A Design Methodology for Pitch Pointing Flight Control Systems

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A methodology is developed for designing pitch pointing flight control laws by using eigenstructure assignment and command generator tracking. Eigenvalues are chosen to obtain desired damping and rise time, and eigenvectors are chosen to decouple the pitch attitude and flight path angle. Feedforward gains are computed which ensure steady-state tracking of the pilot's command. The design methodology is illustrated by application to an AFTI F-16 aircraft.

Introduction

ADVANCED aircraft such as control configured vehicles (CCV) provide the capability for implementing specialized modes for bombing, strafing, and air-to-air combat. The control objectives for these advanced modes include the ability to command a chosen variable without significant motion in another specified variable. For the longitudinal dynamics of an aircraft, one such decoupled mode is pitch pointing, which is characterized by pitch attitude command with constant flight-path angle.

Recently, Smith and Anderson[1] have applied optimal control theory to the design of decoupled mode controllers for the AFTI/F-16 aircraft. This vehicle has both elevator and flaperon surfaces which allow for the implementation of advanced longitudinal control modes. Unfortunately, it is quite difficult to translate the military specifications for piloted vehicles, which are described in Ref. 2, into the form required for the optimal control problem. For example, such specifications as damping, frequency, and decoupling are not easily incorporated within a quadratic performance index. This results in quadratic weight adjustment by trial and error which can become computationally expensive. Furthermore, the design described in Ref. 1 requires either full-state feedback or a state estimator.

An alternative design, proposed by Ridgely, et al.,[3] utilizes high-gain error actuated control in conjunction with singular perturbation methods. This method allows eigenstructure assignment which is closely related to damping, frequency, and decoupling specifications. Unfortunately, this design yields both eigenvalues and feedback gains with magnitudes in excess of 1000.

In this paper we propose a design methodology which utilizes eigenstructure assignment[4] in conjunction with the command generator tracker.[5] For the pitch pointing controller design, the eigenvectors are chosen to obtain the desired decoupling, and the eigenvalues are chosen to obtain the desired damping and rise time. Feedforward gains are computed which ensure steady-state tracking of the pilot's command. The design methodology is illustrated by application to an AFTI F-16 aircraft.

Feedback Design Methodology

Consider the linear time invariant system described by

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

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

where $x \in R^n$, $u \in R^m$, $y \in R^r$.

We shall assume that

$$\text{rank}[B] = m \tag{3}$$

and

$$\text{rank}[C] = r \tag{4}$$

Under the above assumptions, the feedback problem can be stated as follows:

Given a set of desired eigenvalues $\{\lambda_i^d\}$, $i = 1, 2, \ldots, r$ and a corresponding set of desired eigenvectors, $\{v_i^d\}$, $i = 1, 2, \ldots, r$, find a real $m \times r$ matrix F such that the eigenvalues of $A + BFC$ contain $\{\lambda_i^d\}$ as a subset, and the corresponding eigenvectors of $A + BFC$ are close to the respective members of the set $\{v_i^d\}$.

The following theorem, due to Srinathkumar,[6] describes the number of eigenvalues and eigenvector entries which can be exactly assigned.

Theorem

Given the controllable and observable system described in Eqs. (1) and (2) and the assumptions that the matrices B and C are of full rank, then $\max(m, r)$ closed-loop eigenvalues can be assigned and $\max(m, r)$ eigenvectors can be partially assigned with $\min(m, r)$ entries in each vector arbitrarily chosen using constant gain output feedback.

In general, we may desire to exercise some control over more than $\min(m, r)$ entries in a particular eigenvector. Therefore we shall now discuss the problems of first characterizing desired eigenvectors v_i^d which can be assigned as closed loop eigenvectors, and second determining the best possible set of achievable eigenvectors in case a desired eigenvector v_i^d is not achievable.

The solution to these problems was shown in Ref. 4 and begins with the closed-loop system

$$\dot{x}(t) = (A + BFC)x(t) \tag{5}$$

For an eigenvalue/eigenvector pair, λ_i and v_i

$$(A + BFC)v_i = \lambda_i v_i \tag{6}$$

or

$$v_i = (\lambda_i I - A)^{-1} BFCv_i \tag{7}$$

Define the m-vector m_i as

$$m_i = FCv_i \tag{8}$$

Then Eq. (8) becomes

$$v_i = (\lambda_i I - A)^{-1} m_i \tag{9}$$

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The importance of Eq. (9) is the need for the eigenvector v_i to be in the subspace spanned by the columns of $(\lambda_i I - A)^{-1}B$. This subspace is of dimension m which is the number of independent control variables. Therefore the number of control variables determines the dimension of the subspace in which the achievable eigenvectors must reside. The orientation of the subspace is determined by the open-loop parameters described by A, B, and the desired closed-loop eigenvalue λ_i. We conclude that if we choose an eigenvector v_i which lies precisely in the subspace spanned by the columns of $(\lambda_i I - A)^{-1}B$, it will be achieved exactly.

In general, however, a desired eigenvector v_i^d will not reside in the prescribed subspace and, hence, cannot be achieved. Instead, a "best possible" choice for an achievable eigenvector is made. This "best possible" eigenvector is the projection of v_i^d onto the subspace spanned by the columns of $(\lambda_i I - A)^{-1}B$.

To further complicate the situation, complete specification of v_i^d is neither required nor known in most practical situations. When the designer is interested only in certain elements of the eigenvector, we assume that the desired eigenvector has a structure given by

$$v_i^d = [v_{i1}, x, x, x, x, v_{ij}, x, x, x, v_{in}]^T$$

where v_{ij} = designer specified components and x = unspecified components. We define a reordering operator $\{\ \}^{R_i}$ as follows:[7]

$$\{v_i^d\}^{R_i} = \begin{bmatrix} l_i^d \\ d_i \end{bmatrix} \tag{10}$$

where l_i^d is the vector of specified components of v_i^d and d_i is the vector of unspecified components of v_i^d.

We begin the computation of an achievable eigenvector, v_i^a, by defining

$$L_i = (\lambda_i I - A)^{-1} B \tag{11}$$

An achievable eigenvector must reside in the required subspace and hence,

$$v_i^a = L_i z_i \tag{12}$$

We reorder the rows of L_i to conform with the reordered components of v_i^d. Thus, as shown in Ref. 7, we have

$$\{L_i\}^{R_i} = \begin{bmatrix} \tilde{L}_i \\ D_i \end{bmatrix} \tag{13}$$

To find the value of z_i corresponding to the projection of l_i^d onto the "achievability subspace," we choose z_i which minimizes

$$J = \|l_i^d - l_i^a\|^2 = \|l_i^d - \tilde{L}_i z_i\|^2 \tag{14}$$

Upon setting the first derivative of J with respect to z_i equal to zero, we obtain

$$z_i = (\tilde{L}_i^T \tilde{L}_i)^{-1} \tilde{L}_i^T l_i^d \tag{15}$$

and

$$v_i^a = L_i (\tilde{L}_i^T \tilde{L}_i)^{-1} \tilde{L}_i^T l_i^d \tag{16}$$

As shown in Ref. 4, the feedback gain matrix is computed by using a transformation to obtain a system equivalent to Eqs. (1) and (2). This transformed system is described by $(\tilde{A}, \tilde{B}, \tilde{C})$ where

$$\tilde{A} = T^{-1} A T \tag{17}$$

$$\tilde{B} = T^{-1} B = \begin{bmatrix} I_m \\ 0 \end{bmatrix} \tag{18}$$

$$\tilde{C} = CT \tag{19}$$

Under this transformation,

$$x = T\tilde{x} \tag{20}$$

$$\lambda_i = \tilde{\lambda}_i \tag{21}$$

$$v_i^a = T\tilde{v}_i^a \tag{22}$$

We partition $\tilde{v}_i^a$ and $\tilde{A}$ conformally with $\tilde{B}$ to obtain

$$\tilde{v}_i^a = \begin{bmatrix} \tilde{s}_i \\ \tilde{w}_i \end{bmatrix} \tag{23}$$

$$\tilde{A} = \begin{bmatrix} \tilde{A}_1 \\ \tilde{A}_2 \end{bmatrix} \tag{24}$$

Then, as shown in Ref. 4, the feedback gain matrix is computed by using

$$F = (\tilde{S} - \tilde{A}_1 \tilde{V})(\tilde{C}\tilde{V})^{-1} \tag{25}$$

where

$$\tilde{S} = [\lambda_1 \tilde{s}_1, \lambda_2 \tilde{s}_2, \ldots, \lambda_r \tilde{s}_r] \tag{26}$$

and

$$\tilde{V} = [\tilde{v}_1^a, \tilde{v}_2^a, \ldots, \tilde{v}_r^a] \tag{27}$$

In general the number of control surfaces m will be less than the number of sensors r. Thus, we may summarize with the following:

1) The matrix F in Eq. (25) will assign r eigenvalues exactly. It will also assign m components exactly in each of the corresponding r eigenvectors.

2) If more than m components are specified for a particular eigenvector, then an achievable eigenvector is computed by projecting the desired eigenvector onto the allowable subspace.

3) If control over a larger number of eigenvalues is required, then additional independent sensors must be added.

4) If improved eigenvector assignability is required, then additional independent control surfaces must be added.

Feedforward Gain Methodology

The feedforward gain methodology is a special case of Broussard's command generator tracker.[5] Suppose that an explicit model representing the desired behavior of an aircraft is described by

$$\dot{x}_m = A_m x_m + B_m u_m \tag{28}$$

$$y_m = C_m x_m + D_m u_m \tag{29}$$

Further, define the controlled (or tracked) variables of the aircraft by

$$y_t = Hx \tag{30}$$

The objective is to determine a flight control law such that the controlled aircraft variables closely approximate the outputs of the explicit model.

We shall now briefly summarize the derivation of the command generator tracker.[5] Suppose that $y_t = y_m$ at some $t = t_o$. Then, let u^* be the input to the aircraft which guarantees that $y_t = y_m$ for all $t > t_o$. We also let x^* be the corresponding aircraft state. The vectors x^* and u^* are called the ideal aircraft state and the ideal aircraft input, respectively. These quantities satisfy the same dynamics as the aircraft such that

$$\dot{x}^* = Ax^* + Bu^* \tag{31a}$$

$$y^* = Cx^* \tag{31b}$$

$$y_t^* = Hx^* \tag{31c}$$

The controlled variable of the ideal aircraft is equal to the model output. Thus,

$$y_t^* = y_m \tag{32}$$

Assume that x^* and u^* are given by

$$x^* = S_{11}x_m + S_{12}u_m + H.O.D.(u_m) \tag{33}$$

$$u^* = S_{21}x_m + S_{22}u_m + H.O.D.(u_m) \tag{34}$$

Both x^* and u^* are linear in the explicit model's state and input. The matrices S_{ij} are assumed to be constant and the expression $H.O.D.(\cdot)$ represent higher-order derivatives.

If we restrict our discussion to model step inputs, then u_m is a step function, and consequently $H.O.D.(u_m) = 0$ for all $t > t_o$. For the aircraft control problem, u_m is simply the pilot's command.

Subject to the restriction on u_m, the ideal aircraft state and input become

$$\begin{bmatrix} x^* \\ u^* \end{bmatrix} = \begin{bmatrix} S_{11} & S_{12} \\ S_{21} & S_{22} \end{bmatrix} \begin{bmatrix} x_m \\ u_m \end{bmatrix} \tag{35}$$

Under very mild restrictions,[5] the solution for the ideal aircraft input u^* is given by

$$u^* = S_{21}x_m + S_{22}u_m \tag{36}$$

where

$$S_{11} = \Omega_{11}S_{11}A_m + \Omega_{12}C_m \tag{37}$$

$$S_{21} = \Omega_{21}S_{11}A_m + \Omega_{22}C_m \tag{38}$$

$$S_{22} = \Omega_{21}S_{11}B_m + \Omega_{22}D_m \tag{39}$$

and the Ω_{ij} are given by

$$\Omega = \left[\begin{array}{c|c} \Omega_{11} & \Omega_{12} \\ \hline \Omega_{21} & \Omega_{22} \end{array}\right] = \left[\begin{array}{c|c} A & B \\ \hline H & 0 \end{array}\right]^{-1} \tag{40}$$

For our flight control problem we desire the aircraft's controlled variables to track the pilot's commands. This may be achieved by choosing an identity model. This model's output is equal to its input. That is,

$$y_m = u_m \tag{41}$$

Hence, this model's output is also the pilot's command we want the aircraft's controlled variables y_t to track.

The identity model is described by

$$A_m = 0,\ B_m = 0,\ C_m = 0,\ D_m = I \tag{42}$$

Using Eq. (42), Eqs. (34)-(36) become

$$S_{11} = 0 \tag{43}$$

$$S_{21} = \Omega_{12} \tag{44}$$

$$S_{22} = \Omega_{22} \tag{45}$$

and the ideal aircraft state and input become

$$x^* = \Omega_{12}u_m \tag{46}$$

$$u^* = \Omega_{22}u_m \tag{47}$$

We note that the ideal aircraft state x^* and the ideal aircraft input u^* depend only on the pilot command u_m and the feedforward gains Ω_{12} and Ω_{22}. Recall that Ω_{12} and Ω_{22} depend only on the aircraft matrices A, B and the tracking matrix H.

To incorporate output feedback into the design, we let

$$\tilde{x} = x - x^* \tag{48}$$

$$\tilde{u} = u - u^* \tag{49}$$

$$\tilde{y} = y - y^* \tag{50}$$

Then

$$\dot{\tilde{x}} = A\tilde{x} + B\tilde{u} \tag{51}$$

$$\tilde{y} = C\tilde{x} \tag{52}$$

The feedback control law for Eqs. (51)-(52) is given by

$$\tilde{u} = F\tilde{y} = F(y - y^*) \tag{53}$$

or

$$u = u^* + \tilde{u} = u^* + F(y - y^*) = u^* + FC(x - x^*) \tag{54}$$

Upon substituting Eq. (46) for x^* and Eq. (47) for u^* into Eq. (53) we obtain

$$u = \underbrace{[\Omega_{22} - FC\Omega_{12}]u_m}_{\text{Feedforward}} + \underbrace{Fy}_{\text{Feedback}} \tag{55}$$

Observe that the feedforward gains depend upon the feedback gains which were previously computed by using eigenstructure assignment.

Example

The model of AFTI/F-16 will be described by the short period approximation equations augmented by control actuator dynamics (elevator and flaperons). The flight condition corresponds to an altitude $h = 3000$ feet and Mach number $M = 0.6$. The equations of motion are described by Eqs. (1)-(2) where

$$x = \begin{bmatrix} \theta \\ q \\ \alpha \\ \delta_e \\ \delta_f \end{bmatrix} \begin{array}{l} \text{– pitch attitude} \\ \text{– pitch rate} \\ \text{– angle of attack} \\ \text{– elevator deflection} \\ \text{– flaperon deflection} \end{array}$$

$$u = \begin{bmatrix} \delta_{ec} \\ \delta_{fc} \end{bmatrix} \begin{array}{l} \text{– elevator deflection command} \\ \text{– flaperon deflection command} \end{array}$$

$$A=\begin{bmatrix}0 & 1 & 0 & 0 & 0\\ 0 & -0.86939 & 43.223 & -17.251 & -1.5766\\ 0 & 0.99335 & -1.3411 & -0.16897 & -0.25183\\ 0 & 0 & 0 & -20 & 0\\ 0 & 0 & 0 & 0 & -20\end{bmatrix}$$

$$B=\begin{bmatrix}0 & 0\\ 0 & 0\\ 0 & 0\\ 20 & 0\\ 0 & 20\end{bmatrix}$$

The eigenvalues of the open loop system are given by

$$\left.\begin{matrix}\lambda_1=-7.662\\ \lambda_2=5.452\end{matrix}\right\}\ \text{unstable short period mode}$$

$\lambda_3=0.0$ pitch attitude mode

$\lambda_4=-20$ elevator actuator mode

$\lambda_5=-20$ flaperon actuator mode

The maximum control surface deflections are given by

$$(\delta_e)_{\max}=\pm 25 \text{ deg}$$

$$(\delta_f)_{\max}=\pm 20 \text{ deg}$$

and the maximum control surface deflection rates (which are taken to be 70% of the no load values) are given by

$$(\dot{\delta}_e)_{\max}=42 \text{ deg/s}$$

$$(\dot{\delta}_f)_{\max}=56 \text{ deg/s}$$

The normal acceleration at the pilot's station and at the center of gravity are given by

$$n_{zp}=[-0.268,47.76,-4.56,4.45]\begin{bmatrix}q\\ \alpha\\ \delta_e\\ \delta_f\end{bmatrix} \tag{56}$$

and

$$n_{zcg}=[0.137,27.61,3.48,5.19]\begin{bmatrix}q\\ \alpha\\ \delta_e\\ \delta_f\end{bmatrix} \tag{57}$$

where n_{zcg} and n_{zp} are in g's and q, α, δ_e, and δ_f are in radians or rad/s.

The objective in pitch pointing control is to command the pitch attitude θ while maintaining zero perturbation in the flight path angle. The measurements are chosen to be pitch rate, normal acceleration, altitude rate, and control surface deflections. The altitude rate is obtained from the air data computer and it is used to obtain the flight path angle via the relationship

$$\gamma \approx \dot{h}/\text{TAS} \tag{58}$$

where TAS is true airspeed. The surface deflections are measured by using linear variable differential transformers (LVDT).

We include γ as a state because this is the variable whose perturbation we require to remain zero. Thus, we replace θ by $\gamma+\alpha$ in the state equations and obtain an equation for γ. The resulting state space model is given by Eqs. (1)-(2) with

$$x=[\gamma,q,\alpha,\delta_e,\delta_f]^T \tag{59}$$

$$u=[\delta_{ec},\delta_{fc}]^T \tag{60}$$

Table 1 Eigenvector summary

Design No. 1	0	0	1	X	X	γ	0	0	0.667	−0.0057	−0.0137
	1	X	0	X	X	q	1	−9.5	−0.333	1.07	0.0601
	X	1	0	X	X	α	−0.9286	1	−0.333	−0.0508	0.0106
	X	X	X	1	X	δ_e	−5.13	0.1286	−0.808	1	0
	X	X	X	X	1	δ_f	8.36	−5.16	−0.321	0	1
Design No. 2	0	0	1	X	X	γ	0	0	1	−0.0057	−0.0137
	1	X	0	X	X	q	1	−9.5	0	1.07	0.0601
	X	1	X	X	X	α	−0.9286	1	−1	−0.0508	0.0106
	X	X	X	1	X	δ_e	−5.13	0.1286	−2.80	1	0
	X	X	X	X	1	δ_f	8.36	−5.16	3.23	0	1

Table 2 Control law summary

	Desired eigenvalues	Feedforward gains	Feedback gains				
			q	n_z	γ	δ_e	δ_f
Design No. 1	$\lambda^d_{1,2}=-5.6\pm j4.2$, $\lambda^d_3=-1.0$, $\lambda^d_4=-19$, $\lambda^d_5=-19.5$	$\begin{bmatrix}-2.89 & 0.778\\ 1.98 & 3.34\end{bmatrix}$	0.930 / −0.954	0.147 / −0.209	2.11 / −5.32	0.136 / −0.525	−0.755 / 1.04
			q	n_z	γ	δ_e	δ_f
Design No. 2	$\lambda^d_{1,2}=-5.6\pm j4.2$, $\lambda^d_3=-1.0$, $\lambda^d_4=-19$, $\lambda^d_5=-19.5$	$\begin{bmatrix}-2.88 & -0.367\\ 2.02 & 4.08\end{bmatrix}$	0.931 / −0.954	0.149 / −0.210	3.25 / −6.10	0.153 / −0.537	−0.747 / 1.04

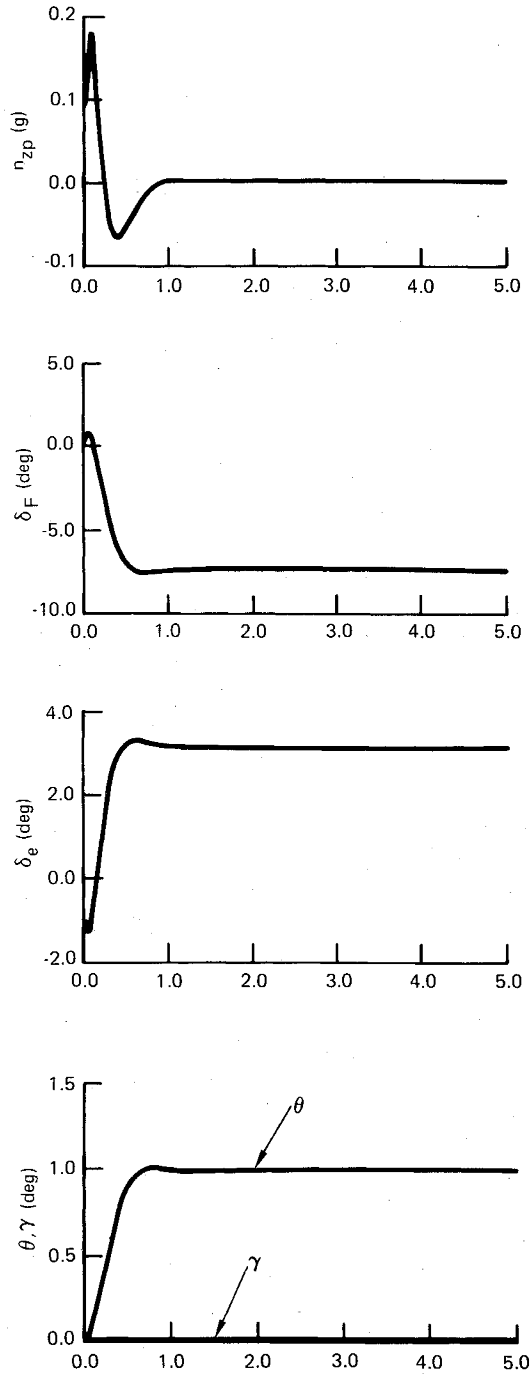

Fig. 1 Pitch pointing design no. 1.

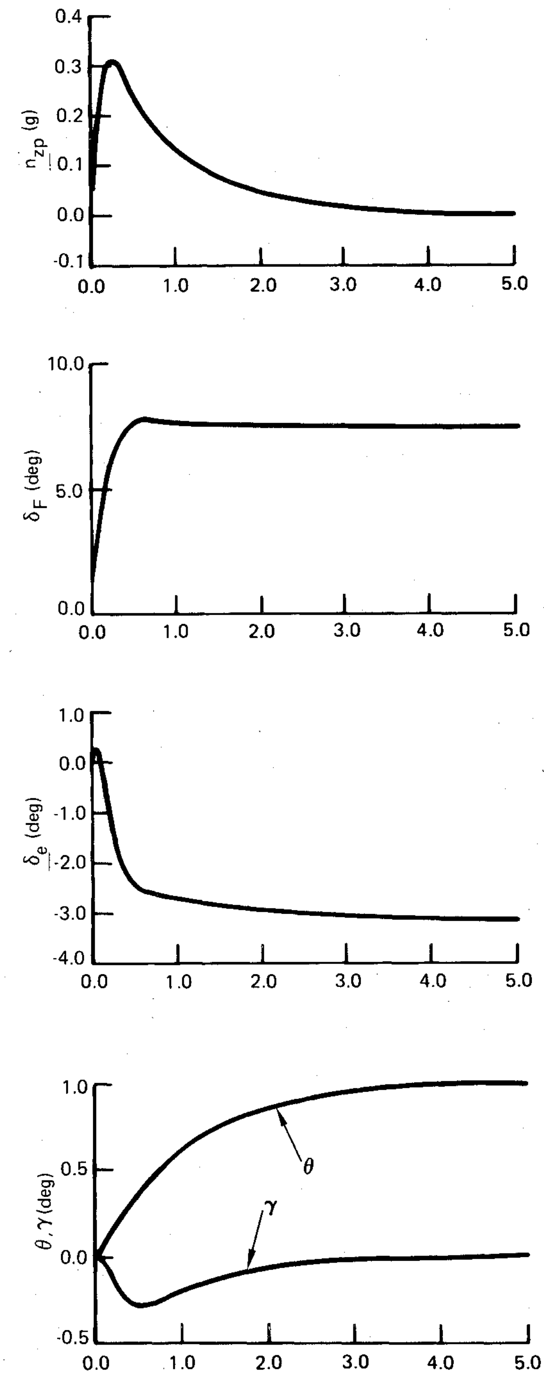

Fig. 2 Vertical translation design no. 1.

and

$$y = \left[q, n_{zp}, \gamma, \delta_e, \delta_f \right]^T \tag{61}$$

Our first step in the design is to compute the feedback matrix F. The desired short period frequency and damping are chosen to be $\zeta = 0.8$ and $\omega_n = 7$ rad/s. These values were chosen to meet MIL-F-8785C specifications for category A, level 1 flight. Category A includes nonterminal flight phases that require rapid maneuvering, precision tracking, or precise flight path control. Level 1 flying qualities are those which are clearly adequate for the mission objectives.

We can arbitrarily place all five eigenvalues because we have five measurements. We can also arbitrarily assign two entries in each eigenvector because we have two inputs. Alternatively, we can specify more than two entries in a particular eigenvector and then the algorithm will compute a corresponding achievable eigenvector by taking the projection of the desired eigenvector onto the allowable subspace.

We choose the desired eigenvectors to decouple the short period and flight path modes. Such a choice should prevent an attitude command from causing significant flight path change. That is, the performance specification is to minimize the coupling between pitch rate and flight path angle. We should also consider, eigenvalue sensitivity in our choice of the desired eigenvectors. Gilbert has shown in Ref. 8 that the sensitivity of the closed-loop eigenvalues to perturbations in the stability derivatives is a minimum when the closed loop eigenvectors are mutually orthogonal. Thus, choosing an entire row of the desired modal matrix as zero would be unwise with regard to sensitivity considerations. The desired eigenvectors and achievable eigenvectors are shown in Table 1 as

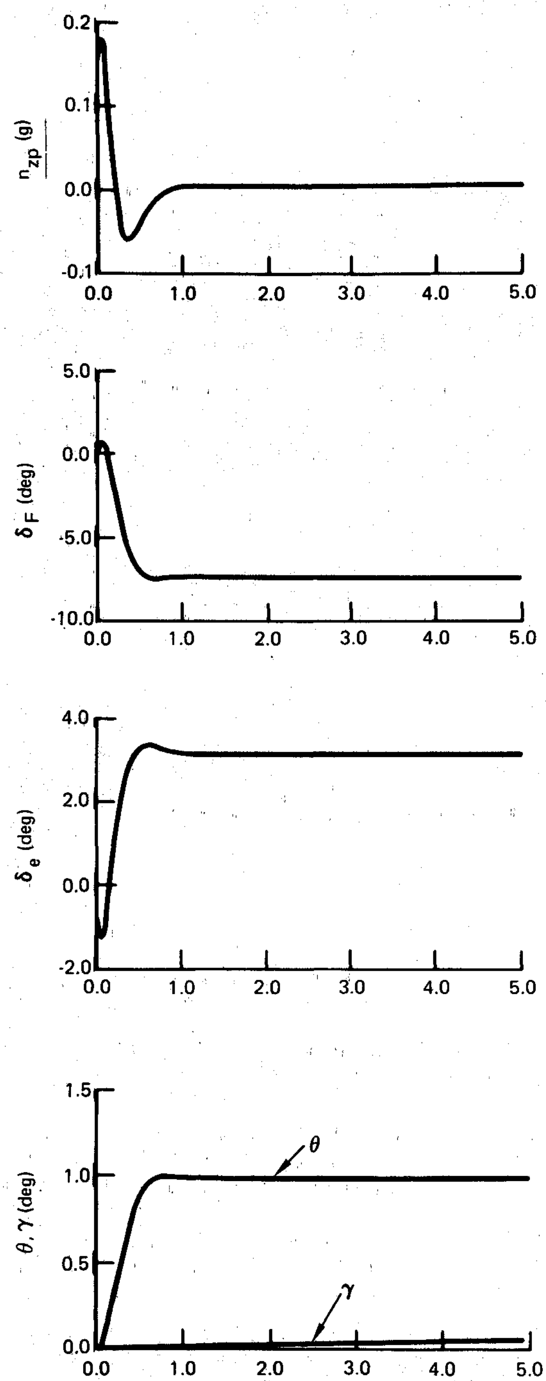

Fig. 3 Pitch pointing design no. 2.

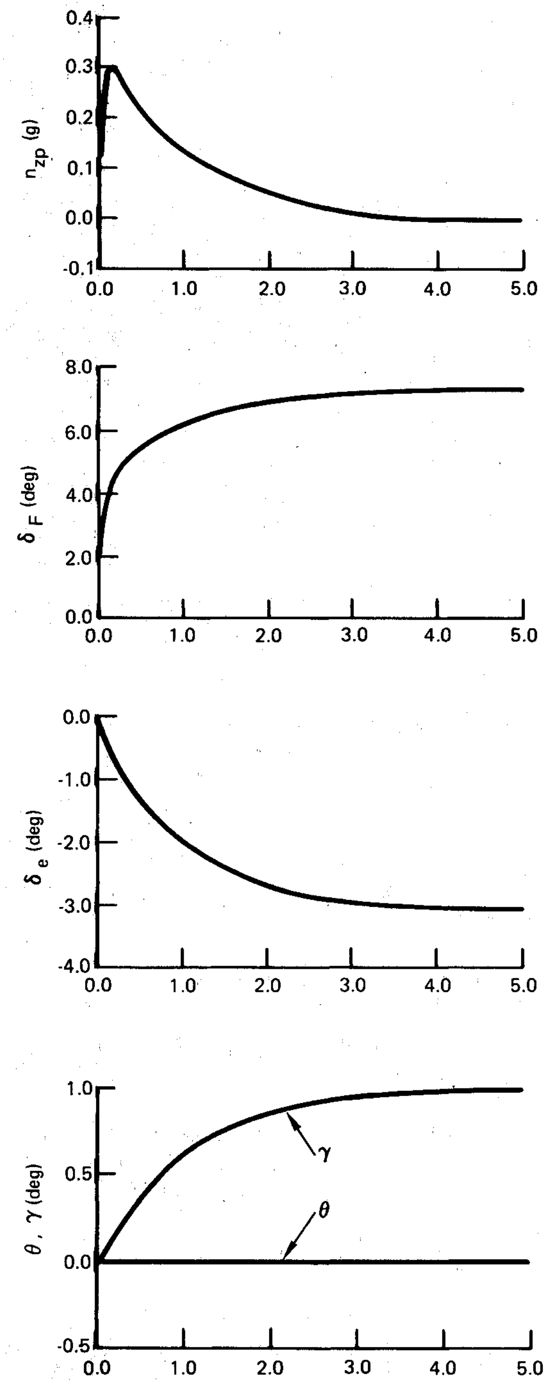

Fig. 4 Vertical translation design no. 2.

design no. 1. We observe that the mode decoupling was not achieved exactly as specified, because we specified greater than m entries in the eigenvector corresponding to the flight path mode.

We now compute the feedforward gains by using Eqs. (40) and (55). For the pitch pointing problem

$$y_t = Hx = [\theta, \gamma]^T \tag{62}$$

$$u_m = [\theta_c, \gamma_c]^T \tag{63}$$

where θ_c = pilot's pitch attitude command and γ_c = pilot's flight path angle command.

$$H = \begin{bmatrix} 1 & 0 & 1 & 0 & 0 & 0 \\ 1 & 0 & 0 & 0 & 0 & 0 \end{bmatrix} \tag{64}$$

The feedforward gain matrix consists of four gains which couple the commands θ_c and γ_c to the actuator inputs. The control law is described by

$$u = \begin{bmatrix} g_{11} & g_{12} \\ g_{21} & g_{22} \end{bmatrix} \begin{bmatrix} \theta_c \\ \gamma_c \end{bmatrix} + Fy \tag{65}$$

When $\gamma_c = 0$ we can command pitch attitude without a change in flight path angle (pitch pointing). Alternatively, when $\theta_c = 0$ we can command flight path angle without a change in pitch attitude (vertical translation). Of course, we could also command both θ and γ if this were required.

The feedback gains and feedforward gains are summarized in Table 2 as design no. 1. Recall that the desired eigenvalues should all be achieved exactly.

The pitch pointing and vertical translation responses are shown in Figs. 1 and 2, respectively. The pitch pointing response is excellent, but the vertical translation response exhibits a moderate attitude transient.

For design no. 2, we emphasize the desired decoupling of pitch rate and flight path angle in our choice of the desired eigenvectors. This is done at the expense of the mode decoupling, which may cause the achievable eigenvectors to be less orthogonal. The desired eigenvectors and achievable eigenvectors are shown in Table 1, from which we observe that we have achieved an exact decoupling between pitch rate and flight path angle. The feedforward gains, and feedback gains are shown in Table 2.

The pitch pointing and vertical translation responses are shown in Figs. 3 and 4, respectively. Both designs exhibit excellent responses with reasonable control surface deflections.

Conclusion

This paper has presented a design methodology for pitch pointing flight control systems which utilizes eigenstructure assignment with command generator tracking. Eigenvalue-eigenvector assignment type algorithms are suited precisely for this problem due to the relationship between eigenvectors and decoupling. The command generator tracker is used to ensure zero steady-state error to step commands. An interesting feature of the methodology is the fact that one set of gains may be used for both pitch pointing and vertical translation by activating the appropriate command. Furthermore, although not shown in the paper, the aircraft exhibits a conventional response with acceptable flying qualities in the event of a flaperon failure.

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