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DEVELOPMENT OF A FLIGHT TEST MANEUVER AUTOPILOT FOR AN F-15 AIRCRAFT*

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An autopilot can be used to provide precise control to meet the demanding requirements of flight research maneuvers with high-performance aircraft. This paper presents the development of control laws for a flight test maneuver autopilot for an F-15 aircraft. A linear quadratic regulator approach is used to develop the control laws within the context of flight test maneuver requirements by treating the maneuver as a finite time tracking problem with regulation of state rates. Results are presented to show the effectiveness of the controller in insuring acceptable aircraft performance during a maneuver.

Introduction

Conventional piloting techniques are often inadequate to meet the demanding requirements of flight research maneuvers with high-performance aircraft. These maneuvers frequently require precise control of onset rates in extreme flight conditions. Thus, the pilot may be trying to control an aircraft at high angles of attack and high g's while attempting to increase normal acceleration at a prescribed rate through a maneuver specified to the very limits of the accuracy of the cockpit instruments.

A new flight test technique to aid the pilot during these maneuvers was developed at the Dryden Flight Research Facility of the NASA Ames Research Center (Ames-Dryden).[1] The essence of this technique is the application of an autopilot to provide precise control during the required flight test maneuvers. The flight test maneuver autopilot (FTMAP) is designed to provide precise, repeatable control of a high-performance aircraft during certain prescribed maneuvers so that a large quantity of data can be obtained in a minimum of flight time.

The FTMAP can be used for various maneuvers, such as straight-and-level flight, level accelerations and decelerations, pushover pullups, excess-thrust windup turns, and thrust-limited turns. Each of these maneuvers comprise tracking certain states of the aircraft, holding certain states within prescribed values, and maintaining constraints on the derivatives of the states. For example, an excess-thrust windup turn is performed at constant altitude and Mach number with the angle of attack increasing at a specified rate to the final angle of attack.

This paper presents the development of the FTMAP control laws within the context of flight test maneuver requirements. The FTMAP design represents a well-defined and completely specified problem against which various design methodologies can be tested. Not only does this problem represent a real-world situation, but the FTMAP control law design problem also forces consideration of a significant design issue in aircraft controls — controlling a plant that changes as vehicle attitude or flight conditions change.

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Problem Formulation

The functional capability of the FTMAP is derived from its ability to generate control laws required to execute a maneuver. The FTMAP control laws are developed using the Ames-Dryden detailed nonlinear aerodynamic model of the F-15 aircraft. The model is linearized by trimming the aircraft at the desired flight condition and deriving linear models by numerical perturbation. The linear model may be represented in the standard state equation form as

$$\dot{x} = Ax + Bu \tag{1}$$

$$y = Hx \tag{2}$$

where the states x and inputs u represent perturbation around nominal trim values $\bar{x}$ and $\bar{u}$, respectively. Also, y is the output and A, B, and H are matrices of appropriate dimensions.

In order to perform a certain maneuver, a control algorithm is developed to take the aircraft from a certain specified initial trim state to another specified state. The FTMAP control law design can be formulated as an optimal tracking problem, in which it is desired to track a constant final state in finite time. It is also necessary to regulate the state rate in certain maneuvers.

Multivariable linear quadratic control theory is a powerful tool for the development of the flight test controller. The control design problem selects inputs u to drive the perturbation states x to the desired final state by optimizing a quadratic performance index. Because control of the derivatives of states is also desired, additional terms are added to the performance index and the resultant control laws are computed directly from the performance index.

Control Law Synthesis

An optimal tracking problem may be formulated as follows. Reiterating Eqs. 1 and 2, the system is given as

$$\dot{x} = Ax + Bu$$

$$y = Hx$$

It is desired to minimize a performance index of the form

$$J = 1/2 \int_0^T [e(t)^T Q e(t) + u(t)^T R u(t)]\, dt \qquad (3)$$

where e(t) is equal to z(t) - y(t), z(t) is the desired state to be tracked and is a known function, and Q and R are weighting matrices.

To track a constant final state z and also to be able to regulate the derivatives of the states, an additional term is added to the performance index to be minimized. The performance index may be written as

$$J = 1/2 \int_0^T [e(t)^T Q e(t) + u(t)^T R u(t) + (E\dot{x} - E_0)^T S (E\dot{x} - E_0)]\, dt \qquad (4)$$

Here,

$e = z - y(t)$
$Q \geq 0$
$R > 0$
$S \geq 0$

Also, z is the desired constant final state, and E_0 is the desired state rate.

If the performance index is minimized by using Lagrange multiplier, the optimal control u* that will derive the aircraft to the desired terminal condition is

$$u^* = -(R + B^T E^T SEB)^{-1}[B^T P(t) + B^T E^T SEAx - B^T E^T SE_0]$$

$$= -R_1^{-1}[B^T P(t) + B^T E^T SEAx - B^T E^T SE_0] \qquad (5)$$

where

$$R_1 = (R + B^T E^T SEB)$$

$$P(t) = K(t)x(t) - g(t)$$

and K and g are obtained from the solution of the following equations using appropriate terminal conditions.[2] Hence,

$$\dot{K} + \tilde{A}^T K + K\tilde{A} - K\tilde{E}K + \tilde{Q} = 0 \qquad (6)$$

$$\dot{g}(t) + (\tilde{A} - \tilde{E}K)^T g(t) + H^T \tilde{Q} Z - LE_0 = 0 \qquad (7)$$

where

$$\tilde{A} = A - BR_1^{-1} B^T E^T SEA$$

$$\tilde{E} = BR_1^{-1} B^T$$

$$\tilde{Q} = H^T QH + A^T E^T SEA - A^T E^T SEBR_1^{-1} B^T E^T SEA$$

$$L = -A^T E^T S + A^T E^T SEBR_1^{-1} B^T E^T S + K^T B^T R_1^{-1} B^T E^T S$$

However, the optimal control u* need not be derived by a finite time formulation but may be obtained by employing steady-state control laws. This is equivalent to assuming that the z(T) persists as a constant value for a long period. With the foregoing assumption, K is a constant matrix that satisfies the algebraic Riccati equation and $\dot{g}(t) \approx 0$. Thus, the simplified control law becomes

$$u^*(t) = -C_1 x + C_2 z + C_3 E_0 \qquad (8)$$

where

$$C_1 = R_1^{-1} B^T K + R_1^{-1} B^T E^T SEA$$

$$C_2 = -R_1^{-1} B^T L_1^{-1} H^T Q$$

$$C_3 = R_1^{-1}(B^T L_1^{-1} L + B^T E^T SE)$$

$$L_1 = (\tilde{A} - \tilde{E}K)^T$$

Application of Control Laws

The linearized equations of motion of an F-15 aircraft are given by Eq. 1

$$\dot{x} = Ax + Bu$$

where

$$x = \begin{bmatrix} v \\ \alpha \\ q \\ \theta \\ h \\ \beta \\ p \\ r \\ \phi \end{bmatrix} \begin{array}{l} = \text{velocity, ft/sec} \\ = \text{angle of attack, deg} \\ = \text{pitch rate, deg/sec} \\ = \text{pitch angle, deg} \\ = \text{altitude, ft} \\ = \text{sideslip angle, deg} \\ = \text{roll rate, deg/sec} \\ = \text{yaw rate, deg/sec} \\ = \text{roll angle, deg} \end{array}$$

$$u = \begin{bmatrix} \delta_a \\ \delta_e \\ \delta_r \\ \delta_T \end{bmatrix} \begin{array}{l} = \text{aileron deflection} \\ = \text{elevator deflection} \\ = \text{rudder deflection} \\ = \text{throttle displacement} \end{array}$$

For a flight condition corresponding to an altitude of 20,000 ft and Mach 0.8, the matrices A and B are as follows. For matrix A, columns 1 to 9 are

$$\begin{bmatrix} -0.0108 & 24.1966 & 0.0000 & -32.1129 & 0.0000 & 0.0000 & 0.0000 & 0.0000 & 0.0000 \\ -0.0001 & -1.0942 & 1.0000 & 0.0001 & 0.0000 & 0.0000 & 0.0000 & 0.0000 & 0.0000 \\ -0.0001 & -3.2862 & -2.1922 & 0.0009 & 0.0000 & 0.0000 & 0.0000 & 0.0000 & 0.0000 \\ 0.0000 & 0.0000 & 1.0000 & 0.0000 & 0.0000 & 0.0000 & 0.0000 & 0.0000 & 0.0000 \\ 0.0000 & -829.5390 & 0.0000 & 829.5390 & 0.0000 & 0.0000 & 0.0000 & 0.0000 & 0.0000 \\ 0.0000 & 0.0000 & 0.0000 & 0.0000 & -0.2337 & 0.0358 & -0.9994 & 0.0387 & 0.0000 \\ 0.0000 & 0.0000 & 0.0000 & 0.0000 & -40.0103 & -2.1420 & 1.2406 & 0.0000 & 0.0000 \\ 0.0000 & 0.0000 & 0.0000 & 0.0000 & 9.0098 & -0.0340 & -0.6040 & 0.0000 & 0.0000 \\ 0.0000 & 0.0000 & 0.0000 & 0.0000 & 0.0000 & 1.0000 & 0.0358 & 0.0000 & 0.0000 \end{bmatrix}$$

Matrix B is given by

$$\begin{bmatrix} 0.0000 & -1.0734 & 0.0000 & 0.3792 \\ 0.0000 & -0.1504 & 0.0000 & 0.0000 \\ 0.0000 & -16.1223 & 0.0000 & 0.0000 \\ 0.0000 & 0.0000 & 0.0000 & 0.0000 \\ 0.0000 & 0.0000 & 0.0000 & 0.0000 \\ -0.0022 & 0.0000 & -0.0388 & 0.0000 \\ 13.5934 & 0.0000 & -1.4674 & 0.0000 \\ 0.1488 & 0.0000 & -4.5577 & 0.0000 \\ 0.0000 & 0.0000 & 0.0000 & 0.0000 \end{bmatrix}$$

Three sets of trajectories show the results of the application of developed control laws to the aircraft. In the first case, it is necessary to hold the aircraft at trim conditions of the specified flight condition. The flight condition corresponds to an altitude of 20,000 ft and a Mach number of 0.8. The aircraft is in straight-and-level flight. The control law design is intended to maintain the aircraft at the specified altitude and velocity.

For this first case, figures 1 and 2 show the variation in velocity and altitude, respectively, when the aircraft is required to be flown straight and level at a given altitude and Mach number. Figures 3 and 4 show the corresponding variations in the elevator deflection and throttle displacement, respectively. The aircraft is held to the trim values by the controller designed.

In the second case, the flight condition corresponds to an altitude of 20,000 ft and the initial Mach number is 0.5. The aircraft is to be accelerated to Mach 0.8 while it is held at the specified altitude. Because of the variation of the parameters while the plane accelerates, the feedback gains were updated midway through the maneuver. Figures 5 and 6 show the variations of velocity and altitude as the aircraft undergoes the maneuver. The response was not considered satisfactory because of spikes observed at the time of update of gains. Figures 7 and 8 show the variations of the velocity and altitude for the same maneuver when the aircraft is initialized to the flight condition corresponding to Mach 0.8.

In the third case, the maneuver that was performed was the same as in the second case, except that a constraint was placed on the state rate. It is desired to maintain the Mach rate at 0.005 Mach per sec. Figures 9 and 10 show the variations in the velocity and altitude as the aircraft goes through the maneuver. The response obtained in this third case was considered satisfactory.

Concluding Remarks

This paper presents a synthesis technique that is applicable for control of an aircraft undergoing a specified maneuver. It assumes that a given maneuver can be modeled as a state trajectory to be tracked. The technique that is described uses the optimal regulator approach to the trajectory control problem. Results are presented for trajectory tracking for the aircraft undergoing altitude-hold level acceleration.

For more complex maneuvers, it is necessary to generate a set of state and control histories to serve as commands for the flight test maneuver autopilot (FTMAP), using a data base that consists of trim conditions. The maneuver model can be divided into a smaller set of state trajectories for suitable updating of aircraft parameters and gain scheduling. The development has the potential of simultaneously controlling multiple parameters to demanding tolerances.

References

[1]Duke, E.L., Jones, F.P., and Roncoli, R.B., "Development of a Flight Test Maneuver Autopilot for a Highly Maneuverable Aircraft," AIAA 21st Aerospace Sciences Meeting, Reno, Nev., Jan. 1983.

[2]Alag, G.S., and Duke, E.L., "Development of Control Laws for a Flight Test Maneuver Autopilot for an F-15 Aircraft," American Control Conference Proc., Boston, Mass., June 1985, pp. 1209-1210.

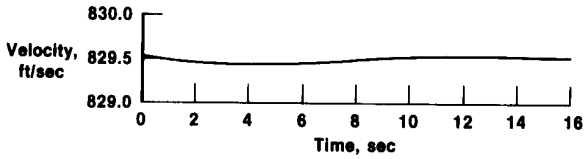

Fig. 1 Variations in velocity with time for straight-and-level flight condition of Mach 0.8 and 20,000-ft altitude

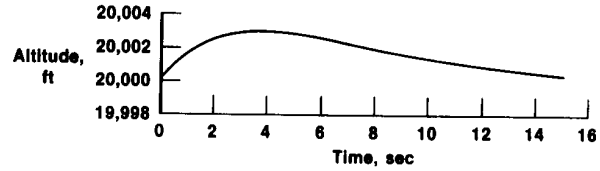

Fig. 2 Variations in altitude with time for straight-and-level flight condition of Mach 0.8 and 20,000-ft altitude

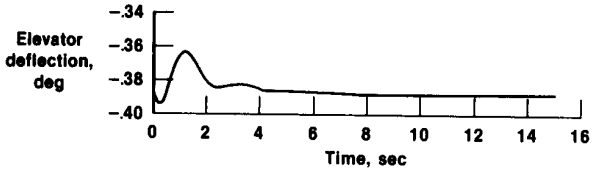

Fig. 3 Variations in elevator deflection with time for straight-and-level flight condition of Mach 0.8 and 20,000-ft altitude

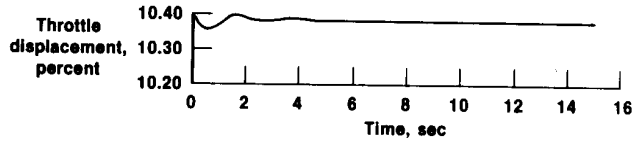

Fig. 4 Variations in throttle displacement with time for straight-and-level flight condition of Mach 0.8 and 20,000-ft altitude

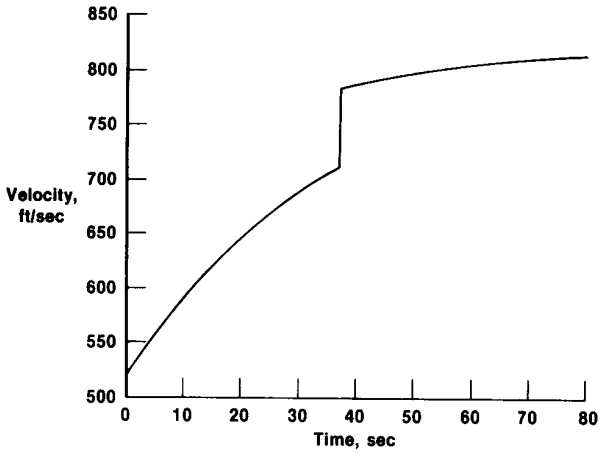

Fig. 5 Variations in velocity with time for flight condition of 20,000-ft altitude with initial Mach 0.5 accelerated to Mach 0.8 (gains updated midway through maneuver)

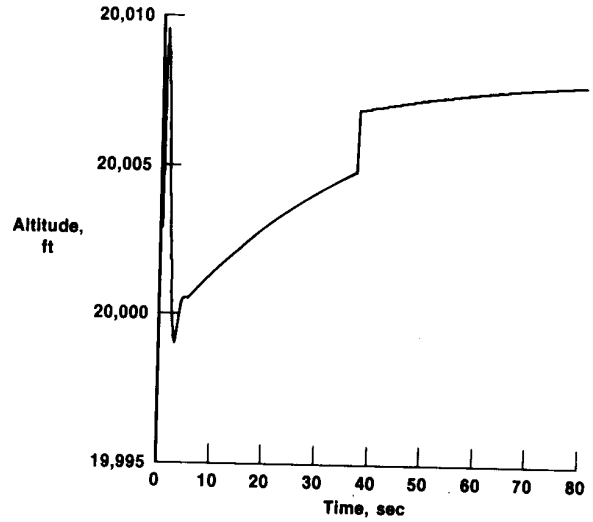

Fig. 6 Variations in altitude with time for flight condition of 20,000-ft with initial Mach 0.5 accelerated to Mach 0.8 (gains updated midway through maneuver)

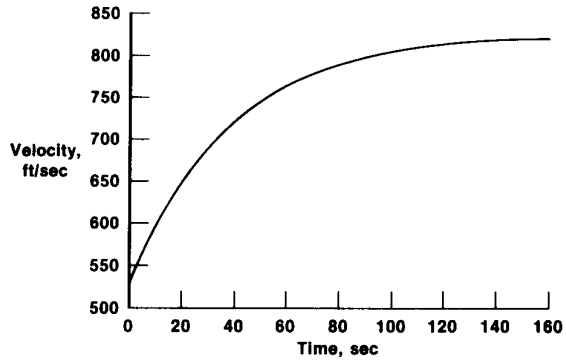

Fig. 7 Variations in velocity with time with aircraft parameters initialized to Mach 0.8 at 20,000-ft altitude

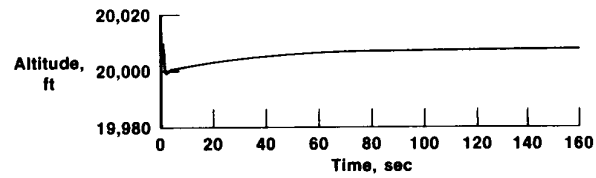

Fig. 8 Variations in altitude with time with aircraft parameters initialized to Mach 0.8 at 20,000-ft altitude

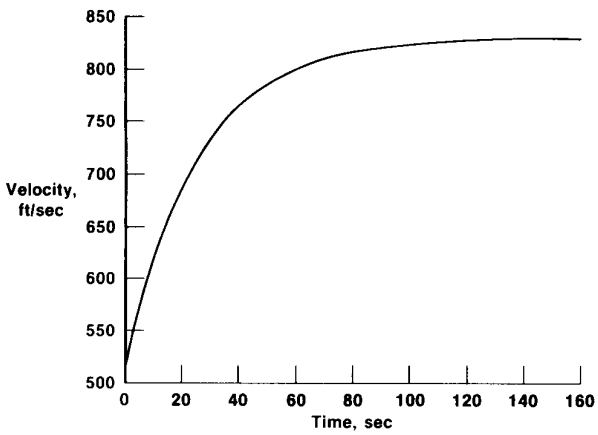

Fig. 9 Variations in velocity with time for flight condition of 20,000-ft altitude and Mach number increased from 0.5 to 0.8 at attempted state rate of 0.005 Mach per sec

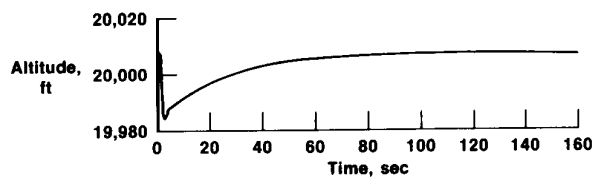

Fig. 10 Variations in altitude with time for flight condition of 20,000-ft altitude and Mach number increased from 0.5 to 0.8 at attempted state rate of 0.005 Mach per sec

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16. Abstract

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