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Derivation and Definition of a Linear Aircraft Model (augusti 1988)
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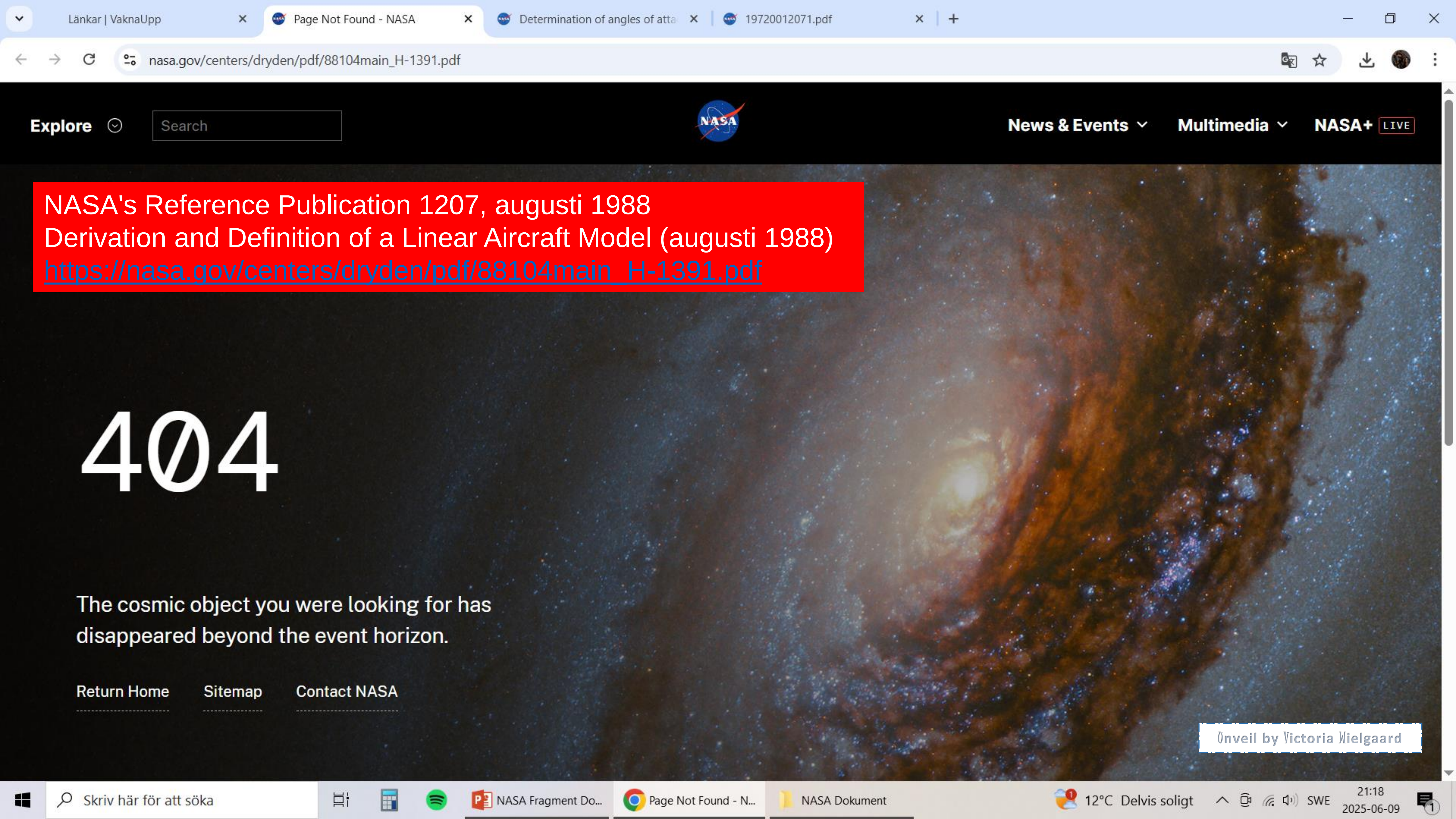

SUMMARY

This report documents the derivation and definition of a linear aircraft model for a rigid aircraft of constant mass flying over a flat, nonrotating earth. The derivation makes no assumptions of reference trajectory or vehicle symmetry. The linear system equations are derived and evaluated along a general trajectory and include both aircraft dynamics and observation variables.

INTRODUCTION

The need for linear models of aircraft for the analysis of vehicle dynamics and control law design is well known. These models are widely used, not only for computer applications but also for quick approximations and desk calculations. Whereas the use of these models is well understood and well documented, their derivation is not. The lack of documentation and, occasionally, understanding of the derivation of linear models is a hindrance to communication, training, and application.

This report details the development of the linear model of a rigid aircraft of constant mass, flying over a flat, nonrotating earth. This model consists of a state equation and an observation (or measurement) equation. The system equations have been broadly formulated to accommodate a wide variety of applications. The linear state equation is derived from the nonlinear six-degree-of-freedom equations of motion. The linear observation equation is derived from a collection of nonlinear equations representing state variables, time derivatives of state variables, control inputs, and flightpath, air data, and other parameters. The linear model is developed about a nominal trajectory that is general.

Whereas it is common to assume symmetric aerodynamics and mass distribution, or a straight and level trajectory, or both (Clancy, 1975; Dommasch and others, 1967; Etkin, 1972; McRuer and others, 1973; Northrop Aircraft, 1952; Thelander, 1965), these assumptions limit the generality of the linear model. The principal contribution of this report is a solution of the general problem of deriving a linear model of a rigid aircraft without making these simplifying assumptions. By defining the initial conditions (of the nominal trajectory) for straight and level flight and setting the asymmetric aerodynamic and inertia terms to zero, one can easily obtain the more traditional linear models from the linear model derived in this report.

Another significant contribution of this report is the derivation and definition of a linear observation (measurement) model. The observation model is often entirely neglected in standard texts. A thorough treatment of common aircraft measurements is presented by Gainer and Hoffman (1972), and Gracey (1980) provides a detailed discussion of speed and altitude measurements. However, neither of these references present linear models of these measurements. This report relies heavily on these two references and uses their results as one of the bases for the nonlinear measurement equations from which the linear measurement model is derived. Also included in this report is a large number of other measurements or variables for observation that have been found to be useful in vehicle analysis and control law design.

Duke and others (1987) describe a FORTRAN program called LINEAR that derives a linear aircraft model by numerical differencing (Dieudonne, 1978). The program LINEAR produces a linear aircraft model (both state and observation matrices) that is equivalent to the linear models defined in this report.

... that define the reference systems and nonlinear state and ... linear model presented in the appendixes (section 2). The ... odynamic model used in this report (app. A), a derivation ... meters (app. B), generalized linear derivatives of the non- ... and the individual derivatives of the state and observation ... results of this report are presented in appendix D.

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$$C^{-1} = \left[\begin{array}{cc|c} C_{11}^{-1} & -C_{11}^{-1}C_{12}C_{22}^{-1} & \\ 0_{3\times3} & C_{22}^{-1} & 0_{6\times6} \\ \hline & 0_{6\times6} & 1_{6\times6} \end{array}\right] \tag{2-64}$$

The elements of the A', B', H', and F' matrices can be determined using the C^{-1} matrix defined in equation (2-64), the A, B, H, G, and F matrices, and the definitions for A', B', H', and F' given in equations (2-21), (2-22), (2-38), and (2-39).

3 CONCLUDING REMARKS

This report derives and defines a set of linearized system matrices for a rigid aircraft of constant mass, flying in a stationary atmosphere over a flat, nonrotating earth. Both generalized and standard linear system equations are derived from nonlinear six-degree-of-freedom equations of motion and a large collection of nonlinear observation (measurement) equations.

This derivation of a linear model is general and makes no assumptions on either the reference (nominal) trajectory about which the model is linearized or the symmetry of the vehicle mass and aerodynamic properties.

Ames Research Center
Dryden Flight Research Facility
National Aeronautics and Space Administration
Edwards, California, January 8, 1987

APPENDIX A—AERODYNAMIC FORCES AND MOMENTS

...ng on an aircraft are the result of multiple factors whose signif-
...as from vehicle to vehicle. In general, these forces and moments
... number, angle of attack, angle of sideslip, altitude, rotational
...he purposes of this report, the aerodynamic forces and moments
...lowing form:

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General Equations of Motion for a Damaged Asymmetric Aircraft.
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General Equations of Motion for a Damaged Asymmetric Aircraft

There is a renewed interest in dynamic characteristics of damaged aircraft both in order to assess survivability and to develop control laws to enhance survivability. This paper presents a set of flight dynamics equations of motion for a rigid body not necessarily referenced to the body's center of mass. Such equations can be used when the body loses a portion of its mass and it is desired to track the motion of the body s previous center of mass/reference frame now that the mass center has moved to a new position. Furthermore, results for equations presented in this paper and equations in standard aircraft simulations are compared for a scenario involving a generic transport aircraft configuration subject to wing damage.

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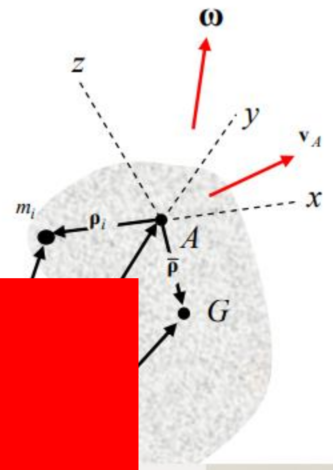
define angle of attack and sideslip angle, independent variables for the aerodynamic forces and moments. The proposed equations of motion are then applied to the problem of modeling large instantaneous shifts in center of mass of a rotating body that suddenly loses a portion of its mass. Benefits in simulation implementation are discussed. As an example, open loop dynamic responses of a generic transport aircraft with wing damage are compared for typical equations of motion used in the aircraft simulation and the equations presented here.

II. Rigid Body Equations of Motion Referenced to an Arbitrary Fixed Point on the Body

There are several approaches that can be used to develop the general equations of motion. The one selected here starts with Newton's laws applied to a collection of particles defining the rigid body (any number of dynamics or physics books can serve as references, e.g. reference 2). In this paper, the rigid body equations of motion over a flat non-rotating earth are developed that are not necessarily referenced to the body's center of mass. Such equations will be used in the next section when the body loses a portion of its mass and it is desired to track the motion of the body's previous center of mass/reference frame now that the mass center has moved to a new position



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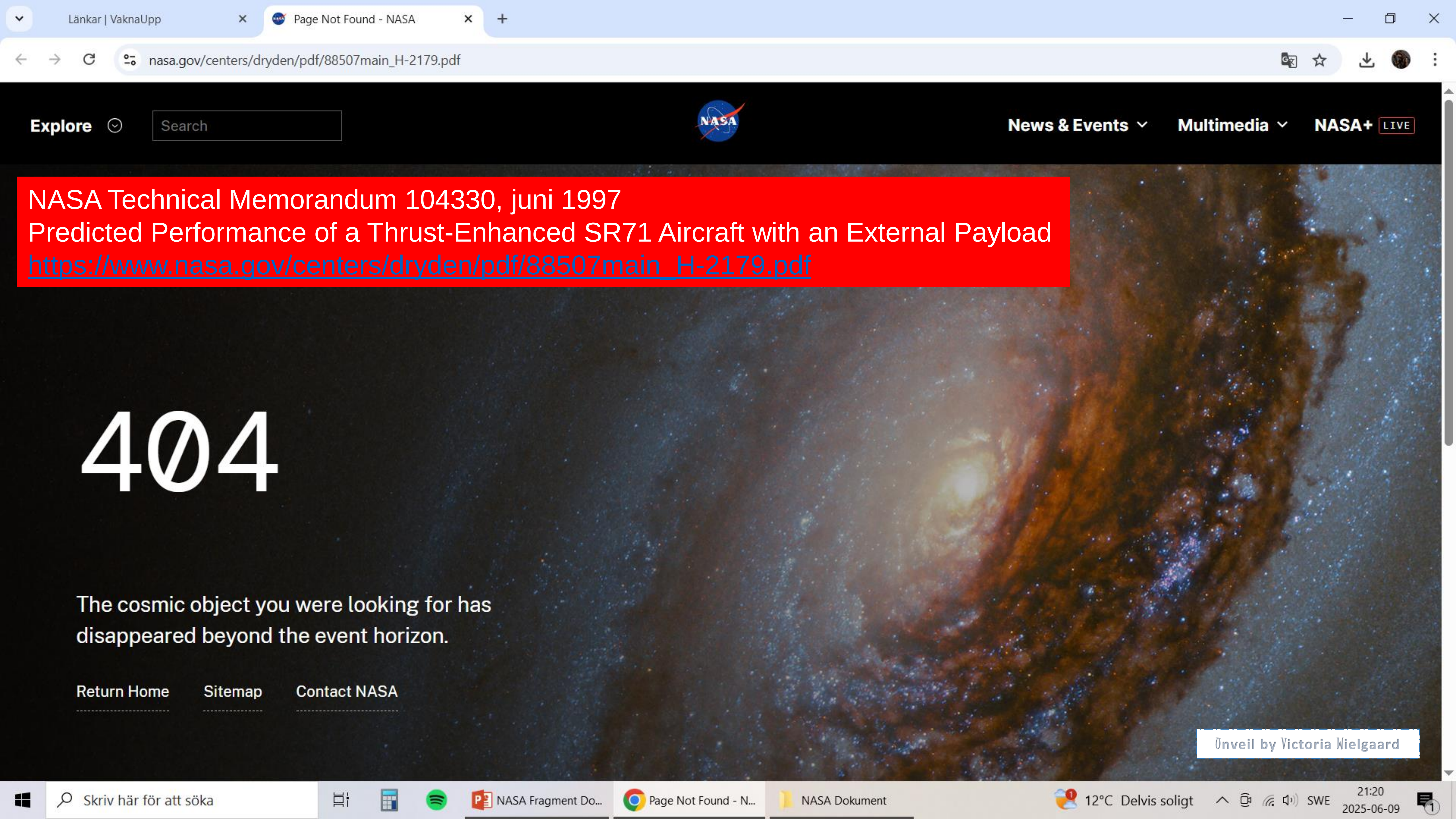

maneuver is to be modeled. Maneuver options include constant Mach and airspeed climbs, level accelerations, and constant *g* pushovers or pull-ups. The user is asked for initial and ending conditions. After execution, the DPS displays the output of the aircraft states and performance as a function of time.

The DPS equations of motion use four assumptions that simplify the program while maintaining its fidelity for most maneuvers and applications: point-mass modeling, nonturbulent atmosphere, zero side forces, and a nonrotating Earth. The primary advantages of using the DPS over a piloted real-time simulator are that it is much easier to modify the aerodynamic and propulsion data tables, and the DPS easily allows back-to-back comparisons of vehicle performance using a maneuver flown exactly the same in each case despite a varying vehicle configuration. Also, the DPS computes aircraft performance in compressed time, allowing much faster viewing of the results than if the maneuvers were flown in real-time on a piloted simulator.

An SR-71 DPS module was developed specifically for this study to predict vehicle performance with and without the drag and weight corresponding to the external payload and the additional thrust and fuel flow associated with the enhanced J58 engines. The aircraft aerodynamic and propulsion data used to build the baseline SR-71 model were supplied by Lockheed.[8] The thrust increase and fuel flow data for the engine enhancement options were supplied by Pratt & Whitney. Isolated hypersonic vehicle drag data were provided by the NASA Ames Research Center and were based on a derivative of the original GHRV. Lockheed supplied estimates of the payload interference drag. The thrust enhancement data and payload drag data were then added to the DPS code as separate arrays and available as increments to the baseline data for use in the performance computations.

Once developed, the baseline DPS SR-71 option was verified against flight data from the pilot's flight manual.[9] Different climb schedules were analyzed at standard day conditions. The resulting predicted elapsed time, fuel usage, and range data were compared to the values in the manual. The predictions compared closely to the actual flight data. It was believed the DPS model could be confidently used for estimating vehicle performance with additional payload drag and the thrust enhancement options.

After refueling, the SR-71 aircraft is normally piloted along a constant Mach 0.9 climb to approximately 33,000 ft (10,058 m), pushed over at a 3000 ft/min (914 m/min) descent to approximately 30,000 ft (9144 m), and pulled out in a level acceleration to 450 *KEAS* (Mach 1.25). This transonic penetration procedure is known as the climb-dive technique. At 450 *KEAS*, the vehicle initiates a constant equivalent airspeed climb to Mach 2.6. At this Mach number, equivalent airspeed is slowly reduced until the vehicle reaches the design cruise condition (3.2 maximum Mach number and 85,000 ft (25,908 m) maximum altitude). The climb-dive technique was discovered by Redin in the early 1970s to use less fuel than level transonic acceleration at 25,000 ft (7620 m) uses.[10]

Using high payload drag increments, the climb-dive technique was reevaluated using different pushover and pull-out altitudes to see if another combination gave greater fuel savings than were realized without payload drag in Redin's research. High pushover altitudes up to 37,000 ft (11,278 m) offered large fuel savings. However, altitudes greater than 33,000 ft (10,058 m) were not used because, subsonically, the SR-71 aircraft can approach uncontrollable pitch-up conditions there. Pull-out altitudes below 28,000 ft (8534 m) offered slightly greater benefits than those above 28,000 ft (8534 m), but were avoided because focused sonic boom effects on the population become an issue.

The most feasible combination was a pushover altitude of 33,000 ft (10,058 m) combined with a pull-out altitude of 28,000 ft (8534 m). Also used was an increased descent rate of 6000 ft/min (1829 m/min) previously found by Redin to still offer a manageable piloting task. The selected schedule used approximately 2800 lbm (1270 kg) less of JP7 than level transonic acceleration at 25,000 ft (7620 m) would have used. This fuel savings for the high-drag configuration was substantially higher than that previously predicted by Redin for a clean (no additional external drag) vehicle.

In the GHRV feasibility report, Lockheed recommended that the traditional 450 *KEAS* supersonic climb schedule be replaced with a 500 *KEAS* profile. The benefit of the higher *KEAS* schedule is an increase in excess thrust and the corresponding ability to handle higher payload drag than the 450 *KEAS* schedule can handle. This equivalent airspeed currently represents the right-hand edge of the conventional envelope and would require a very precise piloting technique to avoid overspeeding the aircraft. Otherwise,

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A method for reducing the sensitivity of optimal nonlinear systems to parameter uncertainty

Method for reducing sensitivity of optimal nonlinear systems to parameter uncertainty

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satisfied. A brief derivation of the necessary relationships with notation common to many steepest-descent programs is provided in appendix A. The algorithm was applied to the following example problem.

A NUMERICAL EXAMPLE

Problem Statement

The example problem is a fixed-time problem in which it is required to determine the thrust-attitude program of a single-stage rocket vehicle starting from rest and going to specified terminal conditions of altitude and vertical velocity which will maximize the final horizontal velocity. The idealizing assumptions made are the following:

(1) A point-mass vehicle
(2) A flat, nonrotating earth
(3) A constant-gravity field, $g = 9.8\ \text{m/sec}^2$ $(32.2\ \text{ft/sec}^2)$
(4) Constant thrust and mass-loss rate
(5) A nonlifting body in a nonvarying atmosphere with a constant drag parameter $K_D = \frac{1}{2}\rho C_D S$, where S is the frontal surface area.

The coordinate system and pertinent geometric relations and terms are shown in figure 1. The differential equations of motion needed in the algorithm setup are

$$\left.\begin{aligned} \frac{du}{dt} &= \frac{1}{m}\left(T\cos\theta - K_D uV\right) = \dot{x}_1 = f_1 \\ \frac{dy}{dt} &= v = \dot{x}_2 = f_2 \\ \frac{dv}{dt} &= \frac{1}{m}\left(T\sin\theta - K_D uV\right) - g = \dot{x}_3 = f_3 \end{aligned}\right\} \tag{12}$$

NASA Technical Note 20040008097, april 1961
Calculation of Wind Compensation for Launching of unguided Rockets
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Calculation of Wind Compensation for Launching of Unguided Rockets

A method for calculating wind compensation for unguided missiles s derived which has a greater degree of flexibility than the previously proposed methods. Utilization of the wind-compensation technique is demonstrated by using the Shotput vehicle as a model. Postflight simulations of four of these missiles with the use of measured winds show that if the winds are known, very good accuracy can be obtained by using the proposed method.

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computed in the IBM 704 electronic data processing machine using the aerodynamic parameters presented above and the trajectory program discussed in reference 8. An ICAO standard atmosphere (ref. 9) and a launch angle of 78° were used in these computations.

ANALYSIS

The wind-compensation procedure derived herein involves four aspects. They are an adequate trajectory simulation, selection of wind profiles, development of wind-compensation graphs, and a wind-weighting procedure.

Trajectory Simulation

The requirements for a trajectory program needed for a wind-compensation procedure are (1) that the trajectory be three dimensional, (2) that provision be made for arbitrary wind velocity and azimuth and (3) that nonlinear aerodynamics with respect to flow incidence angle be included. The first two requirements are obvious since, in the consideration of side winds, the trajectory is three dimensional and the wind velocity and azimuth are arbitrary. The third requirement is imposed because the introduction of surface winds during launch can create angles of attack larger than 90°, which greatly exceed the linear range of the aerodynamic coefficients.

A trajectory simulation incorporating the above requirements is presented in reference 8. In addition to the above requirements, this simulation assumes a vehicle with six degrees of freedom and aerodynamic symmetry in roll and the missile position in space is computed relative to a flat nonrotating earth. This trajectory simulation was programmed on the IBM 704 electronic data processing machine and is the basis for all trajectory computations made in this paper.

Selection of Wind Profiles

The winds at some geographical locations have been measured and
... a year. These measurements
... increases with altitude until
... then decreases rather abruptly.
..., Cocoa, Florida are presented

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User's Manual for LINEAR, a FORTRAN program to Derive Linear Aircraft Models
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NASA Technical Paper 2768

December 1987

User's Manual for LINEAR, a FORTRAN Program to Derive Linear Aircraft Models

Eugene L. Duke,
Brian P. Patterson,
and Robert F. Antoniewicz

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The program LINEAR numerically determines a linear system model using nonlinear equations of motion and a user-supplied nonlinear aerodynamic model. LINEAR is also capable of extracting linearized gross engine effects (such as net thrust, torque, and gyroscopic effects) and including these effects in the linear system model. The point at which this linear system model is defined is determined either by specifying the state and control variables or by selecting an analysis point on a trajectory, selecting a trim option, and allowing the program to determine the control variable and remaining state variables to satisfy the trim option selected.

Because the program is designed to satisfy the needs of a broad class of users, a wide variety of options has been provided. Perhaps the most important of these options are those that allow user specification of the state, control, and observation variables to be included in the linear model derived by LINEAR.

Within the program, the nonlinear equations of motion include 12 states representing a rigid aircraft flying in a stationary atmosphere over a flat nonrotating earth. Thus, the state vector x is computed internally as

$$x = [p \quad q \quad r \quad V \quad \alpha \quad \beta \quad \phi \quad \theta \quad \psi \quad h \quad x \quad y]^T$$

The nonlinear equations used to determine the derivatives of the quantities are presented in the following section (Equations of Motion). The internal control vector u can contain up to 30 controls. The internal observation vector y contains 120 variables, including the state variables, the time derivatives of the state ... f interest.

NASA Technical Paper 2835, september 1988
User's Manual for LINEAR, a FORTRAN Program to Derive Linear Aircraft Models
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User's manual for interactive LINEAR: A FORTRAN program to derive linear aircraft models

An interactive FORTRAN program that provides the user with a powerful and flexible tool for the linearization of aircraft aerodynamic models is documented in this report. The program LINEAR numerically determines a linear system model using nonlinear equations of motion and a user-supplied linear or nonlinear aerodynamic model. The nonlinear equations of motion used are six-degree-of-freedom equations with stationary atmosphere and flat, nonrotating earth assumptions. The system model determined by LINEAR consists of matrices for both the state and observation equations. The program has been designed to allow easy selection and definition of the state, control, and observation variables to be used in a particular model.

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	Patterson, Brian P. (NASA Hugh L. Dryden Flight Research Facility Edwards, CA, United States)
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Robert F. Antoniewicz,
Eugene L. Duke,
and Brian P. Patterson
Ames Research Center
Dryden Flight Research Facility
Edwards, California

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An interactive FORTRAN program that provides the user with a powerful and flexible tool for the linearization of aircraft aerodynamic models is documented in this report. The program LINEAR numerically determines a linear system model using nonlinear equations of motion and a user-supplied linear or nonlinear aerodynamic model. The nonlinear equations of motion used are six-degree-of-freedom equations with stationary atmosphere and flat, nonrotating earth assumptions. The system model determined by LINEAR consists of matrices for both the state and observation equations. The program has been designed to allow easy selection and definition of the state, control, and observation variables to be used in a particular model.

INTRODUCTION

The program LINEAR described in this report was developed at the Dryden Flight Research Facility of the NASA Ames Research Center to provide a standard, documented, and verified tool to derive linear models for aircraft stability analysis and control law design. This development was undertaken to address the need for the aircraft specific linearization programs common in the aerospace industry. Also, the lack of available documented linearization programs provided a strong motivation for the development of LINEAR; in fact, the only available documented linearization program that was found in an extensive literature search of the field is that of Kalviste (1980).

Linear system models of aircraft dynamics and sensors are an essential part of both vehicle stability analysis and control law design. These models define the aircraft system in the neighborhood of an analysis point and are determined by the linearization of the nonlinear equations defining vehicle dynamics and sensors. This report describes a FORTRAN program that provides the user with a powerful
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NASA Technical Paper 2835, september 1988
User's Manual for LINEAR, a FORTRAN Program to Derive Linear Aircraft Models
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15. Supplementary Notes

A listing of the program Interactive LINEAR is provided in the microfiche supplement included with this report (4 sheets total).

16. Abstract

An interactive FORTRAN program that provides the user with a powerful and flexible tool for the linearization of aircraft aerodynamic models is documented in this report. The program LINEAR numerically determines a linear system model using nonlinear equations of motion and a user-supplied linear or nonlinear aerodynamic model. The nonlinear equations of motion used are six-degree-of-freedom equations with stationary atmosphere and flat, nonrotating earth assumptions. The system model determined by LINEAR consists of matrices for both the state and observation equations. The program has been designed to allow easy selection and definition of the state, control, and observation variables to be used in a particular model.

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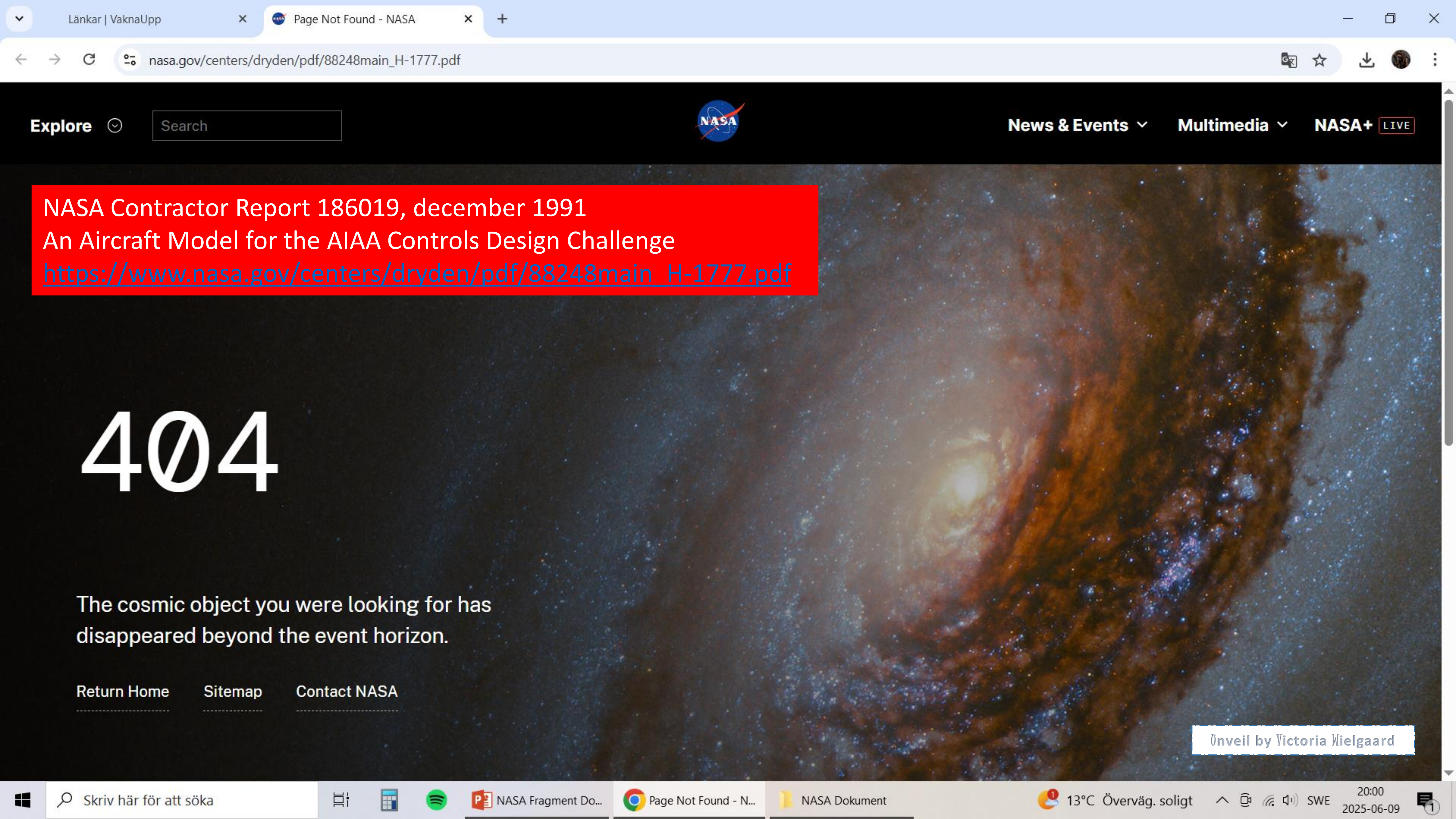
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Randal W. Brumbaugh
PRC Inc.,
Edwards, California

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National Aeronautics and
Space Administration

Dryden Flight Research Facility
Edwards, California 93523-0273

$$p_s = p\cos\alpha + r\sin\alpha$$

$$q_s = q$$

$$r_s = -p\sin\alpha + r\cos\alpha$$

Equations of Motion and Atmospheric Model

The nonlinear equations of motion used in this model are general six-degree-of-freedom equations representing the flight dynamics of a rigid aircraft flying in a stationary atmosphere over a flat, nonrotating Earth. These equations of motion were derived by Etkin, and the derivation is detailed in Duke, Antoniewicz, and Krambeer. The equations for each variable in the state vector are given in the following.

The following equations for rotational acceleration are used

$$\begin{aligned}\dot p = [&(\Sigma L)I_1 + (\Sigma M)I_2 + (\Sigma N)I_3 - p^2(I_{xz}I_2 - I_{xy}I_3)\\ &+ pq(I_{xz}I_1 - I_{yz}I_2 - D_zI_3)\\ &- pr(I_{xy}I_1 + D_yI_2 - I_{yz}I_3) + q^2(I_{yz}I_1 - I_{xy}I_3)\\ &- qr(D_xI_1 - I_{xy}I_2 + I_{xz}I_3)\\ &- r^2(I_{yz}I_1 - I_{xz}I_2)]\,/\det I\end{aligned}$$

$$\begin{aligned}\dot q = [&(\Sigma L)I_2 + (\Sigma M)I_4 + (\Sigma N)I_5 - p^2(I_{xz}I_4 - I_{xy}I_5)\\ &+ pq(I_{xz}I_2 - I_{yz}I_4 - D_zI_5)\\ &- pr(I_{xy}I_2 + D_yI_4 - I_{yz}I_5) + q^2(I_{yz}I_2 - I_{xy}I_5)\\ &- qr(D_xI_2 - I_{xy}I_4 + I_{xz}I_5)\\ &- r^2(I_{yz}I_2 - I_{xz}I_4)]\,/\det I\end{aligned}$$

$$\begin{aligned}\dot r = [&(\Sigma L)I_3 + (\Sigma M)I_5 + (\Sigma N)I_6 - p^2(I_{xz}I_5 - I_{xy}I_6)\\ &+ pq(I_{xz}I_3 - I_{yz}I_5 - D_zI_6)\\ &- pr(I_{xy}I_3 + D_yI_5 - I_{yz}I_6) + q^2(I_{yz}I_3 - I_{xy}I_6)\\ &- qr(D_xI_3 - I_{xy}I_5 + I_{xz}I_6)\\ &- r^2(I_{yz}I_3 - I_{xz}I_5)]\,/\det I\end{aligned}$$

Where ΣL, ΣM and ΣN are the aerodynamic total mo-
...ctively, in-

$- I_zI_{xy}^2$

$$D_z = I_y - I_x$$

The translational acceleration equations used are

$$\begin{aligned}\dot V = [&-D\cos\beta + Y\sin\beta + X_T\cos\alpha\cos\beta\\ &+ Y_T\sin\beta + Z_T\sin\alpha\cos\beta\\ &- mg(\sin\theta\cos\alpha\cos\beta - \cos\theta\sin\phi\sin\beta\\ &\quad - \cos\theta\cos\phi\sin\alpha\cos\beta)]\,/m\end{aligned}$$

$$\begin{aligned}\dot\alpha = [&-L + Z_T\cos\alpha - X_T\sin\alpha\\ &+ mg(\cos\theta\cos\phi\cos\alpha + \sin\theta\sin\alpha)]\,/Vm\cos\beta\\ &+ q - \tan\beta(p\cos\alpha + r\sin\alpha)\end{aligned}$$

$$\begin{aligned}\dot\beta = [&D\sin\beta + Y\cos\beta - X_T\cos\alpha\sin\beta\\ &+ Y_T\cos\beta - Z_T\sin\alpha\sin\beta\\ &+ mg(\sin\theta\cos\alpha\sin\beta + \cos\theta\sin\phi\cos\beta\\ &\quad - \cos\theta\cos\phi\sin\alpha\sin\beta)]\,/Vm\\ &+ p\sin\alpha - r\cos\alpha\end{aligned}$$

where α, β, θ, and ϕ are angles of attack, sideslip, pitch, and roll, respectively; X_T, Y_T, and Z_T are thrust along the x-, y-, and z-body axes; and D is drag force, g gravitational acceleration, L total aerodynamic lift, m total aircraft mass, V total velocity, and Y sideforce.

The equations defining the vehicle attitude rates are

$$\dot\theta = q\cos\phi - r\sin\phi$$

$$\dot\psi = q\sin\phi\sec\theta + r\cos\phi\sec\theta$$

$$\dot\phi = p + q\sin\phi\tan\theta + r\cos\phi\tan\theta$$

where ψ is heading angle.

The equations defining the Earth-relative velocities are

$$\begin{aligned}\dot h = V(&\cos\beta\cos\alpha\sin\theta - \sin\beta\sin\phi\cos\theta\\ &- \cos\beta\sin\alpha\cos\phi\cos\theta)\end{aligned}$$

$$\begin{aligned}\dot x = V[&\cos\beta\cos\alpha\cos\theta\cos\psi\\ &+ \sin\beta(\sin\phi\sin\theta\cos\psi - \cos\phi\sin\psi)\\ &+ \cos\beta\sin\alpha(\cos\phi\sin\theta\cos\psi + \sin\phi\sin\psi)]\end{aligned}$$

NASA Contractor Report 3073, december 1978
Investigation of Aircraft Landing in Variable Wind Fields
https://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/19790005472.pdf

Investigation of aircraft landing in variable wind fields

A digital simulation study is reported of the effects of gusts and wind shear on the approach and landing of aircraft. The gusts and wind shear are primarily those associated with wind fields created by surface wind passing around bluff geometries characteristic of buildings. Also, flight through a simple model of a thunderstorm is investigated. A two-dimensional model of aircraft motion was represented by a set of nonlinear equations which accounted for both spatial and temporal variations of winds. The landings of aircraft with the characteristics of a DC-8 and a DHC-6 were digitally simulated under different wind conditions with fixed and automatic controls. The resulting deviations in touchdown points and the controls that are required to maintain the desired flight path are presented. The presence of large bluff objects, such as buildings in the flight path is shown to have considerable effect on aircraft landings.

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Authors
Frost, W. *(Tennessee Univ. Space Inst. Tullahoma, TN, United States)*
Reddy, K. R. *(Tennessee Univ. Space Inst. Tullahoma, TN, United States)*
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Investigation of Aircraft Landing in Variable Wind Fields

Walter Frost and Kapuluru Ravikumar Reddy

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CHAPTER II

AIRCRAFT LANDING MODEL

1. Equations of Motion

The two-dimensional model for aircraft motion presented in this section follows the general form developed by Frost [12]. It accounts for both vertical and horizontal mean wind components having both time and spatial variations.

The aircraft trajectory model employed in this study was derived based on the following assumptions:

a) The earth is flat and non-rotating.

b) The acceleration of gravity, g, is constant (9.8 m/sec^2).

c) Air density is constant (1.23 kg/m^3).

d) The airframe is a rigid body.

e) The aircraft is constrained to motion in the vertical plane.

f) The aircraft has a symmetry plane (the x-z plane).

g) The mass of the aircraft is constant.

...re for steady-state flight.

... acting on the aircraft. These

Engineering Experiment Station, Georgia Institute of Technology, Prepared for NASA; april 1965 Atmospheric Oscillations https://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/19650015408.pdf

Atmospheric oscillations

Motion, continuity, and adiabatic equations for upper atmospheric oscillation

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mations on both the equations and the model of the atmosphere described. The simplified equations will be discussed first; then the more general approach will be described.

A model frequently used is that of a flat, nonrotating earth. The temperature is assumed either to be constant, to increase or decrease monotonically with altitude, or to be stratified. Gravity is usually considered to be constant. Density and pressure are usually considered to vary exponentially with altitude.

The most one can profitably simplify the problem is to consider an isothermal atmosphere, plane level surfaces, and a nonrotating earth. This case has been handled by Eckart [1960], Lamb [1932], and Hines [1960]. The simplification is not valid for small effects, but general, large effects may be described and discussed. Hines tried with apparent success to relate his results to effects observed experimentally. Eckart went over nearly the same derivation as Hines but included more detail. However, Hines used notation that is more physically meaningful. Both used linearized equations for small perturbations on a stationary

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NASA Technical Memorandum 81238, novebmer 1980
A Mathematical Model of the CH-53 Helicopter
https://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/19810003557.pdf

A mathematical model of the CH-53 helicopter

A mathematical model suitable for real time simulation of the CH-53 helicopter is presented. This model, which is based on modified nonlinear classical rotor theory and nonlinear fuselage aerodynamics, will be used to support terminal-area guidance and navigation studies on a fixed-base simulator. Validation is achieved by comparing the model response with that of a similar aircraft and by a qualitative comparison of the handling characteristics made by experienced pilots.

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Authors Sturgeon, W. R. (NASA Ames Research Center Moffett Field, CA, United States)
Phillips, J. D. (NASA Ames Research Center Moffett Field, CA, United States)
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(NASA-TM-81238) A MATHEMATICAL MODEL OF THE CH-53 HELICOPTER (NASA) 60 p HC A04/MF A01 CSCL 01C

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A Mathematical Model of the CH-53 Helicopter

William R. Sturgeon
James D. Phillips, Ames Research Center, Moffett Field, California

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Equations of Motion

The helicopter equations of motion are given in body axes with respect to a flat, nonrotating Earth. The helicopter is considered a rigid body with mass symmetry about the $x_h - z_h$ plane. The effects due to the engine angular momentum are neglected.

Translational acceleration- The translational equations of motion are

$$C_{h/e}\begin{bmatrix}0\\0\\mg\end{bmatrix} + \begin{bmatrix}X\\Y\\Z\end{bmatrix}_{f,\,h} + \begin{bmatrix}X\\Y\\Z\end{bmatrix}_{r,\,h} = m\left[\begin{bmatrix}\dot{u}\\\dot{v}\\\dot{w}\end{bmatrix}_{cg,\,h} + \begin{bmatrix}0 & -r & q\\ r & 0 & -p\\ -q & p & 0\end{bmatrix}_h \begin{bmatrix}u\\v\\w\end{bmatrix}_{cg,\,h}\right] \quad (45)$$

where

$$C_{h/e} = \begin{bmatrix}\cos\theta\cos\psi & \cos\theta\sin\psi & -\sin\theta\\ \sin\phi\sin\theta\cos\psi - \cos\phi\sin\psi & \cos\phi\cos\psi + \sin\phi\sin\theta\sin\psi & \sin\phi\cos\theta\\ \cos\phi\sin\theta\cos\psi + \sin\phi\sin\psi & \cos\phi\sin\theta\sin\psi - \sin\phi\cos\psi & \cos\phi\cos\theta\end{bmatrix}_h \quad (46)$$

and ϕ_h, θ_h, and ψ_h are the Euler angles that define the orientation of the body axis system (fig. 3). The fuselage aerodynamic forces are given by equation (10), and the rotor forces, which include those due to both main and tail rotors, are given by equation (31). Equation (45) can be rearranged to yield

$$\begin{bmatrix}\dot{u}\\\dot{v}\\ \end{bmatrix} = \frac{1}{m}\left[\begin{bmatrix}X\\Y\\ \end{bmatrix} + \begin{bmatrix}X\\Y\\ \end{bmatrix}\right] - \begin{bmatrix}0 & -r & q\\ r & 0 & -p\\ -q & p & 0\end{bmatrix}_h \begin{bmatrix}u\\v\\w\end{bmatrix}_{cg,} + C_{h/e}\begin{bmatrix}0\\0\\g\end{bmatrix} \quad (47)$$

...nertial velocity, in body coordi-

NASA Technical Paper 2002-210718, juni 2002
Stability and Control Estimation Flight Test Results for the SR-71 Aircraft with Externally Mounted Experiments
https://www.nasa.gov/centers/dryden/pdf/88733main_H-2465.pdf

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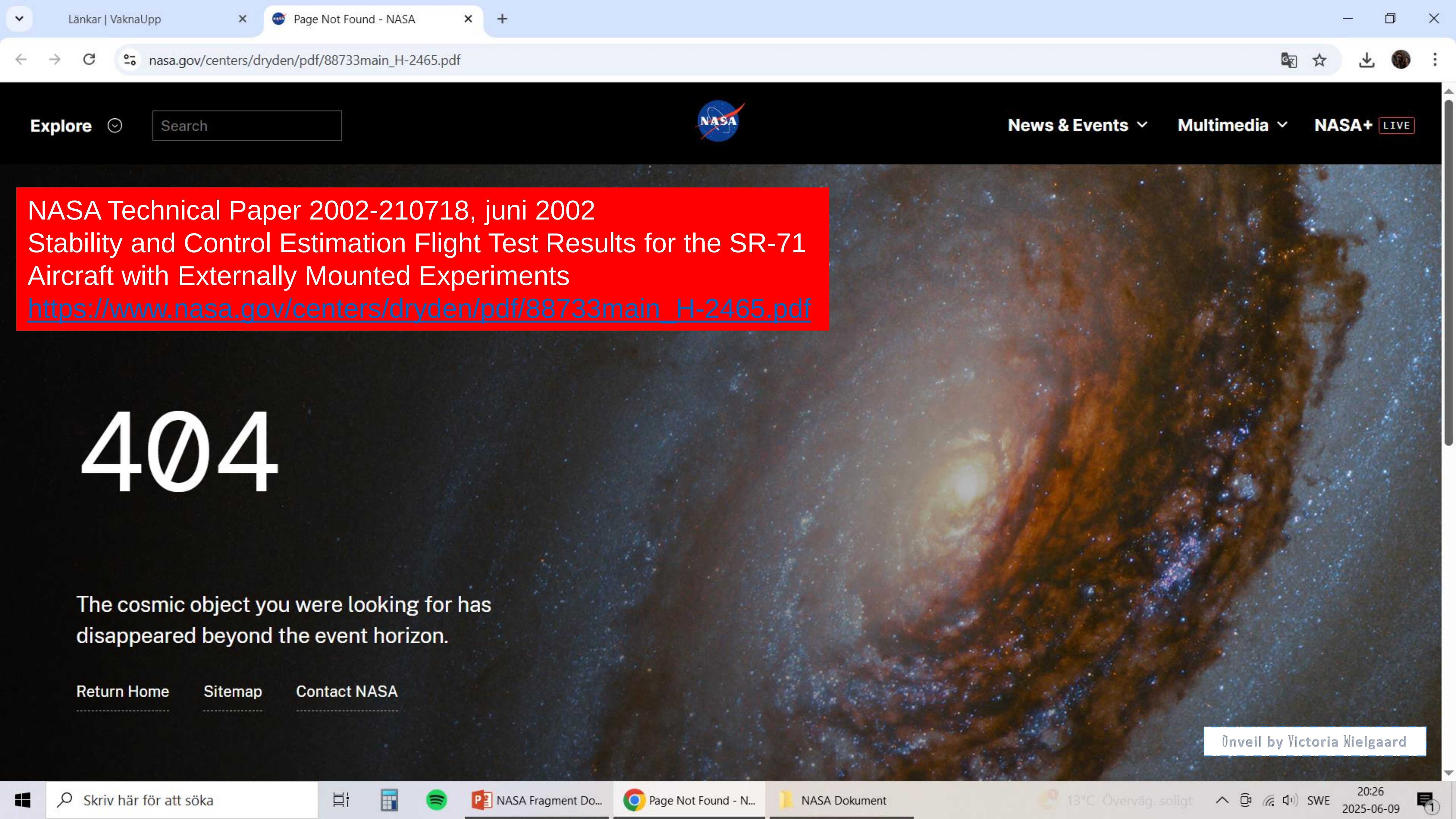

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Stability and Control Estimation Flight Test Results for the SR-71 Aircraft With Externally Mounted Experiments

Timothy R. Moes and Kenneth Iliff
NASA Dryden Flight Research Center
Edwards, California

NASA Technical Paper 2002-210718, juni 2002
Stability and Control Estimation Flight Test Results for the SR-71 Aircraft with Externally Mounted Experiments
https://www.nasa.gov/centers/dryden/pdf/88733main_H-2465.pdf

The estimator also provides a measure of the reliability of each estimate based on the information obtained from each dynamic maneuver. This measure of reliability is called the Cramér-Rao bound (ref. 6). The Cramér-Rao bound is a measure of relative, not absolute, accuracy. A large Cramér-Rao bound indicates poor information content in the maneuver for the derivative estimate.

Equations of Motion

The aircraft equations of motion used in the PID analysis are derived from a general system of nine coupled, nonlinear differential equations that describe the aircraft motion (ref. 4). These equations

assume a rigid vehicle and a flat, nonrotating Earth. The time rate of change of mass and inertia is assumed negligible. The SR-71 configurations studied herein, like most aircraft, are basically symmetric about the vertical-centerline plane. This symmetry is used, along with small angle approximations, to separate the equations of motion into two largely independent sets describing the longitudinal and lateral-directional motions of the aircraft. The equations of motion are written in body axes referenced to the *CG* and include both state and response equations. The applicable equations of motion are as follows for the longitudinal and lateral-directional axes:

Longitudinal state equations:

NASA Technical Paper 2002-210718, juni 2002
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$$\ldots s\alpha + r\sin\alpha) + \frac{gR}{V\cos\beta}(\cos\phi\cos\theta\cos\alpha + \sin\theta\sin\alpha) \qquad (5)$$

$$[rp(I_z - I_x) + (r^2 - p^2)I_{xz}]/R \qquad (6)$$

NASA Technical Memorandum 100996, maj 1988
Flight Testing a VSTOL Aircraft to Identify a Full-Envelope Aerodynamic Model.
https://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/19880014378.pdf

Flight testing a V/STOL aircraft to identify a full-envelope aerodynamic model

Flight-test techniques are being used to generate a data base for identification of a full-envelope aerodynamic model of a V/STOL fighter aircraft, the YAV-8B Harrier. The flight envelope to be modeled includes hover, transition to conventional flight and back to hover, STOL operation, and normal cruise. Standard V/STOL procedures such as vertical takeoff and landings, and short takeoff and landings are used to gather data in the powered-lift flight regime. Long (3 to 5 min) maneuvers which include a variety of input types are used to obtain large-amplitude control and response excitations. The aircraft is under continuous radar tracking; a laser tracker is used for V/STOL operations near the ground. Tracking data are used with state-estimation techniques to check data consistency and to derive unmeasured variables, for example, angular accelerations. A propulsion model of the YAV-8B's engine and reaction control system is used to isolate aerodynamic forces and moments for model identification. Representative V/STOL flight data are presented. The processing of a typical short takeoff and slow landing maneuver is illustrated.

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Authors
Mcnally, B. David
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Bach, Ralph E., Jr.
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rudder frequency sweep and a rudder-fixed aileron frequency sweep provide additional lateral-axis excitation. The pilot uses aileron control to hold the wings-level condition during the sideslip and the rudder sweep portions of the maneuver. An initial roll angle of 40° was chosen for the aileron sweep. Thrust is added at the end of the maneuver to return to the nominal trim point. Thirteen of these lateral maneuvers were performed.

Preliminary Processing

Following real-time acquisition of data during flight test, each recorded maneuver, with instrument calibrations added, is converted to engineering units and made available to researchers as a raw flight-data file. An interactive program called PRODAT (PROcess DATa) is used to read the raw flight-data file, identify wild points, and filter data records. Data are then stored in a "processed" file of selected channels at a submultiple of the main-frame sample rate. PRODAT runs on a VAX-8650. Processing begins by removing wild points from the records. Several methods are

State Estimation

The next step in the processing of each maneuver is to apply SMACK[19-21] (SMoothing for AirCraft Kinematics), a state-estimation program developed at Ames Research Center, to check data consistency and derive unmeasured variables. SMACK runs on a Cray-XMP computer. State estimation in this paper refers to a process that solves a state model,

$$\dot{x} = f(x,w), \quad x(t_o) = x_o \qquad (2)$$

such that $h(x)$ in the measurement model

$$z = h(x) + v \qquad (3)$$

suitably matches the data record over a time interval (t_o,t_f), usually in a least-squares error sense.[7,22] In Eq. (2), x is the state vector and w is a vector that represents unknown forcing functions (derivatives of unmeasured variables, e.g., angular accelerations). For aircraft problems, the state and measurement models together

represent the kinematics of a rigid body for describing motion over a flat, nonrotating Earth. In the SMACK formulation, the state model consists of Euler angles and position variables and their derivatives. When flightpath winds are to be identified, the state model is augmented by wind velocities and accelerations. The measurement model

measured in flight. It should be noted that the propulsion model provides only thrust forces and moments. Any thrust-induced aerodynamic effects are to be included in the VSRA aerodynamic model.

Inputs to the ENCAL routine include all the air-data, reaction control, engine, and weight ...sted in Table 1. Outputs to the ...-data file are the three body-axis ...gine forces and moments. ENCAL ... aircraft weight and inertias, and ...n center-of-gravity (c.g.) location ...r. variations are based on manufac-

NASA Technical Memorandum 100996, maj 1988

Flight Testing a VSTOL Aircraft to Identify a Full-Envelope Aerodynamic Model.

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Singular Arc Time-Optimal Climb Trajectory of Aircraft in a Two-Dimensional Wind Field
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Singular-Arc Time-Optimal Trajectory of Aircraft in Two-Dimensional Wind Field

This paper presents a study of a minimum time-to-climb trajectory analysis for aircraft flying in a two-dimensional altitude dependent wind field. The time optimal control problem possesses a singular control structure when the lift coefficient is taken as a control variable. A singular arc analysis is performed to obtain an optimal control solution on the singular arc. Using a time-scale separation with the flight path angle treated as a fast state, the dimensionality of the optimal control solution is reduced by eliminating the lift coefficient control. A further singular arc analysis is used to decompose the original optimal control solution into the flight path angle solution and a trajectory solution as a function of the airspeed and altitude. The optimal control solutions for the initial and final climb segments are computed using a shooting method with known starting values on the singular arc The numerical results of the shooting method show that the optimal flight path angle on the initial and final climb segments are constant. The analytical approach provides a rapid means for analyzing a time optimal trajectory for aircraft performance.

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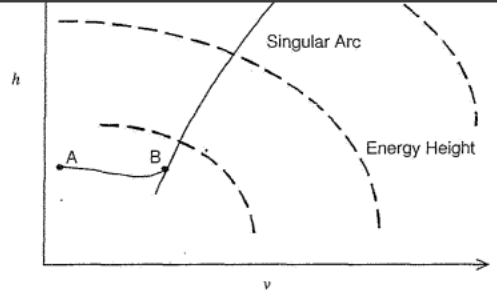

Fig. 1 - Time-Optimal Energy Climb Path

Within the framework of the Pontryagin's minimum principle, the singular-arc optimal control method is an intermediate method for the trajectory optimization. The existence of a singular arc in the time optimal control can simplify the trajectory optimization significantly. Briefly, the singular arc is described by a *switching function* that minimizes a Hamiltonian function when the Hamiltonian function is linear with respect to a control variable.

In this study, we will examine an aspect of the minimum time-to-climb problem for an aircraft flying in the presence of a two-dimensional atmospheric wind field. An analytical solution for the singular arc is obtained. Wind patterns at a local airport can affect the climb performance of aircraft. While the time-optimal climb problems have been thoroughly studied in flight mechanics, the effect of winds are usually not included in these studies. A solution method of a minimum-time to climb will then be presented for computing a minimum time-to-climb flight trajectory.

II. Singular Arc Optimal Control

In our minimum time-to-climb problem, the aircraft is modeled as a point mass and the flight trajectory is strictly confined in a vertical plane on a non-rotating, flat earth. The change in mass of the aircraft is neglected and the engine thrust vector is assumed to point in the direction of the aircraft velocity vector. In addition, the aircraft is assumed to fly in an atmospheric wind field comprising of both horizontal and vertical components that are altitude-dependent. The horizontal wind component normally comprises a longitudinal and lateral component. We assume that the aircraft motion is symmetric so that the lateral wind component is not included. Thus, the pertinent equations of motion for the problem are defined in its state variable form as

$$\dot{h} = v\sin\gamma + w_h \qquad (1)$$

$$\dot{v} = \frac{T - D - W\sin\gamma}{m} - \dot{w}_x\cos\gamma - \dot{w}_h\sin\gamma \qquad (2)$$

$$L - W\cos\gamma \quad \dot{w}_x\sin\gamma - \dot{w}_h\cos\gamma \qquad (3)$$

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Determination of angles of attack and sideslip from radar data and a roll-stabilized platform

Equations for angles of attack and sideslip relative to both a rolling and nonrolling body axis system are derived for a flight vehicle for which radar and gyroscopic attitude data are available. The method is limited to application where a flat, nonrotating earth may be assumed. The gyro measures attitude relative to an inertial reference in an Euler angle sequence. In particular, a pitch, yaw, and roll sequence is used as an example in the derivation. Sample calculations based on flight data are presented to illustrate the method. Results obtained with the present gyro method are compared with another technique that uses onboard camera data.

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16. Abstract

Equations for angles of attack and sideslip relative to both a rolling and nonrolling body axis system are derived for a flight vehicle for which radar and gyroscopic-attitude data are available. The method is limited, however, to application where a flat, nonrotating earth may be assumed. The gyro considered measures attitude relative to an inertial reference in an Euler angle sequence. In particular, a pitch, yaw, and roll sequence is used as an example in the derivation. Sample calculations based on flight data are presented to illustrate the method. Results obtained with the present gyro method are compared with another technique that uses onboard-camera data.

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