircraft deceleration forces. Testing was conducted using a 7.9%-scale exhaust system model with a fan-to-core bypass ratio of approximately 9.0, a supercritical left-hand wing section attached via a pylon, and wing-mounted flow deflectors attached to the wing section. Geometric variations of key design parameters investigated for the wing-mounted thrust reverser concept included flow deflector angle and chord length, deflector edge fences, and the yaw mount angle of the deflector system (normal to the engine centerline or parallel to the wing trailing edge). All tests were conducted with no external flow and high pressure air was used to simulate core and fan engine exhaust flows. Test results indicate that the wing-mounted thrust reverser concept can achieve overall thrust reverser effectiveness levels competitive with (parallel mount), or better than (normal mount) a conventional cascade thrust reverser system. by removing the thrust reverser system from the nacelle, the wing-mounted concept offers the nacelle designer more options for improving nacelle aerodynamics and propulsion-airframe integration, simplifying nacelle structural designs, reducing nacelle weight, and improving engine maintenance access.
Author
Engine Airframe Integration; Deceleration; Deflectors; Design Analysis; Exhaust Systems; Nacelles; Structural Design; Supercritical Wings; Thrust; Thrust Reversal; Weight Reduction; Wings

19980236700 ADA Technologies, Inc., Englewood, CO USA
Evaluation of Pilot-Scale Pulse-Corona-Induced Plasma Device to Remove NOx from Combustion Exhausts from a Subscale Combustor and from a Hush House at Nellis AFB, Nevada ***Final Report, Aug. 1994 - Jan. 1997***
Haythornthwaite, Sheila M., ADA Technologies, Inc., USA; Durham, Michael D., ADA Technologies, Inc., USA; Anderson, Gary L., ADA Technologies, Inc., USA; Rugg, Donald E., ADA Technologies, Inc., USA; May 01, 1997; 83p; In English
Contract(s)/Grant(s): F08637-94-C-6047; AF Proj. 2103
Report No.(s): AD-A352365; ADA-R439198F01; AL/EQ-TR-1997-0022; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

Jet engine test cells (JETCs) are used to test-fire new, installed, and reworked jet engines. Because JETCs have been classified as stationary sources of pollutant emissions, they are subject to possible regulation under Title I of the Clean Air Act (CAA) as amended in 1990. In Phase I of the Small Business Innovation Research (SBIR) program, a novel NOx-control approach utilizing pulsed-corona-induced plasma successfully showed 90% removal of NOx in the laboratory. The objective of Phase II was to repro-

duce the laboratory-scale results in a pilot-scale system. The technology was successfully demonstrated at pilot scale in the field, on a slipstream of JETC flue gas at Nellis Air Force Base. Based on the field data, cost projections were made for a system to treat the full JETC exhaust. The technology efficiently converted NO into ONO, and a wet scrubber was required to achieve the treatment goal of 50-percent removal and destruction of NOx. The plasma simultaneously removes hydrocarbons from the flue gas stream. This project demonstrated that pulse-corona-induced plasma technology is scalable to practical industrial dimensions.
DTIC
Air Pollution; Contaminants; Air Quality; Exhaust Emission; Nitrogen Oxides; Plasmas (Physics)

19980236867 NASA Lewis Research Center, Cleveland, OH USA
Challenges to Laser-Based Imaging Techniques in Gas Turbine Combustor Systems for Aerospace Applications
Locke, Randy J., DYNACS Engineering Co., Inc., USA; Anderson, Robert C., NASA Lewis Research Center, USA; Zaller, Michelle M., NASA Lewis Research Center, USA; Hicks, Yolanda R., NASA Lewis Research Center, USA; Sep. 1998; 14p; In English; 20th; Advanced Measurement and Ground Testing Technology Conference, 15-18 Jun. 1998, Albuquerque, NM, USA; Sponsored by American Inst. of Aeronautics and Astronautics, USA; Original contains color illustrations
Contract(s)/Grant(s): RTOP 537-05-20
Report No.(s): NASA/TM-1998-208649; NAS 1.15:208649; AIAA Paper 98-2778; E-11368; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Increasingly severe constraints on emissions, noise and fuel efficiency must be met by the next generation of commercial aircraft powerplants. At NASA Lewis Research Center (LeRC) a cooperative research effort with industry is underway to design and test combustors that will meet these requirements. To accomplish these tasks, it is necessary to gain both a detailed understanding of the combustion processes and a precise knowledge of combustor and combustor sub-component performance at close to actual conditions. to that end, researchers at LeRC are engaged in a comprehensive diagnostic investigation of high pressure reacting flowfields that duplicate conditions expected within the actual engine combustors. Unique, optically accessible flame-tubes and sector rig combustors, designed especially for these tests. afford the opportunity to probe these flowfields with the most advanced, laser-based optical diagnostic techniques. However, these same techniques, tested and proven on comparatively simple bench-top gaseous flame burners, encounter numerous restrictions and challenges when applied in these facilities. These include high pressures and temperatures, large flow rates, liquid fuels, remote testing, and carbon or other material deposits on combustor windows. Results are shown that document the success and versatility of these nonintrusive optical diagnostics despite the challenges to their implementation in realistic systems.
Author
Gas Turbines; Liquid Fuels; Flow Velocity; Flow Distribution; Flames; Commercial Aircraft; Component Reliability; Combustion Chambers

19980236890 NASA Lewis Research Center, Cleveland, OH USA
Optical Fuel Injector Patternation Measurements in Advanced Liquid-Fueled, High Pressure, Gas Turbine Combustors
Locke, R. J., NYMA, Inc., USA; Hicks, Y. R., NASA Lewis Research Center, USA; Anderson, R. C., NASA Lewis Research Center, USA; Zaller, M. M., NASA Lewis Research Center, USA; Feb. 13, 1998; 18p; In English
Contract(s)/Grant(s): RTOP 537-05-20; RTOP 538-17-10; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Planar laser-induced fluorescence (PLIF) imaging and planar Mie scattering are used to examine the fuel distribution pattern (patternation) for advanced fuel injector concepts in kerosene burning, high pressure gas turbine combustors. Three fuel injector concepts for aerospace applications were investigated under a broad range of operating conditions. Fuel PLIF patternation results are contrasted with those obtained by planar Mie scattering. For one injector, further comparison is also made with data obtained through phase Doppler measurements. Differences in spray patterns for diverse conditions and fuel injector configurations are readily discernible. An examination of the data has shown that a direct determination of the fuel spray angle at realistic conditions is also possible. The results obtained in this study demonstrate the applicability and usefulness of these nonintrusive optical techniques for investigating fuel spray patternation under actual combustor conditions.
Author
Fuel Injection; Injectors; Gas Turbines; Fuel Sprays; Combustion Chambers

08
AIRCRAFT STABILITY AND CONTROL

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

19980231990 NASA Ames Research Center, Moffett Field, CA USA

A Wind-Tunnel Investigation of the Stall-Flutter Characteristics of a Supersonic-Type Propeller at Positive and Negative Thrust

Rogallo, Vernon L., NASA Ames Research Center, USA; Yaggy, Paul F., NASA Ames Research Center, USA; May 1959; 68p; In English

Report No.(s): NASA-MEMO-3-9-59A; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

An investigation has been made utilizing a three-blade, 10-foot- diameter, supersonic-ty-pe propeller to determine propeller flutter characteristics. The particular flutter characteristics of interest were (1) the effect of stall flutter on a propeller operating in positive and negative thrust, (2) the effect of stall flutter on a propeller operating with the thrust axis inclined, and (3) the variation of vibratory blade shear stresses as the stall flutter boundary is penetrated and exceeded. Thrust and power measurements were made for all test conditions. Wake and inflow surveys were made when appropriate, to define the thrust and torque distributions and the magnitude of the inflow velocity. Stress measurements were made simultaneously to obtain the propeller flutter and bending response. It was found when operating both in the positive and negative thrust regions that, for most cases after the onset of flutter, the magnitude of the flutter stresses at first increased rapidly with section blade angle P, after which further increases in 0 resulted in only a moderate increase or a reduction in stress. Thrust-axis inclination up to the limit of the tests (angle of attack of 15 deg and dynamic pressure of 40 psf) appeared to have no effect on stall flutter. The stall flutter stresses were found to be directly associated with the section thrust characteristics of the blades. The onset of flutter was found to occur simultaneously with the divergence of the section thrust variation with blade angle from linearity for stations outboard of the blade 0.8-radius station. The maximum flutter stresses appeared to be a function of the maximum section thrust obtained at or in the vicinity of the blade 0.8-radius station. In an attempt to correlate two-dimensional airfoil data with three-dimensional data to predict the stall angle of attack (divergence of the section thrust) of the blade sections, it was found that no consistent correlation could be obtained. Also, a knowledge of the inflow conditions appeared to be insufficient to account for differences in airfoil characteristics between the two-dimensional and the three-dimensional cases.

Author

Angle of Attack; Dynamic Pressure; Flutter Analysis; Stress Measurement; Shear Stress; Boundaries; Aerodynamic Stalling; Airfoils

19980231993 NASA Langley Research Center, Hampton, VA USA

An Investigation of the Performance of Various Reaction Control Devices

Hunter, Paul A., NASA Langley Research Center, USA; Mar. 1959; 42p; In English

Report No.(s): NASA-MEMO-2-11-59L; L-160; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation of a small-scale reaction control devices in still air with both subsonic and supersonic internal flows has shown that lateral forces approaching 70 percent of the resultant force of the undeflected jet can be obtained. These results were obtained with a tilted extension at a deflection of 40 deg. The tests of tilted extensions indicated an optimum length-to-diameter ratio of approximately 0.75 to 1.00, dependent upon the deflection angle. For the two geometric types of spoiler tabs tested, blockage-area ratio appears to be the only variable affecting the lateral force developed. Usable values of lateral force were developed by the full-eyelid type of device with reasonably small losses in the thrust and weight flow. Somewhat larger values of lateral force were developed by injecting a secondary flow normal to the primary jet, but for conditions of these tests the losses in thrust and weight flow were large. Relatively good agreement with other investigations was obtained for several of the devices. The agreement of the present results with those of an investigation made with larger-scale equipment indicates that Reynolds number may not be critical for these tests. In as much as the effects of external flow could influence the performance and other factors affecting the choice of a reaction control for a specific use, it would appear desirable to make further tests of the devices described in this report in the presence of external flow.

Author

Control Systems Design; Tabs (Control Surfaces); Spoilers; Secondary Flow; Reynolds Number; Internal Flow; Control Surfaces

19980231998 NASA Langley Research Center, Hampton, VA USA

Effect of Tail Dihedral on Lateral Control Effectiveness at High Subsonic Speeds of Differentially Deflected Horizontal-Tail Surfaces on a Configuration having a Thin Highly Tapered Wing

Fournier, Paul G., NASA Langley Research Center, USA; Jan. 1959; 52p; In English

Report No.(s): NASA-MEMO-12-1-58L; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

Tests have been conducted in the Langley high-speed 7- by 10-foot tunnel to determine the effect of tail dihedral on lateral control effectiveness of a complete-model configuration having differentially deflected horizontal-tail surfaces. Limited tests were made to determine the lateral characteristics as well as the longitudinal characteristics in sideslip. The wing had an aspect ratio of 3, a taper ratio of 0.14, 28.80 deg sweep of the quarter-chord line with zero sweep at the 80-percent-chord line, and NACA 65A004 airfoil sections. The test Mach number range extended from 0.60 to 0.92. There are only small variations in the roll effectiveness parameter C(sub iota delta) with negative tail dihedral angle. The tail size used on the test model, however, is perhaps inadequate for providing the roll rates specified by current military requirements at subsonic speeds. The lateral aerodynamic characteristics were essentially constant throughout the range of sideslip angle from 12 deg to -12 deg. A general increase in yawing moment was noted with increased negative dihedral throughout the Mach number range.

Author

Tapering; Thin Wings; Horizontal Tail Surfaces; Aerodynamic Characteristics; Controllability; Swept Wings

19980232000 NASA Langley Research Center, Hampton, VA USA

Longitudinal and Lateral Stability and Control Characteristics of Various Combinations of the Component Parts of Two Canard Airplane Configurations at Mach Numbers of 1.41 and 2.01

Driver, Cornelius, NASA Langley Research Center, USA; Oct. 1958; 108p; In English

Report No.(s): NASA-MEMO-10-1-58L; No Copyright; Avail: CASI; A06, Hardcopy; A02, Microfiche

An investigation has been conducted in the Langley 4- by 4-foot supersonic pressure tunnel to determine the aerodynamic characteristics in pitch and sideslip of a generalized canard airplane model. Two wings of equal area but differing in plan form were investigated. The model was equipped with a trapezoidal canard surface with an area 12 percent of the wing area, a low-aspect-ratio vertical tail, and twin ventral fins. The interference effects of the canard wake on the wing result in little or no gain in the total lift at a Mach number of 1.41 but at a Mach number of 2.01 a substantial portion of the canard lift is retained with a resultant increase in total lift. Because these interference effects of the canard wake appear to be concentrated near the leading edge of the wing, the proper location of the wing leading edge with respect to the center of moments may result in a substantial increase in the moment increment provided by a canard surface even though the total lift provided by the canard is small. For these configurations the trapezoidal wing retained the most lift and had the largest favorable moment increment produced by the canards. The canard configurations have the same characteristic decrease in directional-stability with angle of attack as most conventional high-fineness-ratio supersonic configurations. Although the presence of the canard surface caused a small increase in the directional stability at a Mach number of 1.41 for the delta-wing configuration, the presence of the canards resulted in small decreases in the directional-stability level at a Mach number of 2.01 for both wing configurations. A canard deflection of 15 deg provides an increase in the positive effective dihedral approximately as large as that provided by the presence of the vertical tail. This effect of canard deflection might complicate the lateral-control problem in the case of a rolling pull-up maneuver.

Author

Aerodynamic Characteristics; Aerodynamic Configurations; Longitudinal Stability; Lateral Stability; Aircraft Models; Canard Configurations

19980232001 NASA Langley Research Center, Hampton, VA USA

A Transonic Wind-Tunnel Investigation of the Performance and of the Static Stability and Control Characteristics of a Model of a Fighter-Type Airplane which Embodies Partial Body Indentation

Bielat, Ralph P., NASA Langley Research Center, USA; Mar. 1959; 136p; In English

Report No.(s): NASA-MEMO-12-13-58L; L-476; No Copyright; Avail: CASI; A07, Hardcopy; A02, Microfiche

An investigation was conducted to obtain the aerodynamic characteristics of a model of a fighter-type airplane embodying partial body indentation. The wing had an aspect ratio of 4, taper ratio of 0.5, 35 deg sweepback of the 0.25-chord line, and a modified NACA 65A006 airfoil section at the root and a modified NACA 65A004 airfoil section at the tip. The fuselage has been indented in the region of the wing in order to obtain a favorable area distribution. The results reported herein consist of the performance and of the static longitudinal and lateral stability and control characteristics of the complete model. The Mach number range extended from 0.60 to 1.13, and the corresponding Reynolds number based on the wing mean aerodynamic chord varied from 1.77 x 10(exp 6) to 2.15 x 10(exp 6). The drag rise for both the cambered leading edge and symmetrical wing sections occurred at a Mach number of 0.95. Certain local modifications to the body which further improved the distribution of cross-sectional area gave additional reductions in drag at a Mach number of 1.00. The basic configuration indicated a mild pitch-up tendency at lift coefficients near 0.70 for the Mach number range from 0.80 to 0.90; however, the pitch-up instability may not be too objectionable on the basis of dynamic-stability considerations. The basic configuration indicated positive directional stability and positive effec-

tive dihedral through the angle-of-attack range and Mach number range with the exception of a region of negative effective dihedral at low lifts at Mach numbers of 1.00 and slightly above.
Author
Aerodynamic Characteristics; Longitudinal Stability; Directional Stability; Static Stability; Lateral Stability; Fighter Aircraft

19980232016 NASA Ames Research Center, Moffett Field, CA USA
A Flight Investigation to Determine the Lateral Oscillatory Damping Acceptable for an Airplane in the Landing Approach
McNeill, Walter E., NASA Ames Research Center, USA; Vomaske, Richard F., NASA Ames Research Center, USA; Feb. 1959; 34p; In English
Report No.(s): NASA-MEMO-12-10-58A; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An F-86E airplane, in which servo actuation of the ailerons and rudder provides artificial variation of the important lateral and directional aerodynamic stability parameters, has been flown by test pilots of the NASA, U.S. Air Force, and one aircraft manufacturer to determine satisfactory and acceptable levels of lateral oscillatory damping in the landing approach. In addition to normal operational use, particular consideration was given to the emergency condition of failure of stability-augmentation equipment. In this study, the pilots' opinions of the airplane dynamic stability and control characteristics in smooth and simulated rough air have been recorded according to a numerical rating scale. The results are presented in the form of boundaries in terms of cycles to damp to half amplitude, 1/C(sub 1/2), or time to damp to half amplitude, 1/T(1/2) and bank-to-sideslip ratio, and are discussed in relation to existing flying-qualities criteria. Though the present results, which were obtained at 170 knots indicated airspeed and 10,000-feet altitude, indicated that increased damping is required with increased bank-to-sideslip ratio (as found in previous work), consideration of the dampers-failed condition indicated a great reduction in the minimum acceptable damping. At moderate values of bank-to-sideslip ratio, effects of lateral-oscillation period on pilot-opinion variation with damping appeared to be taken into account by use of the parameter 1/T(sub 1/2).
Author
Flight Characteristics; Lateral Oscillation; Aerodynamic Stability; Approach and Landing Tests (STS); Directional Stability; Dynamic Stability

19980232025 NASA Langley Research Center, Hampton, VA USA
Static Longitudinal Characteristics at High Subsonic Speeds of a Complete Airplane Model with a Highly Tapered Wing having the 0.80 Chord Line Unswept and with Several Tail Configurations
Goodson, Kenneth W., NASA Langley Research Center, USA; Aug. 1961; 62p; In English
Report No.(s): NASA-TN-D-949; L-1699; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

An investigation was made at high subsonic speeds of a complete model having a highly tapered wing and several tail configurations. The basic aspect-ratio-4.00 wing had zero taper and an unswept 0.80 chord line. Several aspect-ratio modifications to the basic wing were made by clipping off portions of the wing tips. The complete model was tested with a chord-plane tail, a T-tail, and a biplane tail (combined T-tail and chord-plane tail). The model was tested in the Langley high-speed 7- by 10-foot tunnel at Mach numbers from 0.60 to 0.92. The data show that, when reduced to the same static margin, all the tail configurations tested on the model provided fairly good stability characteristics, the biplane tail giving the.best overall characteristics as regards pitching-moment linearity. Changes in static margin at zero lift coefficient with Mach number were small for the model with these tails over the Mach number range investigated.
Author
Static Characteristics; Aircraft Models; Zero Lift; Aspect Ratio; Aerodynamic Coefficients

19980232027 Air Force Flight Test Center, Edwards AFB, CA USA
Limited Handling Qualities Evaluation of Longitudinal Flight Control Systems Designed Using Multiobjectives Control Design Techniques (HAVE INFINITY 2) ***Final Report, Jul - Dec. 1997***
Anderson, John R.; Spillman, Mark S.; Boe, Eric A.; Stephens, Michael J.; Cantiello, Maurizio; Jan. 1998; 82p; In English
Report No.(s): AD-A351011; AFFTC-TR-97-48; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

This report presents the flight test results of the Project HAVE INFINITY II limited flight test. The objective of this limited flight test was to evaluate the six HAVE INFINITY II longitudinal flight control designs in support of an Air Force Institute of Technology (AFIT) Master's degree thesis. The thesis investigates the practicality of using modern multiobjective techniques for flight control system design. During the test program. 12 evaluation sorties, totaling 14.2 flight hours, were flown in a Calspan

Variable Stability System (VSS) Learjet. Tests were conducted by the USAF Test Pilot School, Edwards AFB, California, from 29 September through 10 October 1997, at the request of the AFIT, Wright-Patterson AFB, Ohio.
DTIC
Control Systems Design; Flight Control; Lear Jet Aircraft; Longitudinal Control

19980232080 NASA Ames Research Center, Moffett Field, CA USA
A Flight Investigation of the Low-Speed Handling Qualities of a Tailless Delta-Wing Fighter Airplane
White, Maurice D., NASA Ames Research Center, USA; Innis, Robert C., NASA Ames Research Center, USA; May 1959; 36p; In English
Report No.(s): NASA-MEMO-4-15-59A; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Carrier landing-approach studies of a tailless delta-wing fighter airplane disclosed that approach speeds were limited by ability to control altitude and lateral-directional characteristics. More detailed flight studies of the handling-qualities characteristics of the airplane in the carrier-approach configuration documented a number of factors that contributed to the adverse comments on the lateral-directional characteristics. These were: (1) the tendency of the airplane to roll around the highly inclined longitudinal axis, so that significant sideslip angles developed in the roll as a result only of kinematic effects; (2) reduction of the rolling response to the ailerons because of the large dihedral effect in conjunction with the kinematically developed sideslip angles; and (3) the onset of rudder lock at moderate angles of sideslip at the lowest speeds with wing tanks installed. The first two of the factors listed are inseparably identified with this type of configuration which is being considered for many of the newer designs and may, therefore, represent a problem which will be encountered frequently in the future. The results are of added significance in the demonstration of a typical situation in which extraneous factors occupy so much of the pilot's attention that his capability of coping with the problems of precise flight-path control is reduced, and he accordingly demands a greater speed margin above the stall to allow for airspeed fluctuations.
Author
Fighter Aircraft; Delta Wings; Low Speed; Controllability; Flight Paths

19980232086 NASA Ames Research Center, Moffett Field, CA USA
Longitudinal Stability and Control Characteristics at Mach Numbers from 0.70 to 2.22 of a Triangular Wing Configuration Equipped with a Canard Control, a Trailing-Edge-Flap Control or a Cambered Forebody
Boyd, John W., NASA Ames Research Center, USA; Menees, Gene P., NASA Ames Research Center, USA; Apr. 1959; 42p; In English
Report No.(s): NASA-MEMO-4-21-59A; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Results of an investigation to determine the static longitudinal stability and control characteristics of an aspect-ratio-2 triangular wing and body configuration equipped with either a canard control, a trailing-edge-flap control, or a cambered forebody are presented without analysis for Mach numbers from 0.70 to 2.22. The canard surface had a triangular plan form and a ratio of exposed area to total wing area of 7.8 percent. The hinge line of the canard was in the extended wing chord plane, 0.83 wing mean aerodynamic chord ahead of the reference center of moments. The trailing-edge controls were constant-chord full-span flaps with exposed area equal to 10.7 percent of the total wing area. The cambered body was a modified Sears-Haack body with camber only ahead of the wing apex. Data are presented for various canard and flap deflections at angles of attack ranging from -6 deg to +18 deg.
Author
Aerodynamic Configurations; Aircraft Control; Forebodies; Longitudinal Stability; Canard Configurations; Mach Number; Trailing Edges

19980232220 NASA Langley Research Center, Hampton, VA USA
Static Lateral Characteristics at High Subsonic Speeds of a Complete Airplane Model with a Highly Tapered Wing having the 0.80 Chord Line Unswept and with Several Tail Configurations
Goodson, Kenneth W., NASA Langley Research Center, USA; Aug. 1961; 102p; In English
Report No.(s): NASA-TN-D-950; L-1703; No Copyright; Avail: CASI; A06, Hardcopy; A02, Microfiche

An investigation was made at high subsonic speeds of a complete model having a highly tapered wing and several tail configurations. The aspect-ratio-3.50 wing had a taper ratio of 0.067 and an unswept 0.80 chord line. The complete model was tested with a wing-chord-plane tail, a T-tail, and a biplane tail (combined T-tail and wing-chord-plane tail). The model was tested in the Langley high-speed 7- by 10-foot tunnel at Mach numbers from 0.60 to 0.92 over a range of angle of attack of about +/- 20 deg. and a range of sideslip of -15 deg. to 13 deg. Some data were obtained with the horizontal stabilizer deflected. A few tests were also made with the wing tips clipped to an aspect ratio of 3.00. The data show that shock-interference effects between the tail sur-

faces (T-tail) can have considerable influence on the directional stability and effective dihedral. For example, the T-tail configuration with horizontal-tail leading-edge overhang showed a considerable loss in directional stability as the angle of attack was reduced to zero or negative values; whereas, the T-tail with zero leading-edge overhang showed the loss to be considerably less. The directional stability of the model with the low tail was essentially constant over a range of angle of attack of +/- 50 deg. All configurations tested showed a large reduction in stability at positive and negative angles of attack larger than about 15 deg., probably because of adverse sidewash associated with wing stall. The data show that a wing-chord-plane horizontal tail (low tail) tends to give a positive pitching-moment increment with increase in sideslip angle; whereas, a high tail (T-tail) tends to give negative increments in pitching moment.
Author
Static Characteristics; Swept Wings; Tail Surfaces; Wing Tips; Pitching Moments; Leading Edges; Directional Stability; Dihedral Angle; Aspect Ratio; Angle of Attack

19980232887 NASA Langley Research Center, Hampton, VA USA
A Method for Integrating Thrust-Vectoring and Actuated Forebody Strakes with Conventional Aerodynamic Controls on a High-Performance Fighter Airplane
Lallman, Frederick J., NASA Langley Research Center, USA; Davidson, John B., NASA Langley Research Center, USA; Murphy, Patrick C., NASA Langley Research Center, USA; Sep. 1998; 44p; In English
Contract(s)/Grant(s): RTOP 522-22-21-03
Report No.(s): NASA/TP-1998-208464; NAS 1.60:208464; L-17627; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A method, called pseudo controls, of integrating several airplane controls to achieve cooperative operation is presented. The method eliminates conflicting control motions, minimizes the number of feedback control gains, and reduces the complication of feedback gain schedules. The method is applied to the lateral/directional controls of a modified high-performance airplane. The airplane has a conventional set of aerodynamic controls, an experimental set of thrust-vectoring controls, and an experimental set of actuated forebody strakes. The experimental controls give the airplane additional control power for enhanced stability and maneuvering capabilities while flying over an expanded envelope, especially at high angles of attack. The flight controls are scheduled to generate independent body-axis control moments. These control moments are coordinated to produce stability-axis angular accelerations. Inertial coupling moments are compensated. Thrust-vectoring controls are engaged according to their effectiveness relative to that of the aerodynamic controls. Vane-relief logic removes steady and slowly varying commands from the thrust-vectoring controls to alleviate heating of the thrust turning devices. The actuated forebody strakes are engaged at high angles of attack. This report presents the forward-loop elements of a flight control system that positions the flight controls according to the desired stability-axis accelerations. This report does not include the generation of the required angular acceleration commands by means of pilot controls or the feedback of sensed airplane motions.
Author
Thrust Vector Control; Fighter Aircraft; Flight Control; Feedback Control; Directional Control; Control Stability; Angular Acceleration; Angle of Attack; F-18 Aircraft

19980232889 NASA Langley Research Center, Hampton, VA USA
Effect of Wing Thickness and Sweep on the Oscillating Hinge-Moment and Flutter Characteristics of a Flap-Type Control at Transonic Speeds
Moseley, William C., Jr., NASA Langley Research Center, USA; Gainer, Thomas G., NASA Langley Research Center, USA; Oct. 1959; 42p; In English
Report No.(s): NASA-TM-X-123; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Free-oscillation tests were made in the Langley high-speed 7- by 10-foot tunnel to determine the effects of wing thickness and wing sweep on the hinge-moment and flutter characteristics of a trailing-edge flap-type control. The untapered semispan wings had full-span aspect ratios of 5 and NACA 65A-series airfoil sections. Unswept wings having ratios of wing thickness to chord of 0.04, 0.06, 0.08, and 0.10 were investigated. The swept wings were 6 percent thick and had sweep angles of 30 deg and 45 deg. The full-span flap-type controls had a total chord of 50 percent of the wing chord and were hinged at the 0.765-wing-chord line. Tests were made at zero angle of attack over a Mach number range from 0.60 to 1.02, control oscillation amplitudes up to about 12 deg, and a range of control-reduced frequencies. Static hinge-moment data were also obtained. Results indicate that the control aerodynamic damping for the 4-percent-thick wing-control model was unstable in the Mach number range from 0.92 to 1.02 (maximum for these tests). Increasing the ratio of wing thickness to chord to 0.06, 0.08, and then to 0.10 had a stabilizing effect on the aerodynamic damping in this speed range so that the aerodynamic damping was stable for the 10-percent-thick model at all Mach numbers. The 6-percent-thick unswept-wing-control model generally had unstable aerodynamic damping in the Mach

number range from 0.96 to 1.02. Increasing the wing sweep resulted in a general decrease in the stable aerodynamic damping at the lower Mach numbers and in the unstable aerodynamic damping at the higher Mach numbers. The one-degree-of-freedom control-surface flutter which occurred in the transonic Mach number range (0.92 to 1.02) for the 4-, 6-, and 8-percent-thick unswept-wing-control models could be eliminated by further increasing the ratio of thickness to chord to 0.10. Flutter could also be eliminated by increasing the wing sweep angle to either 30 deg or 45 deg. The magnitude of variation in spring moment derivative with Mach number at transonic speeds was decreased by either increasing the ratio of wing thickness to chord or increasing the wing sweep angle.
Author
Flaps (Control Surfaces); Transonic Speed; Swept Wings; Thickness; Oscillations; Flutter Analysis; Trailing Edge Flaps; Aircraft Control; Control Surfaces

19980232925 NASA Langley Research Center, Hampton, VA USA
Effect of the Proximity of the Wing First-Bending Frequency and the Short-Period Frequency on the Airplane Dynamic-Response Factor
Huss, Carl R., NASA Langley Research Center, USA; Donegan, James J., NASA Langley Research Center, USA; 1959; 22p; In English
Report No.(s): NASA-TR-R-12; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The results are presented in the form of preliminary design charts which give a comparison between the dynamic-response factors of the semi-rigid case and the airplane longitudinal short-period case and between the dynamic-response factors of the semi-rigid case and the steady-state value of the airplane longitudinal short-period response. These charts can be used to estimate the first-order effects of the addition of a wing-bending degree of freedom on the short-period dynamic-response factor and on the maximum dynamic-response factor when compared with the steady-state response of the system.
Author
Dynamic Response; Wings; Bending; Steady State

19980233519 Beijing Univ. of Aeronautics and Astronautics, Beijing, China
Influence of Sub-System Performance Parameter Uncertainty on Stability of Whole Flight Control System
Xinghua, Wang, Beijing Univ. of Aeronautics and Astronautics, China; Kebin, You, Beijing Univ. of Aeronautics and Astronautics, China; Li, Pang, Beijing Univ. of Aeronautics and Astronautics, China; Zongji, Chen, Beijing Univ. of Aeronautics and Astronautics, China; Journal of Beijing University of Aeronautics and Astronautics; Dec. 1997; ISSN 1001-5965; Volume 23, No. 6, pp. 757-762; In Chinese; No Copyright; Avail: Issuing Activity, Hardcopy, Microfiche

Analyzing the influence of sub-system performance index beyond the bounds on the whole flight control system is an important task posed by practical engineering. This paper presents a hierarchical analysis method, using a singular value sensitivity technique of sub-system performance parameter uncertainty on the flight control system stability. Some valuable results are derived.
Author
Flight Control; Systems Engineering; Control Stability; Independent Variables

19980234596 Naval Postgraduate School, Monterey, CA USA
Flight Testing and Real-Time System Identification Analysis of a UH-60A Black Hawk Helicopter with an Instrumented External Sling Load
McCoy, Allen H., Naval Postgraduate School, USA; Jun. 1998; 94p; In English
Contract(s)/Grant(s): RTOP 581-30-22
Report No.(s): NASA/CR-1998-196710; A-9809853; NAS 1.26:196710; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

Helicopter external air transportation plays an important role in today's world. For both military and civilian helicopters, external sling load operations offer an efficient and expedient method of handling heavy, oversized cargo. With the ability to reach areas otherwise inaccessible by ground transportation, helicopter external load operations are conducted in industries such as logging, construction, and fire fighting, as well as in support of military tactical transport missions. Historically, helicopter and load combinations have been qualified through flight testing, requiring considerable time and cost. With advancements in simulation and flight test techniques there is potential to substantially reduce costs and increase the safety of helicopter sling load certification. Validated simulation tools make possible accurate prediction of operational flight characteristics before initial flight tests. Real time analysis of test data improves the safety and efficiency of the testing programs. to advance these concepts, the U.S. Army and NASA, in cooperation with the Israeli Air Force and Technion, under a Memorandum of Agreement, seek to develop and validate a numerical model of the UH-60 with sling load and demonstrate a method of near real time flight test analysis. This thesis

presents results from flight tests of a U.S. Army Black Hawk helicopter with various external loads. Tests were conducted as the U.S. first phase of this MOA task. The primary load was a container express box (CONEX) which contained a compact instrumentation package. The flights covered the airspeed range from hover to 70 knots. Primary maneuvers were pitch and roll frequency sweeps, steps, and doublets. Results of the test determined the effect of the suspended load on both the aircraft's handling qualities and its control system's stability margins. Included were calculations of the stability characteristics of the load's pendular motion. Utilizing CIFER(R) software, a method for near-real time system identification was also demonstrated during the flight test program.
Author
Helicopters; Loads (Forces); Systems Stability; Military Operations; Military Helicopters; Flight Tests; Flight Characteristics

19980234622 NASA Marshall Space Flight Center, Huntsville, AL USA
Ascent, Transition, Entry, and Abort Guidance Algorithm Design for the X-33 Vehicle
Hanson, John M., NASA Marshall Space Flight Center, USA; Coughlin, Dan J., NASA Marshall Space Flight Center, USA; Dukeman, Gregory A., NASA Marshall Space Flight Center, USA; Mulqueen, John A., NASA Marshall Space Flight Center, USA; McCarter, James W., NASA Marshall Space Flight Center, USA; 1998; 11p; In English; GN and C Conference, 11 Aug. 1998, Boston, MA, USA; Sponsored by American Inst. of Aeronautics and Astronautics, USA
Report No.(s): AIAA Paper 97-4409; Copyright Waived; Avail: CASI; A03, Hardcopy; A01, Microfiche

One of the primary requirements for X-33 is that it be capable of flying autonomously. That is, onboard computers must be capable of commanding the entire flight from launch to landing, including cases where a single engine failure abort occurs. Guidance algorithms meeting these requirements have been tested in simulation and have been coded into prototype flight software. These algorithms must be sufficiently robust to account for vehicle and environmental dispersions, and must issue commands that result in the vehicle operating, within all constraints. Continual tests of these algorithms (and modifications as necessary) will occur over the next year as the X-33 nears its first flight. This paper describes the algorithms in use for X-33 ascent, transition, and entry flight, as well as for the powered phase of PowerPack-out (PPO) aborts (equivalent in thrust impact to losing an engine). All following discussion refers to these phases of flight when discussing guidance. The paper includes some trajectory results and results of dispersion analysis.
Author
X-33 Reusable Launch Vehicle; Airborne/Spaceborne Computers; Applications Programs (Computers); Flight Control; Control Simulation

19980235211 Beijing Univ. of Aeronautics and Astronautics, Beijing, China
Analysis of Fuzzy Control for Pitch-Attitude of Pilotless Helicopter
FenXian, Yu, Beijing Univ. of Aeronautics and Astronautics, China; Hongzhuan, Qiu, Beijing Univ. of Aeronautics and Astronautics, China; Lan, Wang, Beijing Univ. of Aeronautics and Astronautics, China; Hongbin, Zhang, Beijing Univ. of Aeronautics and Astronautics, China; Journal of Beijing University of Aeronautics and Astronautics; Dec. 1997; ISSN 1001-5965; Volume 23, No. 6, pp. 794-799; In Chinese; No Copyright; Avail: Issuing Activity, Hardcopy, Microfiche

A design method of Fuzzy control system is introduced. It is based on the Fuzzy logic and can be used, for the control of the pitch-attitude of pilotless helicopters. Because the transient response of the servo actuator is quick, there is a filter in the Fuzzy control system. Its function is to smooth the control signal and to reduce the oscillation of the servo actuator. The simulation shows the method is suitable.
Author
Fuzzy Systems; Pilotless Aircraft; Longitudinal Control; Helicopter Control

19980235622 NASA Langley Research Center, Hampton, VA USA
Low-Speed Measurements of Static and Oscillatory Lateral Stability Derivatives of a 1/5 Scale Model of a Jet-Powered Vertical-Attitude VTOL Research Airplane
Shanks, Robert E., NASA Langley Research Center, USA; Smith, Charles C., Jr., NASA Langley Research Center, USA; 1959; 24p; In English
Report No.(s): NASA-TM-X-143; L-640; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Force tests of the static and dynamic lateral stability characteristics of a VTOL airplane having a triangular wing mounted high on the fuselage with a triangular vertical tail on top of the wing and no horizontal tail have been made in the Langley free-flight

tunnel. The static lateral stability parameters and the rolling, yawing, and sideslipping dynamic stability derivatives are presented without analysis.
Author
Vertical Takeoff Aircraft; Lateral Stability; Attitude (Inclination); Free Flight; Dynamic Stability; Static Stability

19980235623 NASA Langley Research Center, Hampton, VA USA
An Experimental Investigation to Determine the Effect of Speed-Brake Position on the Longitudinal Stability and Trim of a Swept-Wing Fighter Airplane
Taylor, Robert T., NASA Langley Research Center, USA; 1959; 50p; In English
Report No.(s): NASA-TM-X-188; L-381; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche
A 0.10-scale model of a swept-wing fighter airplane was tested in the Langley high-speed 7- by 10-foot tunnel at Mach numbers from 0.60 to 0.92 to determine the effects of adding underfuselage speed brakes. The results of brief spoiler-aileron lateral control tests also are included. The tests show acceptable trim and drag increments when the speed brakes are installed at the 32-71-inch fuselage station.
Author
Lateral Control; Scale Models; Ailerons; Brakes (For Arresting Motion); Fighter Aircraft; Swept Wings

19980235625 NASA Langley Research Center, Hampton, VA USA
Supersonic Jet Tests of Missile Stabilizers
Vosteen. Louis F., NASA Langley Research Center, USA; Rosecrans, Richard, NASA Langley Research Center, USA; Dec. 1959; 28p; In English
Report No.(s): NASA-TM-X-121; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche
Seven stabilizers were tested at a Mach number of 2 in order to determine the effects of aerodynamic heating and loading on the structural stability of the stabilizer. The models differed in internal structure and postcure temperatures of the laminated Fiberglas skin. Tests were made at various stagnation temperatures between 440 F and 625 F. The postcure temperatures of the Fiberglas skins were found to affect significantly the ability of the model to withstand the imposed test conditions.
Author
Structural Stability; Missiles; Aerodynamic Heating; Supersonic Jet Flow; Supersonic Speed

19980236423 NASA Langley Research Center, Hampton, VA USA
Spin Tunnel Investigation of a 1/30 Scale Model of the North American A-5A Airplane
Lee, Henry A., NASA Langley Research Center, USA; Jun. 1964; 28p; In English
Report No.(s): NASA-TM-SX-946; NACA-AD-3140; L-3663; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche
An investigation has been made in the Langley spin tunnel to determine the erect and inverted spin and recovery characteristics of a 1/30-scale dynamic model of the North American A-5A airplane. Tests were made for the basic flight design loading with the center of gravity at 30-percent mean aerodynamic chord and also for a forward position and a rearward position with the center of gravity at 26-percent and 40-percent mean aerodynamic chord, respectively. Tests were also made to determine the effect of full external wing tanks on both wings, and of an asymmetrical condition when only one full tank is carried.
Author
A-5 Aircraft; Scale Models; Dynamic Models; Center of Gravity; Asymmetry

19980236453 NASA Langley Research Center, Hampton, VA USA
Lateral Directional Characteristics of a 1/10 Scale Free Flight Model of a Variable Sweep Fighter Airplane at High Angles of Attack
Boisseau, Peter C., NASA Langley Research Center, USA; Chambers, Joseph R., NASA Langley Research Center, USA; Dec. 1972; 53p; In English
Report No.(s): NASA-TM-SX-2649; N-AM-154; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche
An investigation was conducted to determine the lateral-directional characteristics of a 1/10-scale free-flying model of a variable-sweep fighter airplane at high angles of attack. The flight tests were conducted in the Langley full-scale tunnel and included steady flight at high angles of attack, 1g stalls, and studies of various piloting techniques for lateral control at high angles of attack. In addition, flights were made to evaluate the effects of artificial angular rate stabilization in yaw and roll. Tests were conducted

for wing-sweep angles of 22 deg, 35 deg, 50 deg, and 68 deg. Static and dynamic (forced-oscillation) wind-tunnel force tests and theoretical calculations of dynamic stability characteristics were also made as part of the investigation.
Author
Angle of Attack; Lateral Control; Fighter Aircraft; Flight Tests; Sweep Angle; Dynamic Stability; Angular Velocity

19980236455 NASA Langley Research Center, Hampton, VA USA
Theoretical Investigation of the Subsonic and Supersonic Flutter Characteristics of a Swept Wing Employing a Tuned Sting Mass Flutter Suppressor
Yates, E. Carson, Jr., NASA Langley Research Center, USA; 1960; 50p; In English
Report No.(s): NASA-TM-X-358; L-1015; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Flutter calculations were based on the modified strip-analysis method of NACA Research Memorandum L57L10 and covered Mach numbers from 0 to 1.75. The flutter suppressor consists of a small mass connected to the wing by means of a flexible sting. This system may be considered analogous to the spring-mass type of vibration absorber frequently used on heavy rotating machinery. Three suppressor masses were investigated but no attempt was made to make the configuration optimum.
Author
Flutter Analysis; Subsonic Flutter; Supersonic Flutter; Swept Wings; Vibration

19980236841 NASA Langley Research Center, Hampton, VA USA
Transonic Flutter Characteristics of an Aspect-Ratio-4, 45 deg. Sweptback, Taper-Ratio-0.2 Plan Form
Unangst, John R., NASA Langley Research Center, USA; Dec. 1959; 32p; In English
Report No.(s): NASA-TM-X-136; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The results of several flutter investigations to determine the effects of plan-form variations on the flutter characteristics of thin cantilevered wings at transonic Mach numbers have been reported previously. In the present investigation the data are extended to include a wing having an aspect ratio of 4, 45 of sweepback, and a taper ratio of 0.2. The data were obtained in the Langley transonic blowdown tunnel over a Mach number range from 0.6 to 1.4. The experimental results indicate an abrupt and rather large increase in both a flutter-speed parameter and a flutter-frequency parameter as the Mach number is increased from 1.05 to 1.10. The foregoing is interpreted as indicating a marked change in the flutter mode. Calculated flutter speeds, based on incompressible-flow aerodynamic coefficients, were too high by 20 percent or more throughout the subsonic Mach number range of the investigation. Calculated flutter frequencies were about 7 percent too high at a Mach number of 0.65 and were about 20 percent too high at a Mach number of 0.9. No significant independent effects of thickness were indicated for the plan form investigated as the thickness was changed from 3 to 4 percent chord.
Author
Flutter Analysis; Wind Tunnel Tests; Aerodynamic Coefficients; Incompressible Flow; Transonic Flutter; Sweptback Wings

09
RESEARCH AND SUPPORT FACILITIES (AIR)

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

19980232002 NASA Lewis Research Center, Cleveland, OH USA
A Three-Dimensional Flow Expander as a Device to Increase the Mach Number in a Supersonic Wind Tunnel
Salmi, Reino J., NASA Lewis Research Center, USA; Dec. 1958; 10p; In English
Report No.(s): NASA-MEMO-10-6-58E; L-5108; No Copyright; Avail: CASI; A02, Hardcopy; A01, Microfiche

A preliminary investigation of a simple 5 deg conical-flow expander was made to determine the feasibility of using this type of device to increase the Mach number in the test section of a supersonic wind tunnel. The inlet-to-exit area ratio of the nozzle was that required to increase one-dimensional flow from a Mach number of 3.88 to 5.5. The Mach numbers obtained at the expander exit varied from about 5.1 at the centerline to about 5.4 near the walls. No difficulty in operation of the main wind tunnel was experienced.
Author
Three Dimensional Flow; Supersonic Wind Tunnels; Mach Number; Conical Flow

19980235671 Kelsey Seybold, Huntsville, AL USA
Medical Support for Marshall Space Flight Center's Neutral Buoyancy Simulator
Dye, William B., Kelsey Seybold, USA; Bauer, Anne E., Kelsey Seybold, USA; Bradford, Brenda, Kelsey Seybold, USA; Proceeding from the 1997 NASA Occupational Health Conference: Achieving Quality in Occupational Health; Dec. 11, 1997, pp. 192; In English; Also announced as 19980235636; No Copyright; Avail: CASI; A01, Hardcopy; A03, Microfiche; Abstract Only; Abstract Only

Marshall Space Flight Center's neutral buoyancy simulator, a national historic landmark, was built in 1967 and began operation in 1968. The tank is forty feet deep and seventy-five feet in diameter and holds 1.3 million gallons of 90-degree F water. It has been placed on inactive status as of July 1997, while Johnson space Center's new Weightless Environmental Training Facility, having been built, is now operational. Marshall's Neutral Buoyancy Simulator has trained astronauts for missions ranging from Apollo and Skylab to Hubble Space Telescope and Space Station. In recent years, the Neutral Buoyancy Simulator has averaged one thousand two hundred hours in operation annually. Kelsey-Seybold Clinic, P.A. at Marshall Space Flight Center has provided medical support for divers at the neutral buoyancy tank. Kelsey-Seybold physicians performed an average of one hundred twenty dive physicals annually. HEMSI paramedics performed three thousand eight hundred predive checks annually. In recent years suited astronauts have successfully used a nitrox mixture of 43% oxygen and 57% nitrogen for Hubble Space Telescope related dives. The use of nitrox has allowed astronauts to prolong their dive times without increasing their risk for decompression sickness. Marshall Space Flight Center has a 1990 Houston Hyperbarics portable double lock hyperbaric chamber at the Neutral Buoyancy Simulator facility. A hyperbaric chamber team comprised of NASA personnel, onsite contractors, HEMSI paramedics and Kelsey-Seybold physicians was present onsite for emergencies requiring recompression during neutral buoyancy tank operation. Kelsey-Seybold physicians have received training in hyperbaric medicine. Chamber drills were periodically performed to maintain operating skills. A well-established system and the practice of preventive occupational medicine have resulted in an accident free environment at Marshall Space Flight Center's Neutral Buoyancy Simulator.
Author
Neutral Buoyancy Simulation; Decompression Sickness; Medical Services; Personnel

19980236157 Defence Science and Technology Organisation, Aeronautical and Maritime Research Lab., Melbourne, Australia
Performance Tests of the Original Transonic Wind Tunnel Compressor and Circuit
Link, Yoel Y., Defence Science and Technology Organisation, Australia; Quick, Howard A., Defence Science and Technology Organisation, Australia; May 1998; 98p; In English
Report No.(s): AD-A352590; DSTO-TN-0150; DODA-AR-010-527; No Copyright; Avail: CASI; A05, Hardcopy; A02, Microfiche

A detailed test programme of the AMRL Transonic Wind Tunnel was conducted. The objective of the test programme was to determine the pressure distributions around the tunnel circuit with larger nozzle exit areas. The existing high speed contraction, test section, model support mechanism, and downstream diffuser were removed for the tests. A variable nozzle and collector were designed and installed in place of the removed components to determine the effects of increasing the nozzle exit area. Three nozzle configurations were investigated, with a 38.3%, 44.4% and 58.1% increase in area relative to the existing test section area. Measurements were made of static pressure around the tunnel circuit, total pressure upstream and downstream from the compressor, and temperatures at various locations. Noise measurements were also made outside the tunnel complex and at four locations around the boundary of the site to determine the noise level of the wind tunnel.
DTIC
Transonic Wind Tunnels; Compressors; Circuits; Static Pressure; Noise Measurement

19980236506 Defence Science and Technology Organisation, Information Technology Div., Canberra Australia
A Computer Control Interface to Operate Turntables in the Test Section of a Wind Tunnel
Kent, S. A., Defence Science and Technology Organisation, Australia; May 1998; 44p; In English
Report No.(s): AD-A352634; DSTO-TR-0622; DODA-AR-010-528; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The Low Speed Wind Tunnel at the Aeronautical and Maritime Research Laboratory (AMRL) has, as part of its system, two interchangeable chambers, known as "test sections" where models to be tested are mounted. One of the requirements of a recent upgrade to the Low Speed Wind Tunnel control and data acquisition system was the ability to precisely position the turntables using computer control. This report describes the electronic hardware and software developed to enable computer control of the turntables by wind tunnel personnel.
DTIC
Computer Programs; Test Chambers; Data Acquisition; Low Speed Wind Tunnels

10
ASTRONAUTICS

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

19980232890 NASA Ames Research Center, Moffett Field, CA USA
The Effect of Lift on Entry Corridor Depth and Guidance Requirements for the Return Lunar Flight
Wong, Thomas J., NASA Ames Research Center, USA; Slye, Robert E., NASA Ames Research Center, USA; 1961; 22p; In English
Report No.(s): NASA-TR-R-80; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Corridors for manned vehicles are defined consistent with requirements for avoiding radiation exposure and for limiting values of peak deceleration. Use of lift increases the depth of the entry corridor. Mid-course guidance requirements appear to be critical only for the flight-path angle. Increasing the energy of the transport orbit increases the required guidance accuracy for the flight-path angle. Corrective thrust applied essentially parallel to the local horizontal produces the maximum change in perigee altitude for a given increment of velocity. Energy required to effect a given change in perigee altitude varies inversely with range measured from the center of the earth.
Author
Lift; Lunar Flight; Perigees; Corridors

19980232888 NASA Ames Research Center, Moffett Field, CA USA
Synthesis of Contributed Simulations for OREX Test Cases
Mehta, Unmeel B., NASA Ames Research Center, USA; Jul. 1998; 14p; In English; 1st; High Speed Flow Field Database, 12-14 Nov. 1997, Naples, Italy
Contract(s)/Grant(s): RTOP 509-10-11
Report No.(s): NASA/TM-1998-112238; A-9812224; NAS 1.15:112238; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A synthesis is presented of the computer simulations of the flow over the Orbital Reentry Vehicle (ORV) at the 92.8 km and 63.6 km Earth altitude trajectory points that were discussed at the First Europe-U.S. High Speed Flow Field Database Workshop Part 2, Napoli, Italy, November 1997. For the materials used on the surface of ORV, the non-catalytic wall condition is appropriate at 92.8 km and the finite-rate catalytic wall condition at 63.6 km. Additional simulations are required for establishing the independency of the discussed results from numerics. The proper modeling of natural phenomena needs further sensitivity studies. The uncertainties of inferred flight data are lacking for a proper evaluation of the presented results.
Author
Reentry Vehicles; Computerized Simulation; Flight Simulation; Control Simulation; Computational Fluid Dynamics; Hypersonic Reentry; Hypersonic Vehicles

19980236026 NASA Goddard Space Flight Center, Greenbelt, MD USA
GSFC Cutting Edge Avionics Technologies for Spacecraft
Luers, Philip J., NASA Goddard Space Flight Center, USA; Culver, Harry L., NASA Goddard Space Flight Center, USA; Plante, Jeannette, Swales Aerospace, USA; 1998; 9p; In English; Defense and Civil Space Programs, 28-31 Oct. 1998, Huntsville, AL, USA; No Copyright; Avail: CASI; A02, Hardcopy; A01, Microfiche

With the launch of NASA's first fiber optic bus on SAMPEX in 1992, GSFC has ushered in an era of new technology development and insertion into flight programs. Predating such programs the Lewis and Clark missions and the New Millenium Program, GSFC has spearheaded the drive to use cutting edge technologies on spacecraft for three reasons: to enable next generation Space and Earth Science, to shorten spacecraft development schedules, and to reduce the cost of NASA missions. The technologies developed have addressed three focus areas: standard interface components, high performance processing, and high-density packaging techniques enabling lower cost systems. to realize the benefits of standard interface components GSFC has developed and utilized radiation hardened/tolerant devices such as PCI target ASICs, Parallel Fiber Optic Data Bus terminals, MIL-STD-1773

and AS1773 transceivers, and Essential Services Node. High performance processing has been the focus of the Mongoose I and Mongoose V rad-hard 32-bit processor programs as well as the SMEX-Lite Computation Hub. High-density packaging techniques have resulted in 3-D stack DRAM packages and Chip-On-Board processes. Lower cost systems have been demonstrated by judiciously using all of our technology developments to enable "plug and play" scalable architectures. The paper will present a survey of development and insertion experiences for the above technologies, as well as future plans to enable more "better, faster, cheaper" spacecraft. Details of ongoing GSFC programs such as Ultra-Low Power electronics, Rad-Hard FPGAs, PCI master ASICs, and Next Generation Mongoose processors.
Author
Avionics; NASA Space Programs; Fiber Optics; Channels (Data Transmission); Earth Sciences; Spacecraft Design

19980236001 NASA Marshall Space Flight Center, Huntsville, AL USA
Mixing of Supersonic Streams
Hawk, Clark W., NASA Marshall Space Flight Center, USA; Landrum, D. Brian, NASA Marshall Space Flight Center, USA; Turner, Matthew, NASA Marshall Space Flight Center, USA; Wagner, David K., NASA Marshall Space Flight Center, USA; Lambert, James, NASA Marshall Space Flight Center, USA; 1998; 11p; In English; Propulsion Meeting, 16-17 Jul. 1998, Cleveland, OH, USA; Sponsored by Department of the Army, USA; Original contains color illustrations
Contract(s)/Grant(s): NCC8-123; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The Strutjet approach to Rocket Based Combined Cycle (RBCC) propulsion depends upon fuel-rich flows from the rocket nozzles and turbine exhaust products mixing with the ingested air for successful operation in the ramjet and scramjet modes. A model of the Strutjet device has been built and is undergoing test to investigate the mixing of the streams as a function of distance from the Strutjet exit plane. Initial cold flow testing of the model is underway to determine both, the behavior of the ingested air in the duct and to validate the mixing diagnostics. During the tests, each of the two rocket nozzles ejected up to two pounds mass per second into the 13.6 square inch duct. The tests showed that the mass flow of the rockets was great enough to cause the entrained air to go sonic at the strut, which is the location of the rocket nozzles. More tests are necessary to determine whether the entrained air chokes due to the reduction in the area of the duct at the strut (a physical choke), or because of the addition of mass inside the duct at the nozzle exit (a Fabri choke). The initial tests of the mixing diagnostics are showing promise.
Author
Supersonic Flow; Rocket Nozzles; Struts; Ramjet Engines; Gas Streams; Diagnosis; Cycles

11
CHEMISTRY AND MATERIALS

Includes chemistry and materials (general); composite materials; inorganic and physical chemistry; metallic materials; nonmetallic materials; propellants and fuels; and materials processing.

19980236473 Boeing Co., Phantom Works, Saint Louis, MO USA
Mechanical/Thermal Jet Surface Interactions in Paint Stripping Processes ***Final Report, 1 Jul. 1995 - 14 May 1998***
Parekh, D.; Glezer, A.; Crittenden, T.; Rogers, C; Meade, L.; Aug. 14, 1998; 52p; In English; Authors: S. Pothier, J. Kelley, and Y Ikeda. Prepared in cooperation with Tufts University, Medford, MA; Georgia Institute of Technology, Atlanta, GA. Characterization and Control of Two Phase Impinging Jets in Paint Stripping Processes.
Contract(s)/Grant(s): F49620-95-C-0048; AF Proj. 2307
Report No.(s): AD-A352238; AFRL-SR-BL-TR-98-0597; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

Several key accomplishments of this program are highlighted below: Experimental results for the two-phase flow were obtained on a full-scale production paint stripping system since this was the best way to assure that the relevant flow and system parameters were being considered. (1) Several key characteristics of two-phase nozzle flow have been characterized, including CO2 pellet sizing, distribution, velocity, sublimation and breakup. Initial results from a statistical analysis of this data is presented in the paper by Meade et al. (1997). The pellet breakup within the delivery system results in a reduction in the pellet size and an increase in number density. (2) Detailed flow visualization and surface static pressure distributions within a very high aspect ratio rectangular jet have been acquired. Pressure distributions were mapped both with and without CO2 pellets. Sublimation of the pellets within the delivery system results in the nozzle operating at a higher pressure ratio than the baseline air only case. Gaseous CO2 concentration levels were characterized at the nozzle exit. (3) Navier-Stokes simulations of internal nozzle flow were completed and estimates of particle trajectories were obtained by post-processing the steady-state flow solutions, using an analytical model for the particle drag. (4) Supporting diagnostic efforts led to the development of a new two-camera PIV technique and initial

application to two-phase jet flow (Pothier et al., 1997). (5) to provide a suitable actuator for control of this class of particle laden flows, development of a novel piston-cylinder synthetic jet actuator capable of producing supersonic peak velocities was initiated. An isentropic model to predict the time-dependent pressure in the cylinder for use in optimizing the actuator design was formalized and validated.
DTIC
Actuators; Cameras; Carbon Dioxide; Carbon Dioxide Concentration; Flow Characteristics; Flow Visualization; High Aspect Ratio

19980235568 Maverick Corp., Cincinnati, OH USA
Low-Cost Production of Composite Bushings for Jet Engine Applications *Final Report*
Gray, Robert A., Maverick Corp., USA; Aug. 1998; 20p; In English
Contract(s)/Grant(s): NAS3-27714; RTOP 523-21-13
Report No.(s): NASA/CR-1998-208515; NAs 1.26:208515; E-11293; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The objectives of this research program were to reduce the manufacturing costs of variable stator vane bushings by 1) eliminating the expensive carbon fiber braiding operation, 2) replacing the batch mode impregnation, B-stage, and cutting operations with a continuous process, and 3) reducing the molding cycle and machining operations with injection molding to achieve near-net shapes. Braided bushings were successfully fabricated with both AMB-17XLD and AMB-TPD resin systems. The composite bushings achieved high glass transition temperature after post-cure (+300 C) and comparable weight loss to the PNM-15 bushings. ANM-17XLD bushings made with "batch-mode" molding compound (at 0.5 in. fiber length) achieved a +300 lb-force flange break strength which was superior to the continuous braided-fiber reinforced bushing. The non-MDA resin technology developed in this contract appears attractive for bushing applications that do not exceed a 300 C use temperature. Two thermoplastic polyimide resins were synthesized in order to generate injection molding compound powders. Excellent processing results were obtained at injection temperatures in excess of 300 C. Micro-tensile specimens were produced from each resin type and the Tg measurements (by TMA) for these samples were equivalent to AURUM(R). Thermal Gravimetric Analysis (TGA) conducted at 10 C/min showed that the non-MDA AMB-type polyimide thermoplastics had comparable weight loss to PMR-15 up to 500 C.
Author
Jet Engines; Cost Reduction; Fiber Composites; Injection Molding; Machining; Manufacturing; Polyimides; Thermoplastic Resins; Thermogravimetry

19980236866 NASA Lewis Research Center, Cleveland, OH USA
Probabilistic Modeling of High-Temperature Material Properties of a 5-Harness 0/90 Sylramic Fiber/ CVI-SiC/ MI-SiC Woven Composite
Nagpal, Vinod K., Modern Technologies Corp., USA; Tong, Michael, NASA Lewis Research Center, USA; Murthy, P. L. N., NASA Lewis Research Center, USA; Mital, Subodh, Toledo Univ., USA; Oct. 1998; 20p; In English
Contract(s)/Grant(s): RTOP 537-04-22
Report No.(s): NASA/TM-1998-208497; NAS 1.15:208497; E-11292; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An integrated probabilistic approach has been developed to assess composites for high temperature applications. This approach was used to determine thermal and mechanical properties and their probabilistic distributions of a 5-harness 0/90 Sylramic fiber/CVI-SiC/Mi-SiC woven Ceramic Matrix Composite (CMC) at high temperatures. The purpose of developing this approach was to generate quantitative probabilistic information on this CMC to help complete the evaluation for its potential application for HSCT combustor liner. This approach quantified the influences of uncertainties inherent in constituent properties called primitive variables on selected key response variables of the CMC at 2200 F. The quantitative information is presented in the form of Cumulative Density Functions (CDFs). Probability Density Functions (PDFS) and primitive variable sensitivities on response. Results indicate that the scatters in response variables were reduced by 30-50% when the uncertainties in the primitive variables, which showed the most influence, were reduced by 50%.
Author
Ceramic Matrix Composites; Combustion Chambers; Mechanical Properties; Refractory Materials; Supersonic Transports; Woven Composites; Thermodynamic Properties

12
ENGINEERING

Includes engineering (general); communications and radar; electronics and electrical engineering; fluid mechanics and heat transfer; instrumentation and photography; lasers and masers; mechanical engineering; quality assurance and reliability; and structural mechanics.

19980231949 NASA Langley Research Center, Hampton, VA USA

External Interference Effects of Flow Through Static-Pressure Orifices of an Airspeed Head at Several Supersonic Mach Numbers and Angles of Attack

Silsby, Norman S., NASA Langley Research Center, USA; Mar. 1959; 16p; In English

Report No.(s): NASA-MEMO-2-13-59L; L-167; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Wind-tunnel tests have been made to determine the static-pressure error resulting from external interference effects of flow through the static-pressure orifices of an NACA airspeed head at Mach numbers of 2.4, 3.0, and 4.0 for angles of attack of 0 deg, 5 deg, 10 deg, and 15 deg. Within the accuracy of the measurements and for the range of mass flow covered, the static-pressure error increased linearly with increasing mass-flow rate for both the forward and rear sets of orifices at all Mach numbers and angles of attack of the investigation. For a given value of flow coefficient, the static-pressure error varied appreciably with Mach number but only slightly with angle of attack. For example, for a flow coefficient out of the orifices of 0.01 (the approximate value for a vertically climbing airplane for which the airspeed system incorporates an airspeed meter, a Mach meter, and an altimeter), the error increased from about 5 percent to about 12 percent of the static pressure as the Mach number increased from 2.4 to 4.0 with the airspeed head at an angle of attack of 0 deg.

Author

Mass Flow Rate; Airspeed; Orifices; Flow Coefficients; Wind Tunnel Tests; Mass Flow

19980231971 Woods Hole Oceanographic Inst., Dept. of Applied Ocean Physics and Engineering, MA USA

A Practical Hydrodynamic-Based Model of AUV Thruster Dynamics for Use in Closed-Loop Control of Vehicle Motions

Grosenbaugh, Mark A.; Whitcomb, Louis L.; Aug. 13, 1998; 14p; In English

Contract(s)/Grant(s): N00014-96-1-5014

Report No.(s): AD-A351204; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This report documents two novel improvements in the finite-dimensional nonlinear dynamical modeling of marine thrusters. Previously reported models, which fail to capture many of the characteristic nonlinear reponses that occur during unsteady operations, assume that the lift and drag forces on the propeller blades are proportional to the sine and cosine of the angle of attack where the angle of attack is a function of the axial flow velocity and the propeller's angular velocity. We have found that the lift and drag forces are not sinusoidal. We have also incorporated the effects of rotational fluid velocity and inertia on thruster response. The force curves-and model parameters are identified using experimental data from the load cell and acoustic doppler current meters. The accuracy of the model is determined by comparing experimental performance with numerical simulations. The results indicate that thruster models with nonsinusoidal lift and drag curves provide superior accuracy in both transient and steady-state response. Incorporating rotational fluid velocities into our model gave an insignificant improvement for our case. However, rotational fluid flow may be important for other types of thrusters. The research performed under this grant was reported in 9, 4, 14, 3, 5 and is referenced at the end of the text.

DTIC

Angle of Attack; Propeller Blades; Loads (Forces); Fluid Flow; Flow Velocity; Feedback Control

19980231977 NASA Langley Research Center, Hampton, VA USA

Water-Film Cooling of an 80 deg Total-Angle Cone at a Mach Number of 2 for Airstream Total Temperatures up to 3,000 deg R

Carter, Howard S., NASA Langley Research Center, USA; Jan. 1959; 36p; In English

Report No.(s): NASA-MEMO-12-27-58L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Film-cooling tests, with water as the coolant, were made on an 80 deg total-angle cone in a Mach number 2 free jet at sea-level pressure. The tests were made at free-stream total temperatures from 1,500 deg R to 3,000 deg R and at free-stream Reynolds numbers per foot from 8 x 10(exp 6) to 3 x 10(exp 6). The tests showed that the downstream end of the model became very hot if the coolant rate was too small to cover the complete model with a water film. This water film was fairly symmetrical when the model was at zero angle of attack but was very asymmetrical when the model was at an angle of attack of 5 deg. A comparison

with results of a previous transpiration-cooling test showed that, with water as the coolant, transpiration cooling was at least 2.5 times as efficient as the film cooling of the present tests.
Author
Liquid Cooling; Film Cooling; Air Flow; Angle of Attack; Free Flow; Free Jets

19980231978 Nanjing Univ. of Aeronautics and Astronautics, Nanjing, Jiangsu, China
Current State and Development of Research on Face Gear Drive
Rupeng, Zhu, Nanjing Univ. of Aeronautics and Astronautics, Nanjing, China; Shengcai, Pan, Nanjing Univ. of Aeronautics and Astronautics, Nanjing, China; Deping, Gao, Nanjing Univ. of Aeronautics and Astronautics, Nanjing, China; Journal of Nanjing University of Aeronautics and Astronautics; Jun. 1997; ISSN 1005-2615; Volume 29, No. 3, pp. 357-362; In Chinese; No Copyright; Avail: Issuing Activity, Hardcopy, Microfiche

The research state and problems to be further researched on the face gear drive are discussed. It has been reported by other authors that with proper design face gear drive can find a successful application in high power, such as helicopter transmissions. First, this paper presents a new recognition of limiting the large number of shapers and the small tooth width in the application of the face gear drive. For some kind of mechanical products, the model is limited, so are the parameters of gear. by the use of modern design methods, the effective tooth width of the face gear can be fully utilized. Secondly, the strength of bend, Hertzian stress and scuffing of face gear needs to be researched further. The analysis of stresses in face gears can be made by the finite element method and boundary element method. Thirdly, the new idea of the analysis of random elastic engagement is presented. The contact position of bearing is not only related to the geometry size, but also related to elastic deformation, while errors of assembly and manufacturing are random. The development of the random elastic engagement is necessary.
Author
Gears; Helicopter Propeller Drive

19980232009 NASA Ames Research Center, Moffett Field, CA USA
Inclined Bodies of Various Cross Sections at Supersonic Speeds
Jorgensen, Leland H., NASA Ames Research Center, USA; Nov. 1958; 64p; In English
Report No.(s): NASA-MEMO-10-3-58A; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

To aid in assessing effects of cross-sectional shape on body aerodynamics, the forces and moments have been measured for bodies with circular, elliptic, square, and triangular cross sections at Mach numbers 1.98 and 3.88. Results for bodies with noncircular cross sections have been compared with results for bodies of revolution having the same axial distribution of cross-sectional area (and, thus, the same equivalent fineness ratio). Comparisons have been made for bodies of fineness ratios 6 and 10 at angles of attack from 0 deg to about 20 deg and for Reynolds numbers, based on body length, of 4.0 x 10(exp 6) and 6.7 x 10(exp 6). The results of this investigation show that distinct aerodynamic advantages can be obtained by using bodies with noncircular cross sections. At certain angles of bank, bodies with elliptic, square, and triangular cross sections develop considerably greater lift and lift-drag ratios than equivalent bodies of revolution. For bodies with elliptic cross sections, lift and pitching-moment coefficients can be correlated with corresponding coefficients for equivalent circular bodies. It has been found that the ratios of lift and pitching-moment coefficients for an elliptic body to those for an equivalent circular body are practically constant with change in both angle of attack and Mach number. These lift and moment ratios are given very accurately by slender-body theory. As a result of this agreement, the method of NACA Rep. 1048 for computing forces and moments for bodies of revolution has been simply extended to bodies with elliptic cross sections. For the cases considered (elliptic bodies of fineness ratios 6 and 10 having cross-sectional axis ratios of 1.5 and 2), agreement of theory with experiment is very good. As a supplement to the force and moment results, visual studies of the flow over bodies have been made by use of the vapor-screen, sublimation, and white-lead techniques. Photographs from these studies are included in the report.
Author
Aerodynamic Coefficients; Bodies of Revolution; Slender Bodies; Pitching Moments; Moment Distribution; Lift Drag Ratio

19980232074 NASA Lewis Research Center, Cleveland, OH USA
Turbulent Boundary Layer on a Yawed Cone in a Supersonic Stream
Braun, Willis H., NASA Lewis Research Center, USA; 1959; 18p; In English
Report No.(s): NASA-TR-R-7; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The momentum integral equations are derived for the boundary layer on an arbitrary curved surface, using a streamline coordinate system. Computations of the turbulent boundary layer on a slightly yawed cone are made for a Prandtl number of 0.729, wall to free-stream temperature ratios of 1/2, 1, and 2, and Mach numbers from 1 to 4. Deflection of the fluid in the boundary layer

from outer stream direction, local friction coefficient, displacement surface, lift coefficient, and pitching-moment coefficient are presented.
Author
Aerodynamic Coefficients; Turbulent Boundary Layer; Transonic Speed; Supersonic Flow; Prandtl Number; Pitching Moments; Fluid Boundaries; Coefficient of Friction; Boundary Layers; Lift

19980232077 Michigan Univ., Ann Arbor, MI USA
Interaction Effects Produced by Jet Exhausting Laterally Near Base of Ogive-Cylinder Model in Supersonic Main Stream
Vinson, P. W., Michigan Univ., USA; Amick, J. L., Michigan Univ., USA; Liepman, H. P., Michigan Univ., USA; Feb. 1959; 40p; In English
Report No.(s): NASA-MEMO-12-5-58W; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The experimentally determined interaction effects of a side jet exhausting near the base of an ogive-cylinder model are presented and discussed. The interaction force appears to be independent of main-stream Mach number, boundary-layer condition (laminar or turbulent), angle of attack, and forebody length. The ratio of interaction force to jet force is found to be inversely proportional to the square root of the product of jet stagnation-to-free-stream pressure ratio and jet-to-body diameter ratio.
Author
Forebodies; Angle of Attack; Gas Streams; Stagnation Pressure; Supersonic Flow; Ogives

19980232079 NASA Langley Research Center, Hampton, VA USA
Investigation of the Maximum Spin-Up Coefficients of Friction Obtained During Tests of a Landing Gear Having a Static-Load Rating of 20,000 Pounds
Batterson, Sidney A., NASA Langley Research Center, USA; Jan. 1959; 24p; In English
Report No.(s): NASA-MEMO-12-20--58L; L-105; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An experimental investigation was made at the Langley landing loads track to obtain data on the maximum spin-up coefficients of friction developed by a landing gear having a static-load rating of 20,000 pounds. The forward speeds ranged from 0 to approximately 180 feet per second and the sinking speeds, from 2.7 feet per second to 9.4 feet per second. The results indicated the variation of the maximum spin-up coefficient of friction with forward speed and vertical load. Data obtained during this investigation are also compared with some results previously obtained for nonrolling tires to show the effect of forward speed.
Author
Landing Loads; Static Loads; Landing Gear; Coefficient of Friction

19980232090 NASA Langley Research Center, Hampton, VA USA
Effect of Convex Longitudinal Curvature on the Planing Characteristics of a Surface Without Dead Rise
Mottard, Elmo J., NASA Langley Research Center, USA; Feb. 1959; 36p; In English
Report No.(s): NASA-MEMO-1-25-59L; L-159; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A hydrodynamic investigation was made in Langley tank no. 1 of a planing surface which was curved longitudinally in the shape of a circular arc with the center of curvature above the model and had a beam of inches and a radius of curvature of 20 beams. The planing surface had length-beam ratio of 9 and an angle of dead rise of 0 deg. Wetted length, resistance, and trimming moment were determined for values of load coefficient C(sub Delta) from -4.2 to 63.9 and values of speed coefficient C(sub V) from 6 to 25. The effects of convexity were to increase the wetted length-beam ratio (for a given lift), to decrease the lift-drag ratio, to move the center of pressure forward, and ta increase the trim for maximum lift-drag ratio as compared with values for a flat surface. The effects were greatest at low trims and large drafts. The maximum negative lift coefficient C(sub L,b) obtainable with a ratio of the radius of curvature to the beam of 20 was -0.02. The effects of camber were greater in magnitude for convexity than for the same amount of concavity.
Author
Convexity; Curvature; Planing; Flat Surfaces; Aerodynamic Coefficients

19980232106 Nanjing Univ. of Aeronautics and Astronautics, Nanjing, Jiangsu, China
An Experimental Investigation on the Head of Annular Combustor for the Turbo-Shaft Engine
Jianxing, Zhao, Nanjing Univ. of Aeronautics and Astronautics, Nanjing, China; Jin, Hu, Nanjing Univ. of Aeronautics and Astronautics, Nanjing, China; Wangsan, Ding, Nanjing Univ. of Aeronautics and Astronautics, Nanjing, China; Xuewei, Zhan, Nanjing Univ. of Aeronautics and Astronautics, Nanjing, China; Journal of Nanjing University of Aeronautics and Astronautics; Jun. 1997; ISSN 1005-2615; Volume 29, No. 3, pp. 272-276; In Chinese; No Copyright; Avail: Issuing Activity, Hardcopy, Microfiche

An experimental investigation has been carried out to study the aerodynamic characteristics of two types of annular combustor sector for the turbo-shaft engine. Mean velocity profiles and turbulent intensity inside the asymmetric sudden expansion regions between the pre-diffuser exit and the head of the flame tube are measured, by laser Doppler anemometry. Airflow-patterns for seven different configuration sectors are experimentally mapped. The anemometry results reveal that two eddies occur in the sudden expansion regions and the flow patterns change with geometric parameters and inlet velocities. Measurements may be applied to develop and improve combustor design.

Author

Combustion Chambers; Turboshafts; Engines; Aerodynamic Characteristics

19980232230 NASA Langley Research Center, Hampton, VA USA

The Effect of Beam Loading on Water Impact Loads and Motions

Mixson, John S., NASA Langley Research Center, USA; Feb. 1959; 42p; In English

Report No.(s): NASA-MEMO-1-5-59L; L-130; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation of the effect of beam loading on impact loads and motions has been conducted in the Langley impact basin. Water impact tests of flat-bottom 5-inch- and 8-inch-beam models having beam-loading coefficients C(sub Delta) from 62.5 to 544 and a 30 0 dead-rise 5-inch-beam model A having beam-loading coefficients from 208 to 530 are described and the results analyzed to show trends of these heavy-beam-loading data with initial flight-path angle, trim angle, dead-rise angle, and time throughout the impact. Data from flat-bottom model tests, C(sub Delta) = 4.4 to 36.5, and from 300 dead-rise model tests, C(sub Delta)A = 0.58 and 18.8, are included, along with the heavy-beam-loading data; and variations of these data with beam-loading coefficients are shown. Each of the load and motion coefficients is found to be directly proportional to a power factor ofC(sub Delta). For instance, the maximum impact lift coefficient C(sub L,max)is found to be directly proportional to C(sub Delta)(sup 0.33) for the flat-bottom model and C(sub Delta)(sup 0.45) for the 30 deg dead-rise model. These variations of C(sub L,max) C(sub Delta) are found to be in agreement with theoretical variations. Finally, an empirical equation for the prediction of C(sub L,max) is presented and is shown to give good agreement with experimental C(sub L,max) for about 500 fixed-trim smooth-water impacts. The range of variables included dead-rise angles from 0 deg to 30 deg, beam-loading coefficients from 0.48 to 544, trim angles from 3 deg to 45 deg and initial flight-path angles from about 2 deg to about 27 deg.

Author

Aerodynamic Coefficients; Impact Loads; Impact Tests; Loads (Forces); Flight Paths

19980232233 NASA Ames Research Center, Moffett Field, CA USA

Exploratory Investigation of the Effects of Boundary-Layer Control on the Pressure-Recovery Characteristics of a Circular Internal-Contraction Inlet with Translating Centerbody at Mach Numbers of 2.00 and 2.35

Martin, Norman J., NASA Ames Research Center, USA; Feb. 1959; 48p; In English

Report No.(s): NASA-MEMO-12-31-58A; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Exploratory tests of a circular internal-contraction inlet were made at Mach numbers of 2.00 and 2.35 to determine the effect of a cowl-type boundary-layer control located downstream of the inlet throat. The inlet was designed for a Mach number of 2.5. Tests were also made of the inlet modified to correspond to design Mach numbers of 2.35 and 2.25. Surveys near the minimum area section of the inlet without boundary-layer control indicated maximum averaged pressure recoveries between 0.90 and 0.92 at a free-stream Mach number, M(sub infinity), of 2.35 for the inlets. Farther downstream, after partial subsonic diffusion, a maximum pressure recovery of 0.842 was obtained with the inlet at M(sub infinity) = 2.35. The pressure recovery of the inlet was increased by 0.03 at a Mach number of 2.35 and decreased by 0.02 at a Mach number of 2.00 by the application of cowl-type boundary-layer control. Further investigation with the inlet without bleed demonstrated that an increase of angle of attack from 0 deg to 3 deg reduced the pressure recovery 0.04. The effect of Reynolds number was to increase pressure recovery 0.07 (from 0.785 to 0.855) with an increase in Reynolds number (based on inlet diameter) from 0.79 x 10(exp 6) to 3.19 x 10(exp 6).

Author

Angle of Attack; Boundary Layer Control; Centerbodies; Diffusion; Pressure Recovery

19980232923 NASA Langley Research Center, Hampton, VA USA

A Method for Calculation of Hydrodynamic Lift for Submerged and Planing Rectangular Lifting Surfaces

Wadlin, Kenneth L., NASA Langley Research Center, USA; Christopher, Kenneth W., NASA Langley Research Center, USA; 1959; 16p; In English

Report No.(s): NASA-TR-R-14; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A method is presented for the calculation of lift coefficients for rectangular lifting surfaces of aspect ratios from 0.125 to 10 operating at finite depths beneath the water surface, including the zero depth or planing condition. Theoretical values are compared

with experimental values obtained at various depths of submergence with lifting surfaces of aspect ratios from 0.125 to 10. The method can also be applied to hydrofoils with dihedral. Lift coefficients computed by this method are in good agreement with existing experimental data for aspect ratios from 0.125 to 10 and dihedral angles up to 30 deg.
Author
Hydrofoils; Lift; Aerodynamic Coefficients; Aspect Ratio; Dihedral Angle; Hydrodynamics

19980233248 Bell Helicopter Co., Fort Worth, TX USA
Evaluation of Navy 9 cst Oil in Bell Helicopter M412 HP Gearboxes *Final Report*
Henry, Zachary S., Bell Helicopter Co., USA; Stapper, William R., Bell Helicopter Co., USA; Aug. 1998; 18p; In English
Contract(s)/Grant(s): NAS3-25455; RTOP 581-30-13; DA Proj. 1L1-62211-A-47-A
Report No.(s): NASA/CR-1998-208517; E-11296; NAS 1.26:208517; ARL-CR-430; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Tests were conducted with 5 and 9 centistoke lubricants in three different helicopter gearboxes, a main transmission, a 42 deg. gearbox and a tail-rotor gearbox. The objective of the tests was to observe and measure the difference in the performance of the lubrication systems due to the viscosity difference between the two test lubricants. The 9 centilstoke oil has been developed to provide higher component film thickness, increased load carrying capacity and improved corrosion resistance which will provide increased life for drive system gears and bearings. The results of the tests showed that at stabalized operating speeds and powers, the lubrication system performance of the 3 gearboxes with the 9 centistoke lubricant was similar to the performance with the 5 centistoke lubricant. These results allow limited aircraft flight testing using the 9 centistoke lubricant in place of the 5 centistoke lubricant for aircraft with gearboxes similar to the test gear-boxes.
Author
Transmissions (Machine Elements); Bell Aircraft; Tail Rotors; Lubrication Systems; Lubricants; Helicopters; Helicopter Propeller Drive

19980234242 Epilogics, Inc., Los Gatos, CA USA
Design, Fabrication, and Testing of a High-Speed, Over-Running Clutch for Rotorcraft *Final Report*
Fitz, Frank, Epilogics, Inc., USA; Gadd, Craig, Epilogics, Inc., USA; Aug. 1998; 84p; In English
Contract(s)/Grant(s): NAS3-27387; RTOP 581-30-13; DA Proj. 1L1-62211-A-47A
Report No.(s): NASA/CR-1998-208513; NAS 1.26:208513; E-11286; ARL-CR-429; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

The objective of this program was to evaluate the feasibility of a very high overrunning speed one-way clutch for rotor-craft applications. The high speed capability would allow placing the one-way clutch function at the turbine output shaft, that is, the input of the rotorcraft's transmission. The low drive torque present at this location would allow design of a relatively light one-way clutch. During the course of this program, two Mechanical Diode (MD) type overrunning clutches for high speeds were designed. One of the designs was implemented as a set of prototype clutches for high speed overrun testing. A high speed test stand was designed, assembled and qualified for performing overrunning and engagement tests at speeds up to 20,000 rpm. MD overrunning clutches were tested at moderate speed, up to 10,000 rpm and substantial thermal problems associated with oil shear were encountered. The MD design was modified, the modified parts were tested, and by program end, clutches were tested in excess of 20,000 rpm without excessive lubricant temperatures. Some correctable wear was observed and remains as a clutch characteristic which needs further improvement. A load cycle tester with a special, long, sample section was designed, built and then prototype clutches were fatigue tested to verify that the clutch design was suitable for carrying the specified power levels.
Author
Rotary Wing Aircraft; Clutches; Shafts (Machine Elements); Lubricants; Loads (Forces); Turbomachine Blades; Turbomachinery

19980235204 Nissan Motor Co. Ltd., Yokosuka, Japan
Numerical Analysis on Flows in Supersonic Air Intakes
Shimada, Toru, Nissan Motor Co. Ltd., Japan; Tamura, Naoki, Nissan Motor Co. Ltd., Japan; Sekino, Nobuhiro, Nissan Motor Co. Ltd., Japan; Tujimura, Naohisa, Nissan Motor Co. Ltd., Japan; Nissan Technical Review Transaction; 1992; ISSN 0912-9634, pp. 50-57; In Japanese; Original contains color illustrations; Copyright; Avail: Issuing Activity, Hardcopy, Microfiche

Flows in supersonic air intakes are investigated by solving the two-dimensional Reynolds-averaged Navier-Stokes equations. A comparison made with experimental data indicates the present computation is capable of predicting the flow quantities in question with sufficient accuracy. An efficient design approach is presented which combines the method of characteristics and the Nav-

ier-Stokes numerical simulation to optimize the pressure recovery in axisymmetric intakes. Hints obtained from the simulation were used to redesign a curved duct which is shown to provide a significant enhancement of the pressure recovery.
Author
Air Intakes; Flow Distribution; Supersonic Flow; Aerodynamics; Computational Fluid Dynamics; Navier-Stokes Equation

19980235582 Virginia Polytechnic Inst. and State Univ., Blacksburg, VA USA
Aeroelastic Analysis of Modern Complex Wings Using ENSAERO and NASTRAN ***Progress Report, 15 Sep. 1994 - 14 Sep. 1995***
Bhardwaj, Manoj, Virginia Polytechnic Inst. and State Univ., USA; 1995; 27p; In English; Original contains color illustrations
Contract(s)/Grant(s): NCC2-5097; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A process is presented by which static aeroelastic analysis is performed using Euler flow equations in conjunction with an advanced structural analysis tool, NASTRAN. The process deals with the interfacing of two separate codes in the fields of computational fluid dynamics (CFD) and computational structural dynamics (CSD). The process is demonstrated successfully on an F/A-18 Stabilator (horizontal tail).
Author
Computational Fluid Dynamics; Aeroelasticity; Nastran; Dynamic Structural Analysis; F-18 Aircraft; Aerodynamic Characteristics; Wings

19980236451 NASA Langley Research Center, Hampton, VA USA
Transonic Wind Tunnel Tests of an Error Compensated Static Pressure Probe
Capone, Francis J., NASA Langley Research Center, USA; 1961; 20p; In English
Report No.(s): NASA-TM-X-548; L-1562; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Pressure-error characteristics of a self-rotating static-pressure probe mounted on the nose of a missile body were determined at Mach numbers from 0.30 to 1.08 and at angles of attack from -2.7 deg to 15.3 deg. Results showed that at a Mach number of 1.00, the static-pressure error decreased from 3.5 percent to 0.8 percent of the free-stream static pressure as a result of a change in the orifice location from 0.15 maximum missile diameter to 0.20 maximum missile diameter forward of the missile nose. Although compensation for errors due to angles of attack up to 15.3 deg was maintained at Mach numbers from M = 0.30 to M = 0.50, there was an increase in error with an increase in angle of attack for Mach numbers between M = 0.50 and M = 1.08.
Author
Static Pressure; Angle of Attack; Errors; Transonic Wind Tunnels; Pressure Sensors; Missile Bodies

19980236813 NASA Langley Research Center, Hampton, VA USA
Characterization of an Ozone DIAL Receiver for Operation on an Unpiloted Atmospheric Vehicle
Goldschmidt, Soenke, Fachhochscule Ostfriesland, Germany; DeYoung, Russell J., NASA Langley Research Center, USA; Nineteenth International Laser Radar Conference; Jul. 1998, Part 2, pp. 919-922; In English; Also announced as 19980236718; No Copyright; Avail: CASI; A01, Hardcopy; A04, Microfiche

Laser remote sensing from aircraft has become a very important technique for observing ozone in the environment. NASA Langley has an active aircraft based research program which presently uses Nd:YAG-pumped dye lasers that are then doubled into the UV to probe both the stratosphere and troposphere for ozone using the differential absorption lidar (DIAL) technique. This large system can only fly on large (NASA DC-8, Electra) aircraft and has been deployed on many missions throughout the world. In the future it will be desirable to fly autonomous, lightweight, compact ozone DIAL instruments on unpiloted atmospheric vehicles (UAV) aircraft. Such aircraft could fly at high altitudes for extended times collecting science data without risk to the operator. Cost for such missions may be substantially reduced over present large aircraft based missions. Presently there are no ozone DIAL systems capable of flying on an UAV aircraft. In order to facilitate UAV missions, small more efficient laser transmitters need to be developed that emit approximately 25mJ near 300nm for each of the DIAL 'on' and 'off' line pulses. Also lightweight, compact DIAL receiver systems need to be built and demonstrated. Such receiver systems may incorporate fiber optic coupled telescopes for maximum light gathering capability per unit area, high quantum efficiency gated photomultiplier tubes with reasonable gain and very narrow-band filters for background light rejection with high light throughput. A compact high-performance 16-bit digitizer and a data storage system are also required. A conceptional design of such a UAV DIAL instrument is presented. Here a pulsed UV laser emits pulses into the atmosphere where elastic scattering occurs which results in light being

scattered into the receiver telescope. The subject of this paper is the design, construction and testing of a robust, compact ozone DIAL receiver system that would be a prototype for eventual use in a UAV aircraft.
Derived from text
Radar Receivers; Differential Absorption Lidar; Ozone; Radar Measurement; Airborne Radar; Pilotless Aircraft; Remote Sensors; Design Analysis; Photomultiplier Tubes

19980235507 New Energy and Industrial Technology Development Organization, Tokyo, Japan
Investigational research on eco-smart engines ***Eco-smart engine no chosa kenkyu***
Mar. 1998; 179p; In Japanese
Report No.(s): NEDO-PR-9712; DE98-770055; No Copyright; Avail: Issuing Activity (Natl Technical Information Service (NTIS)), Microfiche

The paper investigated the trend of research on eco-smart engines into which optimization function of engine performance, high environmental-adaptability, etc. are integrated. The investigation was made in Japan and abroad on technologies of combustion, structure/material, control, design/analysis, systematization, etc. In case of Japan, specifications were established for three types of engines, subsonic, supersonic and hypersonic aircraft, and the research subjects to fulfil the specifications were extracted. In case of the U.S. and Europe, the survey was made of combustion, materials, noise, design concept, control, etc. Important subjects are selected in priority order. Namely, for the enhancement of efficiency, the following were taken up: three-dimensional fiber-reinforced large-size light-weight structure application technology, heat-resistant advanced-material structure damage-tolerant design technology, pseudo-vesicular structure transpiration cooling technology, etc. For the reduction of NOx emission, the paper took up technologies of environmentally optimization combustion, AI combustion control, and non-cooling combustor liner application. For the noise reduction, technologies of new inclination hole orientation noise absorbing structure material application, super noise control, and innovative CFD utilization low noise aerodynamics. Moreover, the results of fiscal 1997 were outlined to indicate the research in the next fiscal year.
DOE
Aerodynamics; Internal Combustion Engines; Mechanical Engineering

19980235552 Lund Univ., Dept. of Heat and Power Engineering, Sweden
Radial gas turbine design
Krausche, S., Lund Univ., Sweden; Ohlsson, Johan, Lund Univ., Sweden; Apr. 1998; ISSN 0282-1990; 70p; In English
Report No.(s): LUTMDN-TMVK-5301; DE98-764032; No Copyright; Avail: Issuing Activity (Natl Technical Information Service (NTIS)), Microfiche

The objective of this work was to develop a program dealing with design point calculations of radial turbine machinery, including both compressor and turbine, with as few input data as possible. Some simple stress calculations and turbine metal blade temperatures were also included. This program was then implanted in a German thermodynamics program, Gasturb, a program calculating design and off-design performance of gas turbines. The calculations proceed with a lot of assumptions, necessary to finish the task, concerning pressure losses, velocity distribution, blockage, etc., and have been correlated with empirical data from VAT. Most of these values could have been input data, but to prevent the user of the program from drowning in input values, they are set as default values in the program code. The output data consist of geometry, Mach numbers, predicted component efficiency etc., and a number of graphical plots of geometry and velocity triangles. For the cases examined, the error in predicted efficiency level was within (+-) 1-2% points, and quite satisfactory errors in geometrical and thermodynamic conditions were obtained Examination paper.
DOE
Velocity Distribution; Turbine Blades; Turbines; Stress Analysis; Gas Turbine Engines; Engine Design

19980236162 Army Research Lab., Human Research and Engineering Directorate, Aberdeen Proving Ground, MD USA
Vibration Analysis Applied to the Motions of a Gas Turbine Engine ***Final Report, Oct. 1997 - Feb. 1998***
Korjack, T. A.; Aug. 1998; 26p; In English
Report No.(s): AD-A352911; ARL-TR-1750; No Copyright; Avail: Issuing Activity (Defense Technical Information Center (DTIC)), Microfiche

A method has been suggested to compare and evaluate vibration environments or sinusoidal and random specifications through the use of a single analysis technique. The comparison methodology utilized in this analysis could be significantly implemented and exploited in interpreting spectrum analysis results of vibration data such as in the analysis of a gas turbine engine of a modern tank. A generalized, multi-degree-of-freedom (DOF) lumped parameter structural system model was used to allow an initial evaluation of the environments such as shock, sine, and random so as to possibly eliminate a detailed separate dynamic

analysis for each environment Results indicate a possible reduction of analysis time and costs. In addition, since both sine and random analyses depend heavily on modal damping assumptions for the accuracy of their predictions, the simple methodology proposed herein should prove to be useful and productive.
DTIC
Gas Turbine Engines; Dynamic Structural Analysis; Random Variables

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GEOSCIENCES

Includes geosciences (general); earth resources and remote sensing; energy production and conversion; environment pollution; geophysics; meteorology and climatology; and oceanography.

19980236474 California Univ., Dept. of Chemistry, Santa Barbara, CA USA
Gas-Phase and Surface Reactivity of Highly Vibrationally and Translationally Excited Molecules ***Final Report, 1 Mar. 1995 - 28 Feb. 1998***
Wodtke, Alec M.; Aug. 20, 1998; 10p; In English
Contract(s)/Grant(s): F49620-95-1-0234; AF Proj. 2303
Report No.(s): AD-A352243; AFRL-SR-BL-TR-98-0601; No Copyright; Avail: CASI; A02, Hardcopy; A01, Microfiche

Understanding the gas-phase and gas-surface collision dynamics of highly vibrationally excited NO is necessary to model the rovibrational population distribution and infrared signatures of this important molecules in the upper atmosphere. Energy and momentum exchange between gas phase molecules and surfaces (especially oxidized surfaces) is also an important factor in the calculation of satellite drag coefficients. In this research the stimulated emission pumping method is used in combination with conventional molecular beams techniques to control the vibrational excitation, quantum state identity and collision energy of reactants in gas phase and gas-surface reactions. Results allow a better understanding of the collision dynamics controlling the state-specific population of vibrationally excited NO in the upper atmosphere and around satellites in low earth orbit.
DTIC
Vibrational States; Vapor Phases; Satellite Drag; Molecular Beams; Low Earth Orbits; Excitation; Aerodynamic Coefficients; Aerodynamic Drag

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LIFE SCIENCES

Includes life sciences (general); aerospace medicine; behavioral sciences; man/system technology and life support; and space biology.

19980232012 NASA Ames Research Center, Moffett Field, CA USA
The Effects of Longitudinal Control-System Dynamics on Pilot Opinion and Response Characteristics as Determined from Flight Tests and from Ground Simulator Studies
Sadoff, Melvin, NASA Ames Research Center, USA; Oct. 1958; 66p; In English
Report No.(s): NASA-MEMO-10-1-58A; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

The results of a fixed-base simulator study of the effects of variable longitudinal control-system dynamics on pilot opinion are presented and compared with flight-test data. The control-system variables considered in this investigation included stick force per g, time constant, and dead-band, or stabilizer breakout force. In general, the fairly good correlation between flight and simulator results for two pilots demonstrates the validity of fixed-base simulator studies which are designed to complement and supplement flight studies and serve as a guide in control-system preliminary design. However, in the investigation of certain problem areas (e.g., sensitive control-system configurations associated with pilot- induced oscillations in flight), fixed-base simulator results did not predict the occurrence of an instability, although the pilots noted the system was extremely sensitive and unsatisfactory. If it is desired to predict pilot-induced-oscillation tendencies, tests in moving-base simulators may be required. It was found possible to represent the human pilot by a linear pilot analog for the tracking task assumed in the present study. The criterion used to adjust the pilot analog was the root-mean-square tracking error of one of the human pilots on the fixed-base simulator. Matching the tracking error of the pilot analog to that of the human pilot gave an approximation to the variation of human-pilot behavior over a range of control-system dynamics. Results of the pilot-analog study indicated that both for optimized control-system

dynamics (for poor airplane dynamics) and for a region of good airplane dynamics, the pilot response characteristics are approximately the same.
Author
Pilot Induced Oscillation; Flight Tests; Human Behavior; Flight Simulators; Dynamic Response

19980235661 Edgerton, Germeshausen and Grier, Inc., Industrial Hygiene Branch, Cocoa Beach, FL USA
Remediation of Indoor Air Quality Concerns: Base Operations Building-Kennedy Space Center
Taffer, Jim, Edgerton, Germeshausen and Grier, Inc., USA; Geyer, Bart, Edgerton, Germeshausen and Grier, Inc., USA; Proceeding from the 1997 NASA Occupational Health Conference: Achieving Quality in Occupational Health; Dec. 11, 1997, pp. 156-160; In English; Also announced as 19980235636; No Copyright; Avail: CASI; A01, Hardcopy; A03, Microfiche

The Base Operations Contractor (BOC) Industrial Hygiene (IH) Office has received employee complaints concerning Indoor Air Quality (IAQ) at the Base Operations Building (BOB) since the late 1980s. Complaints continued to increase and in 1994/1995 several personnel reported to medical clinics with symptoms related to IAQ. The IH Office performed extensive evaluations to determine humidity, temperature, carbon dioxide, ozone, formaldehyde, carbon monoxide, various hydrocarbons, and respirable dust levels. No source of reported symptoms was identified. In 1995 a questionnaire was submitted to personnel to identify personnel complaints and to identify specific problem areas: 62 of the 64 (97%) employees responded with 25 (40%) reporting symptoms and 37 (60%) reporting complaints, mostly related to temperature, humidity, or dust build-up. Due to these findings, a BOC working group (members represented the Medical Office, Industrial Hygiene, HVAC, Energy Management, Structures, and Janitorial Departments) was formed to investigate this and other problem facilities. The group identified problems within this facility and offered corrective actions as follows: 1) Since HVAC systems on KSC are deactivated when facilities are not occupied, allowing humid air to enter air intakes, louvers were installed on air intakes which close when the system is deactivated. Humidity sensors were installed in the HVAC ducting which automatically activate when humidity levels exceed 60%; 2) to reduce dust deposition on horizontal surfaces in the facility, HEPA filter vacuums were purchased for the facility. Through time these vacuums should reduce respirable size dust, reducing personnel symptoms. One year later, the questionnaire was resubmitted. Only 17 (25%) of 67 personnel responded. Of those, 8 reported symptoms and 14 reported complaints attributed to IAQ. Complaints were mostly concerning temperature. Dust and humidity complaints were greatly reduced from the previous year. It is believed that most of those who did not respond no longer had complaints or symptoms. The number of personnel reporting to Medical with symptoms continues to decrease. Most personnel reporting symptoms or complaints work in one area of the facility. In this area old carpeting will be replaced to reduce contamination and the inner walls inspected for moisture build-up and mold growth.
Author
Indoor Air Pollution; Air Quality; Air Conditioning; Air Intakes; Humidity; Contamination; Dust; Industrial Safety

19980236495 Army Aeromedical Research Lab., Fort Rucker, AL USA
Designing Optimal Hierarchies for Information Retrieval with Multifunction Displays ***Final Report***
Francis, Gregory, Army Aeromedical Research Lab., USA; Jul. 1998; 64p; In English
Contract(s)/Grant(s): Proj-30162787A879
Report No.(s): AD-A352470; USAARL-98-33; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

Modern aircraft use computer screens with a push button interface to replace a variety of single purpose instruments. Such multifunction displays (MFDs) are gradually being introduced into military helicopters, with future aircraft likely to be highly dependent on computers. Studies have shown that poor design of MFD hierarchies has a significant impact on user satisfaction and performance. The purpose of this study was to extend a theoretical analysis of hierarchy search into a methodology for gathering data and building a hierarchy layout that minimized the time needed to find items in a hierarchy. Pilot studies demonstrate the effectiveness of the methodology and show that optimizing hierarchy layout may lead to a 25% reduction in search times.
DTIC
Human Factors Engineering; Flight Instruments; Airborne/Spaceborne Computers; Military Helicopters

19980236906 Georgia Inst. of Tech., Center for Human-Machine Systems Research, Atlanta, GA USA
Human-Centered Design of Human-Computer-Human Dialogs in Aerospace Systems ***Final Report***
Mitchell, Christine M., Georgia Inst. of Tech., USA; Aug. 1998; 7p; In English
Contract(s)/Grant(s): NCC2-824; No Copyright; Avail: CASI; A02, Hardcopy; A01, Microfiche

A series of ongoing research programs at Georgia Tech established a need for a simulation support tool for aircraft computer-based aids. This led to the design and development of the Georgia Tech Electronic Flight Instrument Research Tool (GT-EFIRT). GT-EFIRT is a part-task flight simulator specifically designed to study aircraft display design and single pilot interaction. ne simulator, using commercially available graphics and UNIX workstations, replicates to a high level of fidelity the Electronic Flight

Instrument Systems (EFIS), Flight Management Computer (FMC) and Auto Flight Director System (AFDS) of the Boeing 757/767 aircraft. The simulator can be configured to present information using conventional looking B757n67 displays or next generation Primary Flight Displays (PFD) such as found on the Beech Starship and MD-11.
Derived from text
Simulators; Flight Management Systems; Aircraft Design; Systems Management; Flight Instruments

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MATHEMATICAL AND COMPUTER SCIENCES

Includes mathematical and computer sciences (general); computer operations and hardware; computer programming and software; computer systems; cybernetics; numerical analysis; statistics and probability; systems analysis; and theoretical mathematics.

19980234597 Computer Sciences Corp., Hampton, VA USA
Stereo-Video Data Reduction of Wake Vortices and Trailing Aircrafts
Alter-Gartenberg, Rachel, Computer Sciences Corp., USA; Sep. 1998; 42p; In English
Contract(s)/Grant(s): NAS1-20431; RTOP 548-10-11-01
Report No.(s): NASA/CR-1998-208719; NAS 1.26:208719; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This report presents stereo image theory and the corresponding image processing software developed to analyze stereo imaging data acquired for the wake-vortex hazard flight experiment conducted at NASA Langley Research Center. In this experiment, a leading Lockheed C-130 was equipped with wing-tip smokers to visualize its wing vortices, while a trailing Boeing 737 flew into the wake vortices of the leading airplane. A Rockwell OV-10A airplane, fitted with video cameras under its wings, flew at 400 to 1000 feet above and parallel to the wakes, and photographed the wake interception process for the purpose of determining the three-dimensional location of the trailing aircraft relative to the wake. The report establishes the image-processing tools developed to analyze the video flight-test data, identifies sources of potential inaccuracies, and assesses the quality of the resultant set of stereo data reduction.
Author
Flight Tests; Image Processing; Imaging Techniques; Video Data; Vortices; Wakes; Wings

19980236958 Computer Sciences Corp., Hampton, VA USA
Evaluation of Frameworks for HSCT Design Optimization
Krishnan, Ramki, Computer Sciences Corp., USA; Oct. 1998; 42p; In English
Contract(s)/Grant(s): NAS1-20431
Report No.(s): NASA/CR-1998-208731; NAS 1.26:208731; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This report is an evaluation of engineering frameworks that could be used to augment, supplement, or replace the existing FIDO 3.5 (Framework for Interdisciplinary Design and Optimization Version 3.5) framework. The report begins with the motivation for this effort, followed by a description of an "ideal" multidisciplinary design and optimization (MDO) framework. The discussion then turns to how each candidate framework stacks up against this ideal. This report ends with recommendations as to the "best" frameworks that should be down-selected for detailed review.
Author
Multidisciplinary Design Optimization; Aircraft Design; Supersonic Aircraft; Computer Aided Design

19980234594 Virginia Polytechnic Inst. and State Univ., Multidisciplinary Analysis and Design Center for Advanced Vehicles, Blacksburg, VA USA
Variable-Complexity Multidisciplinary Optimization on Parallel Computers *Final Report, 7 Dec. 1993 - 31 Dec. 1997*
Grossman, Bernard, Virginia Polytechnic Inst. and State Univ., USA; Mason, William H., Virginia Polytechnic Inst. and State Univ., USA; Watson, Layne T., Virginia Polytechnic Inst. and State Univ., USA; Haftka, Raphael T., Florida Univ., USA; Jun. 1998; 8p; In English
Contract(s)/Grant(s): NAG1-1562
Report No.(s): NASA/CR-1998-208337; NAS 1.26:208337; No Copyright; Avail: CASI; A02, Hardcopy; A01, Microfiche

This report covers work conducted under grant NAG1-1562 for the NASA High Performance Computing and Communications Program (HPCCP) from December 7, 1993, to December 31, 1997. The objective of the research was to develop new multidisciplinary design optimization (MDO) techniques which exploit parallel computing to reduce the computational burden of aircraft MDO. The design of the High-Speed Civil Transport (HSCT) air-craft was selected as a test case to demonstrate the utility of our MDO methods. The three major tasks of this research grant included: development of parallel multipoint approximation

methods for the aerodynamic design of the HSCT, use of parallel multipoint approximation methods for structural optimization of the HSCT, mathematical and algorithmic development including support in the integration of parallel computation for items (1) and (2). These tasks have been accomplished with the development of a response surface methodology that incorporates multi-fidelity models. For the aerodynamic design we were able to optimize with up to 20 design variables using hundreds of expensive Euler analyses together with thousands of inexpensive linear theory simulations. We have thereby demonstrated the application of CFD to a large aerodynamic design problem. For the predicting structural weight we were able to combine hundreds of structural optimizations of refined finite element models with thousands of optimizations based on coarse models. Computations have been carried out on the Intel Paragon with up to 128 nodes. The parallel computation allowed us to perform combined aerodynamic-structural optimization using state of the art models of a complex aircraft configurations.
Author
Aircraft Configurations; Parallel Processing (Computers); Parallel Computers; Finite Element Method; Multidisciplinary Design Optimization; Design Analysis; Supercomputers

19980232051 California Univ., Dept. of Mechanical and Aerospace Engineering, Los Angeles, CA USA
Innovative Scaling Laws for Study of Nonlinear Aeroelastic and Aeroservoelastic Problems *Final Report, 1 Aug. 1994 - 31 Dec. 1997*
Friedmann, Peretz P., California Univ., USA; Apr. 02, 1998; 6p; In English
Contract(s)/Grant(s): F49620-94-1-0400
Report No.(s): AD-A351094; AFRL-SR-BL-TR-98-0564; No Copyright; Avail: CASI; A02, Hardcopy; A01, Microfiche

Developments in adaptive materials (or smart structures) have led to their use for actuators in aeroservoelastic applications. Tests demonstrating feasibility of adaptive materials based actuation have been conducted on small geometrically scaled models, and aeroelastic scaling has been disregarded. The primary objectives of our research activity were: (1) development of innovative aeroelastic scaling laws for aeroservoelastic and nonlinear aeroelastic problems, which allow one to extrapolate results, obtained from model tests to the full-scale configuration, and (2) application of the scaling laws to configurations illustrating difference between geometric and aeroelastic scaling. The primary accomplishments were: (1) development of a novel two pronged approach for generating innovative aeroelastic scaling laws for nonlinear aeroelastic and aeroservoelastic problems, and (2) developed scaling laws for flutter suppression in subsonic and transonic flow. In addition to conventional scaling parameters these requirements also address the sealing of control hinge moments and power required for flutter suppression. The research described has made an important contribution to the state-of-the-art.
DTIC
Actuators; Aeroelasticity; Aeroservoelasticity

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PHYSICS

Includes physics (general); acoustics; atomic and molecular physics; nuclear and high-energy; optics; plasma physics; solid-state physics; and thermodynamics and statistical physics.

19980232066 National Aerospace Lab., Amsterdam Netherlands
Computation of Aircraft Noise Propagation Through the Atmospheric Boundary Layer
Schulten, J. B. H. M., National Aerospace Lab., Netherlands; Jul. 30, 1997; 15p; In English; Presented at the Fifth International Congress on Sound and Vibration, University of Adelaide, Australia, Dec. 15-18, 1997.
Contract(s)/Grant(s): NAL-01607N
Report No.(s): AD-A351505; NLR-TP-97374-U; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Of all outdoor noise sources, aircraft probably have the largest impact on communities. As a result, the accurate prediction of aircraft noise exposure is of great interest. Nevertheless, conventional procedures for quantifying aircraft noise draw heavily on empirical data in which source and propagation effects are more or less statistically lumped together. A physically more relevant modeling of aircraft noise propagation is the ray acoustics approximation. Whereas ray acoustics techniques are well developed for stationary sources, they are not often applied to aircraft noise because the aircraft motion in principle requires many time-consuming computations to obtain the time history of a single takeoff or landing event. The present paper describes the application of the method of ray-tracing to a source moving along a three-dimensional path in a realistic atmosphere. The method is illustrated by typical examples of the effects of a non-uniform wind and temperature profile such as the formation of acoustic

shadow zones without any noise and, alternatively, zones with multiple reflections. It is shown that large reductions in computation time can be obtained if the flight path is close to level, which is factual for the majority of civil aircraft movements.
DTIC
Computation; Civil Aviation; Noise Propagation; Atmospheric Boundary Layer; Aircraft Noise; Noise Prediction (Aircraft)

19980232084 NASA Langley Research Center, Hampton, VA USA
Flight Performance of a Transonic Turbine-Driven Propeller Designed for Minimum Noise
OBryan, Thomas C., NASA Langley Research Center, USA; Hammack, Jerome B., NASA Langley Research Center, USA; May 1959; 24p; In English
Report No.(s): NASA-MEMO-4-19-59L; L-204; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Results are presented of a flight investigation to determine the aerodynamic characteristics of a transonic-type propeller. This propeller was designed for an advance ratio of 4.0 at a forward Mach number of 0.82 in an effort to limit the noise production. The measured efficiency of the propeller was 68 percent at the design Mach number of 0.82. This value compares with an efficiency as much as 15 percent higher with the same Mach number for a propeller designed for an optimum advance ratio of about 3.0. This penalty in efficiency must be considered in light of the resulting noise reduction. The noise under static and take-off conditions was measured to be 117.5 decibels, which represents a noise reduction of about 5 decibels (at 1,400 horsepower) compared with the advance-ratio-3 -design.
Author
Flight Characteristics; Noise Reduction; Propellers; Supersonic Turbines; Aerodynamic Characteristics

19980236567 Boeing Commercial Airplane Co., Seattle, WA USA
Boeing 18-Inch Fan Rig Broadband Noise Test
Ganz, Ulrich W., Boeing Commercial Airplane Co., USA; Joppa, Paul D., Boeing Commercial Airplane Co., USA; Patten, Timothy J., Boeing Commercial Airplane Co., USA; Scharpf, Daniel F., Boeing Commercial Airplane Co., USA; Sep. 1998; 204p; In English; Original contains color illustrations
Contract(s)/Grant(s): NAS1-20090; RTOP 538-03-11-01
Report No.(s): NASA/CR-1998-208704; NAS 1.26:208704; No Copyright; Avail: CASI; A10, Hardcopy; A03, Microfiche

The purposes of the subject test were to identify and quantify the mechanisms by which fan broadband noise is produced, and to assess the validity of such theoretical models of those mechanisms as may be available. The test was conducted with the Boeing 18-inch fan rig in the Boeing Low-Speed Aeroacoustic Facility (LSAF). The rig was designed to be particularly clean and geometrically simple to facilitate theoretical modeling and to minimize sources of interfering noise. The inlet is cylindrical and is equipped with a boundary layer suction system. The fan is typical of modern high-by-pass ratio designs but is capable of operating with or without fan exit guide vanes (stators), and there is only a single flow stream. Fan loading and tip clearance are adjustable. Instrumentation included measurements of fan performance, the unsteady flow field incident on the fan and stators, and far-field and in-duct acoustic fields. The acoustic results were manipulated to estimate the noise generated by different sources. Significant fan broadband noise was found to come from the rotor self-noise as measured with clean inflow and no boundary layer. The rotor tip clearance affected rotor self-noise somewhat. The interaction of the rotor with inlet boundary layer turbulence is also a significant source, and is strongly affected by rotor tip clearance. High level noise can be generated by a high-order nonuniform rotating at a fraction of the fan speed, at least when tip clearance and loading are both large. Stator-generated noise is the loudest of the significant sources, by a small margin, at least on this rig. Stator noise is significantly affected by propagation through the fan.
Author
Noise Measurement; Turbofans; Ducted Fans; Propeller Noise; Aerodynamic Noise; Unsteady Flow; Performance Tests; Aeroacoustics

19980236838 NASA Lewis Research Center, Cleveland, OH USA
Sound Pressures and Correlations of Noise on the Fuselage of a Jet Aircraft in Flight
Shattuck, Russell D., NASA Lewis Research Center, USA; Aug. 1961; 34p; In English
Report No.(s): NASA-TN-D-1086; E-1140; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Tests were conducted at altitudes of 10,000, 20,000, and 30,000 feet at speeds of Mach 0.4, 0.6, and O.8. It was found that the sound pressure levels on the aft fuselage of a jet aircraft in flight can be estimated using an equation involving the true airspeed

and the free air density. The cross-correlation coefficient over a spacing of 2.5 feet was generalized with Strouhal number. The spectrum of the noise in flight is comparatively flat up to 10,000 cycles per second.
Author
Flight Conditions; Sound Pressure; Subsonic Speed; Altitude; Fuselages; Jet Aircraft Noise

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GENERAL

19980232022 NASA, Washington, DC USA
Aeronautics and Space Report of the President: Fiscal Year 1997 Activities
Sep. 1998; 80p; In English; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

This report covers aerospace activities of US federal government agencies during FY 1997. Appendices include historical budget data and human spaceflight records.
Derived from text
Aerospace Sciences; General Overviews; NASA Space Programs; Defense Industry; Manned Space Flight; Aeronautics

19980234247 NASA Langley Research Center, Hampton, VA USA
NASA Langley Highlights, 1997
Jul. 1998; 84p; In English
Report No.(s): NASA/TM-1998-208451; L-17765; NAS 1.15:208451; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

Langley's mission is accomplished by performing innovative research relevant to national needs and Agency goals, transferring technology to users in a timely manner, and providing development support to other USA Government Agencies, industry, other NASA Centers, the educational community, and the local community. This report contains highlights of some of the major accomplishments and applications that have been made by Langley researchers and by our university and industry colleagues during the past year. The highlights illustrate the broad range of research and technology activities carried out by NASA Langley Research Center and the contributions of this work toward maintaining USA' leadership in aeronautics and space research.
Author
Research and Development; Technology Utilization; NASA Programs; Technology Transfer; Space Commercialization; Aerospace Engineering; Aeronautical Engineering

Subject Term Index

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