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Adaptive Control Applications to Aerospace Vehicles

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Introduction

THE word "adaptive" is used to describe the capability of human beings, animals, and plant life to adjust their characteristics in accordance with a changing environment. The adaptive control system thus is a system which will adjust its own characteristics in order to provide a desired control system performance in a changing dynamic environment. There has been considerable literature published on the subject of adaptive automatic control systems which has been concerned with aerospace vehicle control, process control, and machine control, as well as control in other technical areas.

Research has included very general investigations which could apply to automatic controls of all types, as well as specific investigations concerned with the development of systems designed to adapt to particular environmental changes and for a particular application.

The present survey is of the latter type, and is concerned with adaptive control systems which adapt for changing dynamics of the aerospace vehicle, when the vehicle is considered as a rigid body. Vehicle aerodynamic characteristics are changing due to changes in vehicle configuration, flight speed, Mach number, and altitude. These changes result in changes in the dynamics of the aerospace vehicle. The research approach used in the current survey was to perform search of patents granted in the United States on adaptive control systems. This search located 45 patents which were granted between the years of 1925 and 1967. Of these 45 patients, only 26 were determined to be within the scope of this paper. In addition to the patented systems, four systems of interest were located from a survey of current papers on adaptive flight control systems.

The objective of the survey was to investigate the requirements for adaptive control systems, and to determine the means which have been developed to implement these requirements. The techniques discussed are generally applicable for use on all three axes of the aerospace vehicle. However, the examples presented are with reference to the pitch axis.

Control System Performance Criteria

General Comments

The block diagram in Fig. 1 illustrates a typical system for one axis of an aerospace vehicle. The control system is simplified in that cross coupling between the control channels is omitted. However, this simplification does not invade

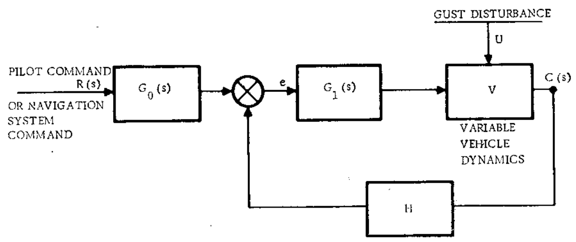

Fig. 1 Aerospace vehicle control system block diagram—illustrating the adaptive problem.

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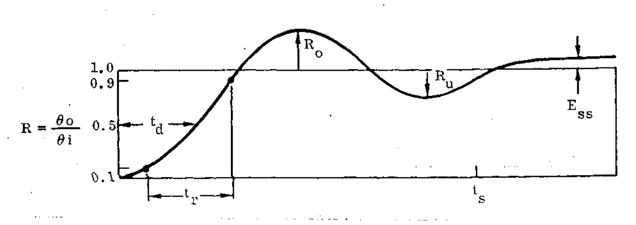


Fig. 2 Transient response characteristics—used in performance criteria.

the control principles and the fundamental control system performance objectives which are subsequently discussed.

The flight control system has two inputs. One input is a command signal from the pilot or automatic navigation system. The second input is a disturbance input caused by atmospheric wind gusts.

For all aerospace vehicles, the control system must satisfy established performance criteria with respect to the following factors: 1) control system stability, 2) transient response to control command inputs, 3) transient response to wind gust disturbance inputs, and 4) control system noise. Let us briefly consider these criteria.

Stability

In order to satisfy performance objectives the closed-loop system must be dynamically stable. Low-frequency instabilities (for example, dynamic modes with periods in excess of 30 sec) may be tolerated for some aerospace vehicles providing these instabilities can be readily controlled by the pilot or navigation system. Generally, it may be stated that it is essential to stabilize all dynamic modes which result from the control system, the vehicle rigid body dynamics, and the vehicle structural dynamics.

Response to Input Commands

Requirements on control system response to input commands are frequently presented as transient response requirements to step commands. This type of presentation of response requirements is convenient from the viewpoint of control system designer.

The control system transient response characteristics to a step command are illustrated in Fig. 2. The characteristics illustrated are t_d (delay time), time required for the system to attain a level of 50% of the command signal; t_r (rise time), time required for the system output to increase from 10 to 90% of the command signal; R_o and R_u, overshoot and undershoot of oscillatory response as a percentage of the command signal; t_s (settling time), time beyond which the input will remain within a specified percentage of the command signal; e_{ss} (steady-state error), the system error at a present time after the step command was initiated, generally, a time in excess of the time required for the transient caused by the primary mode to decay.

Formal manned aircraft dynamic requirements are frequently based on laboratory and flight simulator tests, which have resulted in boundaries of acceptability for aircraft short-period dynamics. Typical boundaries for the aircraft pitch axis are shown in Fig. 3. These boundaries define limitations to the short-period mode with respect to damping ratio and frequency. The boundaries presented are based on tests performed by the Cornell Aeronautical Laboratory, North American Rockwell- Columbus Division, and USAF MIL-8785.[1]

The need for transient response design objectives for manned aircraft control systems has been recognized by control designers and an interpretation of the short-period damping and frequency boundaries given as a normal acceleration transient response boundary was derived.[2] A typical response boundary is shown in Fig. 4. This boundary is considered acceptable for all but the low-speed flight regime, where vehicle attitude response is considered to be more significant.

In addition to this boundary, it is necessary to specify the minimum acceptable damping ratio on the dynamic modes of higher frequency than the vehicle short-period modes. This is apparent since one could remain within the boundary and have unacceptable oscillatory transient response for these modes. This design objective will subsequently be considered with respect to airframe dynamic variations and the ability to satisfy the requirements without recourse to adaptive control.

Response to Wind Gusts

The magnitude and nature of the dynamic response of the aerospace vehicle and control system to wind gust disturbances may influence: 1) the vehicle structural design; 2) the crew and passenger comfort; and 3) the ability of the vehicle to be controlled to a specified trajectory. The flight control system should not adversely affect the vehicle response to the wind gust. In fact, a somewhat improved response should be achieved through increased damping of the vehicle short-period mode.

The dynamic response of the aerospace vehicle to a wind gust disturbance of specified intensity and frequency distribution (power spectrum) may be defined by the rms (root mean square) value, and by the power spectrum of vehicle acceleration. Both the frequency distribution of the response (as indicated by the shape of the power spectrum), as the average intensity (as indicated by the rms g's) are important.[3]

It may be noted that some adaptive control systems may adversely influence vehicle response to wind gusts while attempting to optimize control system response to pilot commands. The total control system performance could be unacceptable even though the response to pilot commands was excellent.

System Noise

Excessive noise generated in the control system could adversely affect system operation and be disconcerning to the

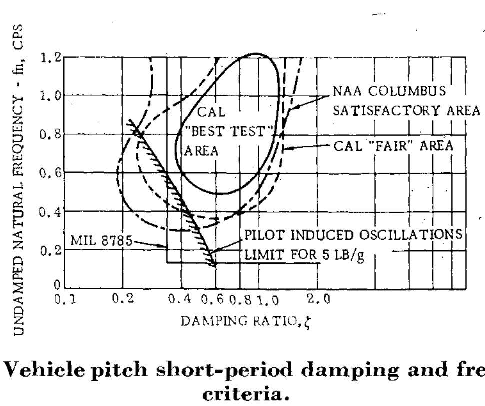


Fig. 3 Vehicle pitch short-period damping and frequency criteria.

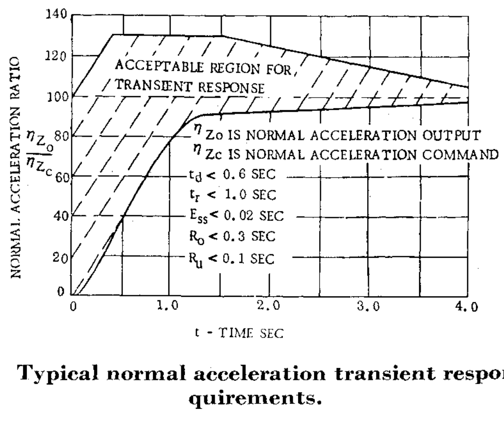


Fig. 4 Typical normal acceleration transient response requirements.

pilot. Control systems employing high gain and/or intentional limit cycles in order to obtain more uniform response to pilot commands could be unsatisfactory because of excessive noise. Consequently, adaptive systems employing residual oscillations or intentional disturbances to facilitate identification of vehicle dynamics must be carefully evaluated with respect to acceptable control system noise and/or vehicle motion. Conventional systems employing excessively high gain (wide bandwidth), even though satisfying stability and transient response requirements, must be evaluated wth respect to the degrading effects of high noise levels.

Variations in Vehicle Dynamics

General

As previously noted, the need for the adaptive control system results from the variation which occurs in the dynamics of the aerospace vehicle. The objective of this portion of the paper is to identify the characteristics of the vehicle dynamics and the extent of the variations which occur. It is believed that clarification of vehicle dynamic variations will aid in placing the adaptive problem in proper perspective. The longitudinal axis is selected for the example, but other control axes could have been used as well. As previously noted, this paper is concerned only with control systems designed to adapt to variations in vehicle rigid body dynamics.

Rigid Body Equations of Motion

The dynamics of the aerospace vehicle are presented in various tests (e.g., Ref. 4 and 5). The longitudinal equation for perturbations from steady-state conditions are given in the stability axis system as

$$\dot{u} + g_\theta \cos\gamma_0 = \sum_{i=0}^{m} X_i + \cos\xi T_u u, \quad \text{drag equation}$$

$$\dot{\alpha} U_0 - qU_0 + g_\theta \sin\gamma_0 = \sum_{i=0}^{m} Z_i + \sin\xi T_u u, \quad \text{lift equation}$$

$$\dot{q} = \sum_{i=0}^{m} M_i + \frac{Z_T m}{I_{yy}} T_u u, \quad \text{moment equation}$$

where the forces and moments (X_i, Z_i, and M_i) are obtained by perturbation with respect to the pertinent aircraft variables (e.g., angle of attack and its derivatives, and flap deflection and its derivatives). For analyses of the adaptive control problem, it is possible to neglect the dynamics of the low-frequency phugoid mode which is introduced by the drag equation.

Transfer Functions

The transfer functions for the vehicle response in pitch attitude rate ($\dot{\theta}$ or q), angle of attack (α), and normal acceleration (a_Z) to control surface command (δ) may be derived from the lift and pitching moment equations to be

$$\dot{\theta}/\delta = K_{\dot{\theta}}\,(T_{\dot{\theta}}s + 1)/[(s^2/\omega_n{}^2) + (2\zeta s/\omega_n) + 1]$$

$$\alpha/\delta = K_\alpha\,(T_\alpha s + 1)/[(s^2/\omega_n{}^2) + (2\zeta s/\omega_n) + 1]$$

$$\alpha_Z = -K_\delta\,(T_{\alpha 2}s + 1)\,(-T_{\alpha 3}s + 1) \Big/ [(s^2/\omega_n{}^2) + (2\zeta s/\omega_n) + 1]$$

where

$$\omega_n = (M_\delta Z_w - U_0 M_w)^{1/2}$$

$$\zeta = -(U_0 M_{\dot{w}} + Z_w + M_\delta)/2\omega_n$$

$$K_{\dot{\theta}} = (Z_\delta M_w - M_\delta Z_w)/(M_\delta Z_w - U_0 M_w)$$

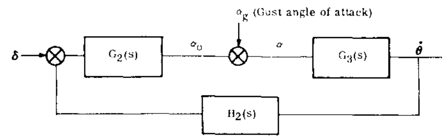


Fig. 5 Block diagram of the airframe.

$$T_\theta = (M_\delta + Z_\delta M_{\dot{w}})/(Z_\delta M_{\dot{w}} - M_\delta Z_w)$$

$$K_\alpha = (1/U_0)\,(U_0 M_\delta - Z_\delta M_\delta)/(M_\delta Z_w - U_0 M_w)$$

$$T_\alpha = Z_\delta/(U_0 M_\delta - Z_\delta M_\delta)$$

$$T_{\alpha 2} = 2Z_\delta/[(M_\delta Z_q - Z_\delta M_\delta) + 2(Z_\delta M_\delta U_0 Z_w)^{1/2}]$$

$$T_{\alpha 3} = 2Z_\delta/[(M_\delta Z_q - Z_\delta M_\delta) - 2(Z_\delta M_\delta U_0 Z_w)^{1/2}]$$

Airframe Transfer Functions with Command and Gust Inputs

To assist in subsequent discussion, it is convenient to construct a block diagram (Fig. 5) of the airframe illustrating the airframe as a servo device and wind gust as inputs and aircraft pitch rate as the output.

It has previously been noted that

$$\dot{\theta}/\delta = K_{\dot{\theta}}\,(T_{\dot{\theta}}s + 1)/[(s^2/\omega_n{}^2) + (2\zeta s/\omega_n) + 1] = M_\delta\,(s + \omega_T)/(s + 2\zeta\omega_n s + \omega_n{}^2)$$

$$\alpha_0/\delta = K_\alpha\,(T_\alpha s + 1)/[(s^2/\omega_n{}^2) + (2\zeta s/\omega_n) + 1]$$

Then

$$G_3(s) = (\dot{\theta}/\delta)\cdot\delta/\alpha_0 = K_{\dot{\theta}}(T_{\dot{\theta}}s + 1)/K_\alpha(T_\alpha s + 1)$$

It may be shown that if

$$G_2(s) = 1/G_3(s) = K_\alpha(T_\alpha s + 1)/K_{\dot{\theta}}(T_{\dot{\theta}}s + 1)$$

and

$$1 + H_2(s) = [(s^2/\omega_n{}^2) + (2\zeta s/\omega_n) + 1]/K_{\dot{\theta}}(T_{\dot{\theta}}s + 1)$$

the block diagram provides a solution which corresponds to the transfer functions for θ/δ and α/δ.

From the previous, it follows that $\dot{\theta}/\alpha_g = G_3/(1 + G_2G_3H_2)$ which may be evaluated.

This form for the airframe transfer function is useful in subsequent discussion of adaptive control system requirements.

Variation with Flight Conditions of Airframe Gain

A Bodé plot of the airframe pitch rate gain (M_δ) can readily be made by using the convenient servo technique of identifying the corner frequency, and using a gain vs frequency slope of 6-db octave increase for a simple lead and 12 db/octave decrease for a second-order lag. The airframe lead (T_θ) generally occurs at a frequency which is lower than the undamped natural frequency (ω_n) of the vehicle short-period mode. Thus, the gain vs frequency is shown in Fig. 6 for a high-

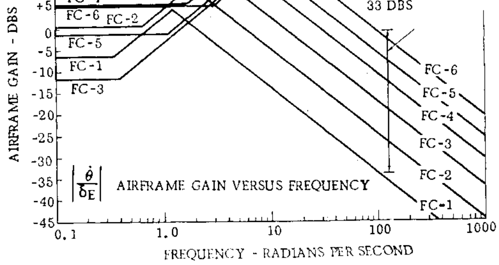


Fig. 6 Bodé diagram illustrating vehicle gain variation.

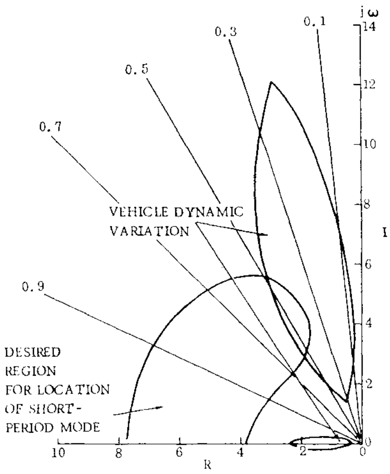


Fig. 7 Complex plane plot of vehicle dynamic variation and desired root location.

performance aircraft at six different flight conditions which were selected to include the extremes of the flight envelope. These conditions represent the extremes in airframe gain for which the control system must compensate.

On the basis of similar curves constructed for four high performance fighters (Mach number range from about 0.15 to 2.0) and one re-entry vehicle, the variation shown is considered fairly typical for the high-frequency gain variation which can be expected.

The low-frequency gain variation over the flight envelope was as little as 20 db in one case and was 35 db for the vehicle with the greatest variation.

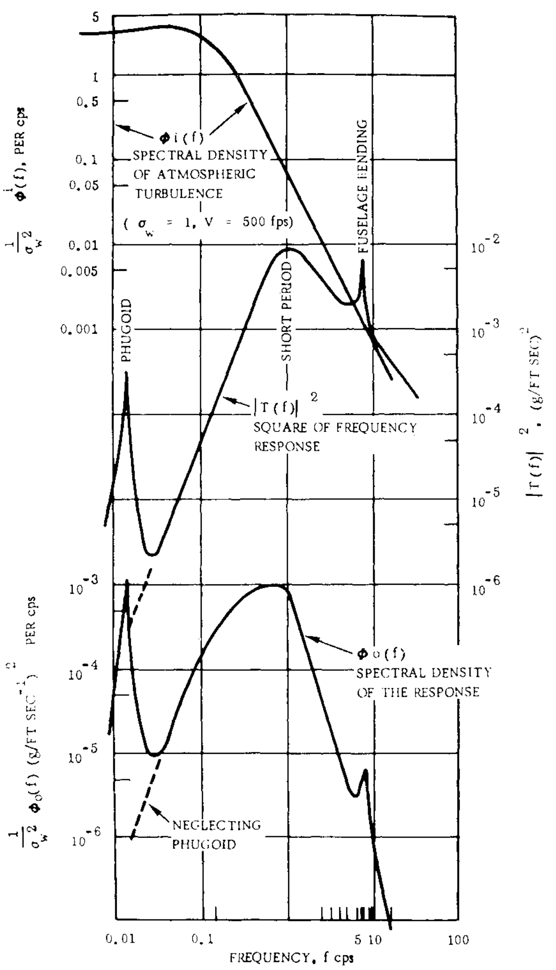


Fig. 8 Typical aircraft response in acceleration due to random atmospheric turbulence.

The high-frequency gain which will subsequently be demonstrated as being more significant to the control problem had variations of 25, 32, 33, 35, and 38.5 db for the five vehicles investigated. The data on the vehicle with the 25 db variation were relatively limited, and it is believed that the variation would have been greater if more complete data were available.

The low-speed landing approach condition is designated as flight condition 1 in Fig. 6. If the landing approach conditions were omitted, the variation in high-frequency gain would be about 5–8 db less for all the aircraft investigated.

Variation with Flight Conditions—Pitch Short-Period Roots

It has previously been noted that a desirable range of damping and frequency of the pitch short-period mode has been established (e.g., Fig. 3) based on pilot evaluation. Figure 7 illustrates the range of variation of frequency and damping that is typical for a high-performance fighter aircraft. The roots of the short-period mode are plotted in the s plane. The vertical axis represents the damped frequency, and a constant damping ratio is represented by a radial line from the origin. As shown, the unstabilized airframe varies in damping ratio from about 0.05 to 0.5, and the frequency from about 1.5 to 12 rad/sec. The lead time constant varies from about 0.4 to 5.0 sec (or corner frequency ω from 2.5 to 0.2 rps). The objective of the control system is to place the controlled short-period mode within the desirable boundary shown (derivated from Fig. 4), and to restrict the variation of the lead time constant.

Response to Wind Gust Spectrum

The requirement for providing a control system which does not degrade the aircraft response to gusts has already been noted. In the longitudinal plane, the normal acceleration response and pitch rate response are both important. To evaluate properly, the effect of the wind gust on the aircraft, the atmospheric turbulence must be considered in terms of its statistical nature. The wind gust may be represented by a power spectrum $\Phi_i(\omega)$ which is frequency dependent. The response of the aircraft may be computed from this input spectrum[3] as $\Phi_0(\omega) = |T(j\omega)|^2\Phi_i(\omega)$ where $|T(j\omega)|^2$ is the modulus squared of the frequency response function. A representative gust spectrum and typical normal acceleration frequency response and spectral density output of the airframe are shown in Fig. 8. The objective of the control system is as a minimum not to degrade the normal acceleration and pitch rate response of the uncontrolled vehicle.

Adaptive Control System Requirements

Basic Control Loop Considerations

A block diagram which illustrates the basic control loop is shown in Fig. 9. As illustrated, the pilot or navigation system command signal may be operated on by a prefilter designated as $G_0(S)$. The forward loop contains loop compensation and actuation servos designated as $G_1(S)$. The

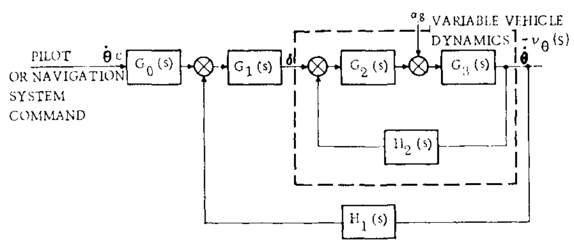


Fig. 9 Control system block diagram showing transfer functions with pilot and gust as inputs.

variable airframe dynamics are indicated as $V_\theta(S)$, and the feedback compensation as $H_1(S)$. The airframe inputs are control surface deflection (δ) and the wind gust disturbances (α_g). The design objective is to satisfy the performance criteria with a control system of minimum complexity.

A control system which has minor variation in the control system performance with major variations in vehicle dynamics is referred to as a control system of low sensitivity.[6] A fundamental automatic control approach to the design of a system with low sensitivity to vehicle dynamic changes is to extend the control loop bandwidth through an increase in control loop gain. The decrease in control loop sensitivity through increase in gain and bandwidth can be illustrated readily by synthesis of control loop transfer functions. From Fig. 9, $\dot\theta/\dot\theta_c = G_0G_1/(1 + H_2 + G_1H_1)$, since $G_2G_3 = 1$, $\dot\theta/\alpha_g = G_3/(1 + H_2 + G_1H_1)$. If $G_1H_1 \gg 1 + H_2$, $\dot\theta/\dot\theta_c \cong G_0/H_1$, and $\dot\theta/\alpha_g \cong G_3/G_1H_1$.

Hence, for high loop gain, the control system response characteristics become independent of variation in vehicle characteristics (except for the effect of variations in G_3 on gust response).

However, the control loop gain and bandwidth are limited both by considerations of 1) control loop noise and 2) dynamic stability of structural modes. The modern aerospace vehicle has structural modes for which the lowest frequency is generally 20–30 rad/sec, and may be lower in some cases. To avoid severe structural coupling, the cutoff frequency of the control system should be such as to attenuate the structural modes. A bandwidth of about 20–30 rad/sec appears to be a maximum in view of these requirements.

Forward vs Inverse Model

Two basic control approaches which have been used to achieve the required transient response to pilot commands are referred to as 1) the prefilter or forward model and 2) the inverse model.

The prefilter, or forward model, uses the transfer function, $G_0(S)$, to model the desired response. The control loop gain is increased so as to obtain substantial cancellation (i.e., low residue) of the vehicle short-period mode. Thus, with unity feedback, $H_1 = 1$, $\dot\theta/\dot\theta_c \cong G_0(S)$. This approach has the undesirable features of relatively high gain and noise, as previously noted, since to obtain good cancellation, the control loop gain must be increased as high as practicable. Even so, the transient response will be modified by the residue to the short-period mode. In addition, for high gain, it has been noted that $\dot\theta/\alpha_g = G_3/G_1$.

Hence, the vehicle basic response to wind gust i.e., (G_3/H_2) is modified by H_2/G_1. Hence, if G_1 is selected with the objective of cancellation of the vehicle short-period mode, it could adversely affect the vehicle response to gusts.

The root locus plot (Fig. 10) illustrates a typical dynamic situation using the forward model. The vehicle short-period mode is increased in frequency through feedback so as to derive a response sufficiently fast as not to affect the optimum pilot response which is set into the prefilter $G_0(S)$. The ability to maintain a uniformly satisfactory response with the vehicle dynamic variations noted requires adequate control of the vehicle short-period mode roots. Nevertheless, even if sufficient control is provided to maintain good pilot response, the conditions of control system noise and wind gust response must be satisfied. The compensation, $G_1 \cong K_1\,(S + 8)/(S + 1)$, will significantly influence the characteristic equation for gust response.

The inverse model technique uses the feedback compensation (H_1) to control and to desensitize the response to variations in vehicle short-period dynamics. If a unity prefilter (i.e., $G_0 = 1$) is used as gain G_1H_1 increases, the response characteristics approach the following functions: $\dot\theta/\dot\theta_c \cong 1/H_1$ and $\dot\theta/\alpha_g \cong G_3/G_1H_1$.

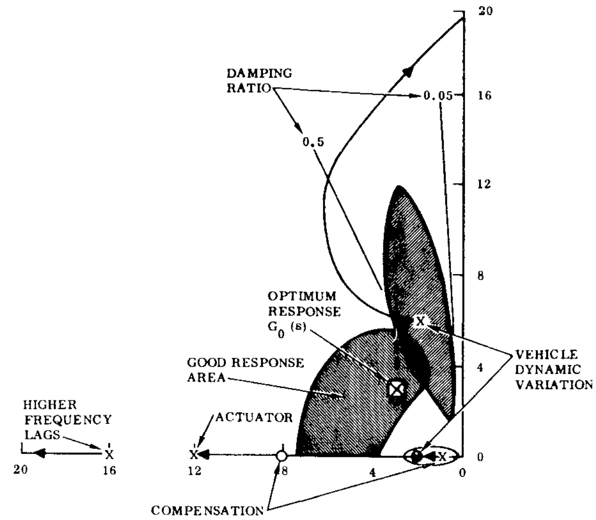


Fig. 10 Root locus of control system with forward model.

The root locus plot shown in Fig. 11 illustrates a typical dynamic situation using the inverse model. The compensation is of the form

$$H_1 = \frac{1}{[(s^2/\omega_n^2) + (2\zeta s/\omega_n) + 1]} \cong \frac{1}{[(s^2/20) + (1.8/4.5)s + 1]}$$

$$G_1 = K_1\,\frac{1}{(s + \omega_1)} \cdot \frac{1}{(s + \omega_2)} = \frac{1}{K_1\,(s + 20)\,(s + 0.2)}$$

The loop gain need only be high enough to draw the short-period poles into the "good response area." Since the compensation G_1 does not contain lead, the characteristic differential equation which influences the vehicle response to gust disturbances is unaltered in the closed loop.

Thus, the inverse model has the advantage over the forward model in that 1) the inverse model introduces no extraneous roots in the short-period frequency regime to affect the vehicle response to pilot commands, 2) the inverse model requires only sufficient loop gain to control the vehicle short-period roots into the desired response area and consequently is not inherently a high gain system, and 3) the characteristic differential equation which determines the vehicle response to wind gust is not altered through use of the inverse model and associated compensation.

The Bodé analysis (e.g., Ref. 7, pp. 2–31 to 2–56) states that the system crossover frequency is the frequency at

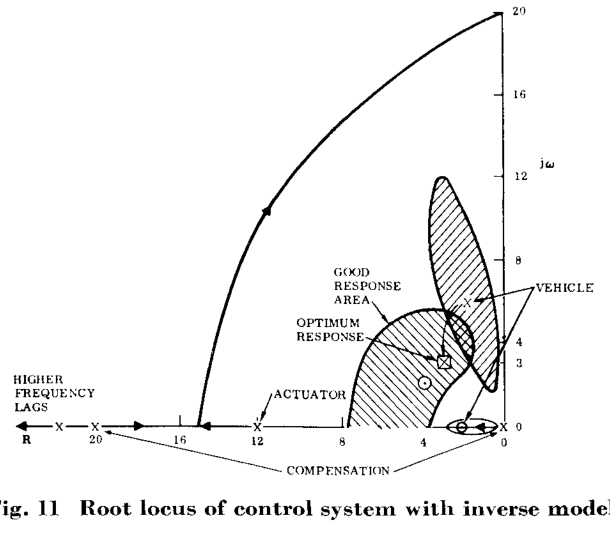


Fig. 11 Root locus of control system with inverse model.

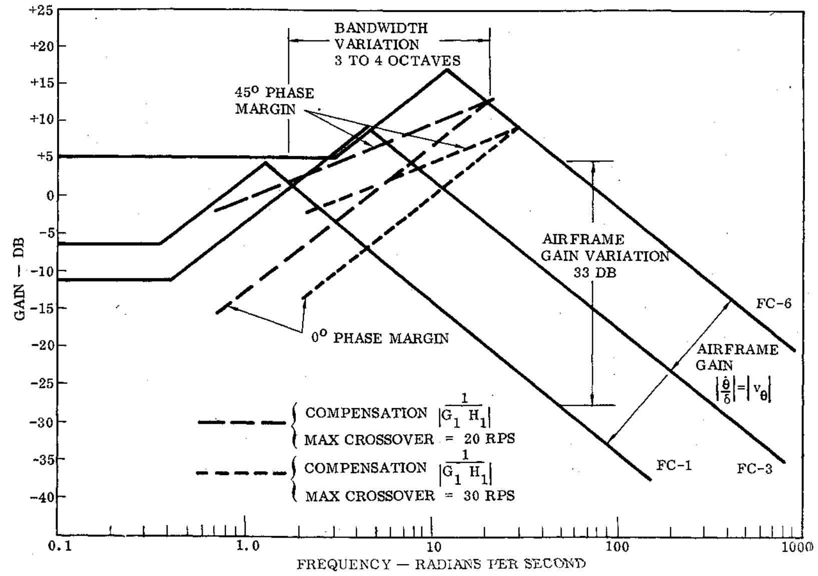


Fig. 12 Airframe gain (V_θ) and inverse of compensation gain $1/(G_1H_1)$ vs frequency.

which the open-loop control system gain is unity. The bandwidth of the control system is defined as the frequency band for which the open-loop gain is greater than unity. Thus, with reference to Fig. 9, the requirements on open-loop gain at the crossover frequency are $G_1V_\theta H_1 = 1.0$ or $|V_\theta| = |1/G_1H_1|$. Furthermore, Bodé analysis indicates that, in order to achieve control-loop stability with a specific phase margin, the open-loop transfer functions must attenuate for a frequency range in the regime of crossover at a rate as summarized in columns 1–3 in Table 1.

The Bodé diagram for the airframe gain vs frequency was previously described and is shown in Fig. 6. For the frequency range at which crossover must occur in order to control the vehicle short-period mode, the airframe attenuation rate is -6 db/octave. This vehicle attenuation rate requires a compensation attenuation rate at crossover as noted in column 5 of Table 1 in order to provide 0°, 45°, and 90° of "phase margin." The zero phase margin case is, of course, unacceptable for good control characteristics but is presented to illustrate the limit from stability considerations.

Two extremes and one medium condition for airframe gain $|V_\theta|$ were selected from Fig. 6, and plotted in Fig. 12 vs the inverse of control loop gain, $|1/G_1H_1|$. The frequency at which the gain curves intersect is the crossover frequency.

With the maximum crossover frequency limited to 20 and 30 rad/sec, in order to minimize structural coupling and/or control loop noise, the variation in control loop bandwidth due to variations in airframe gain are as shown in Fig. 12. A bandwidth variation of between 3 and 4 octaves will occur for a fixed parameter control system if a 45° phase margin is assumed. Thus, the bandwidth could vary with flight condition from 1.7 to 20.0 rps or from 2.4 to 30.0 rps, depending upon whether a maximum bandwidth of 20 or 30 rps is assumed. In either case, the low dynamic pressure flight conditions which result in low airframe gain do not provide sufficient bandwidth for proper control of the vehicle short-period modes.

If the vehicle gain at a frequency near gain crossover (e.g., a frequency above the highest short-period frequency but below the lowest structural frequency) is identified and the control loop gain is adjusted so as to maintain the product of control loop and airframe gain constant at that frequency, the crossover frequency will be independent of flight condition (i.e., the control system bandwidth would be independent of flight condition).

This is illustrated in Fig. 13. In the illustration, the compensation is written as $K_xG_1H_1$ where K_x is the variable gain at flight condition x. For convenience in illustration $K_x V_\theta$ and $1/G_1H_1$ are plotted vs frequency. For the case cited, $K_6 = 1.0$ (0 db), gain at flight condition 6, $K_3 = 6.3$ (16.2 db), gain at flight condition 3, and $K_1 = 44.7$ (33 db), gain at flight condition 1. With this adjustment in gain, proper control of vehicle short-period mode, and thus satisfactory response to pilot commands, is achieved at the low dynamic pressure flight conditions.

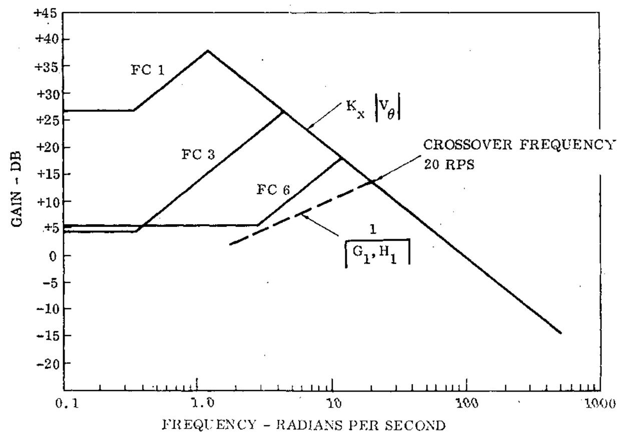


Fig. 13 Product of airframe and variable gain and inverse of compensation vs frequency.

Transient Response Dependency on Bandwidth

The transient response of the control system to pilot commands is critically dependent on control system bandwidth. The approximate effect of bandwidth on time delay is described in Ref. 6. It is shown that if a pitch rate time delay of 0.25 sec is permissible, a servo bandwidth of at least 8 to 10 rps is required.

Table 1 Attenuation rate vs frequency for specified phase margins

1	2	3	4	5
Attenuation rate at crossover		Phase[b] margin, deg	Airframe attenuation rate at crossover, db/octave	Compensation attenuation rate at crossover, db/octave
db/decade[a]	db/octave			
−40	−12	0	−6	−6
−30	−9	45	−6	−3
−20	−6	90	−6	0

[a] db = 20 log$_{10}$ (amplitude ratio).
[b] Additional phase lag at the crossover frequency which will cause the control system to become unstable.

Constant closed-loop, high-frequency gain achieved by adjusting control system gain results in essentially constant damping ratio for the closed-loop resonant mode (as illustrated in Fig. 14). Generally, a damping ratio, $\zeta = 0.3$ for the dynamic mode near gain crossover is considered to be a desirable performance goal.

Aerospace Vehicle Gain Identification

As previously noted, $\dot{\theta}/\delta = M_\delta\,(s + \omega_\iota)/(s^2 + 2\zeta\omega_n s + \omega_n^2)$. At high frequency, in the region of crossover (i.e., $\omega \gg \omega_n$), the vehicle transfer function may be approximated as $\theta/\delta = M_\delta \cdot (1/s)\ (\omega \gg \omega_n)$.

Hence, the high-frequency gain of the airframe is proportional to M_δ. If M_δ is identified, this is sufficient information on vehicle gain variation to adjust the loop gain so as to maintain constant servo bandwidth.

Summary

The principal facts established with respect to control system requirements are 1) Vehicle dynamic changes over the flight envelope for a high performance aircraft will result in a nominal gain variation at control system crossover of about 33 db. This gain variation will cause a control system bandwidth variation of 3 to 4 octaves (i.e., a frequency ratio of between 6 and 8). 2) In order to achieve the desired performance the control system bandwidth must remain within the range of 10 to 20 rps over the flight envelope. 3) Considerations 1 and 2 previous, dictate that a control system gain variation of approximately 16 to 1 is required (this assumes a 45° phase margin and an attenuation rate at crossover of 9 db. 4) The control system performance requirements can be met with a gain variation somewhat less than 16 to 1 if desensitization of the control system is effected by means such as the inverse model.

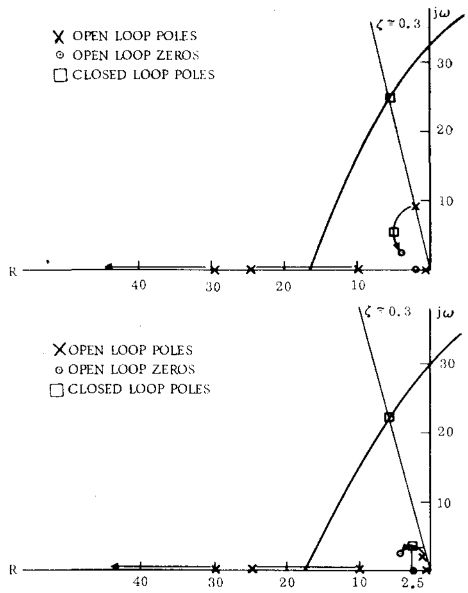


Fig. 14 Root locus plots of adaptive control system at two extreme flight conditions.

Adaptive Techniques

General

There are various definitions of adaptive control which are presented in the literature. In the present paper, adaptive systems are considered as those which are designed to provide uniform system performance over a flight envelope which involves major changes in vehicle dynamics. According to this general definition, linear systems and nonlinear systems that do not require control system parameter changes, but which are designed to be insensitive to variations in vehicle dynamics, could be considered adaptive. The present survey is primarily concerned, however, with techniques which use gain or system parameter adjustment to complement the adaptation.

The research approach used in the survey was organized to perform a patent search of adaptive flight control systems. The patents identified on the search are listed in the patent bibliography. For completeness, some of the patents obtained in the search, but which were considered not applicable to the present survey, are also given. In addition, adaptive systems described in reference reports 14 (or 15) and 10–12 are considered to be systems of interest but for which patents were not granted at the time the search was made.

Classes of system which could have been adaptive, but which were not investigated in the present survey, were those which may be considered either as learning systems or optimal control systems. Examples of these types of systems are given in Refs. 16–19.

The function which the adaptive system may perform in order to accomplish the adaption may be considered to consist of the following: 1) identify the vehicle or system dynamic changes which have taken place, 2) evaluate the effect on system performance and determine the parameter adjustments required, and 3) implement the parameter adjustments.

Of the various adaptive systems which have been patented and/or developed, there is a variety of techniques which vary in detail to effect each of these adaptive functions. There is a basic similarity in the philosophy of these adaptive systems although the detailed implementations may appear superficially to be unrelated.

The basic characteristic which appears to distinguish best the various adaptive techniques is the method used for identification. Hence, for purposes of organization and presentation of the theory of operation of these adaptive systems, they have been organized herein on the basis of the method of identification which is used.

Analysis of the adaptive systems has indicated that all of the systems investigated could be classified into the following categories on the basis of the identification process used: 1) no identification, 2) open-loop gain and/or phase identification, 3) vehicle coefficient identification, and 4) performance identification.

Table 2 contains a list of 28 adaptive control systems which are organized according to the identification process. Table 2 contains the report or patent reference and a summary of the functional characteristics of the systems. A discussion of these adaptive systems by category is given below.

No Identification Process

There are systems which do not contain an identification process either of system characteristics or system performance. Primarily, these may be placed in two subcategories which are herein referred to as passive linear or simple nonlinear.

Passive Linear

A passive linear system is one which has been designed to desensitize the control system to variations in vehicle dynamics using linear techniques. An excellent presentation on methods for designing linear systems to accomplish this objective is presented in Ref. 6. The inverse model is a good example of passive method for achieving a control system for which the performance characteristics are less sensitive to changes in vehicle dynamics. There will be no further discussion of passive adaptive methods except to note those systems which used the inverse model to complement the active system.

Table 2 Characteristics of adaptive techniques applicable for adaptation to variation in rigid body dynamics of aerospace vehicles

System reference number	Patent or report reference number	Authors and assignees	Nature of identification	Method of performance assurance	Control system adjustments	Adaptive system class (based on ID process)
1	P2	M. F. Marx (General Electric Co.)	None	Through nonlinear gain or error-signal	None	Simple nonlinear
2	P20	T. B. Albright (North American Rockwell—Autonetics)	None	Through nonlinear gain on error signal	None	Simple nonlinear
3	P1	H. Surtees (Fairey Aviation Co., England)	1) Measures loop gain—uses constant frequency oscillator for gain identification. 2) Indirectly measures loop gain by measuring amplitude of oscillation near gain crossover frequency.	1) Forward model 2) High-performance control system	Loop gain	Gain identification
4	P8	R. K. Smyth (North American Rockwell—Autonetics)	Measures loop gain—uses constant frequency oscillator for gain identification.	1) Inverse model 2) Constant bandwidth control system	Loop gain	Gain identification
5	P24	R. K. Smyth (North American Rockwell—Autonetics)	Measures loop gain and phase shift—uses constant frequency oscillator for identification.	1) Inverse model 2) Constant bandwidth control system 3) Constant phase margin	Loop gain and phase	Gain and phase identification
6	P7	R. S. Congleton and R. E. Paulsen (Sperry Rand Corp.)	Measures servo phase shift—uses variable frequency oscillator for identification.	Constant phase margin	Loop gain	Gain and phase identification
7	P21	G. Vasu and G. W. Hiller (Authors are assignees.)	Measures loop gain and phase—uses variable frequency oscillator for identification.	Either: constant gain margin Or: constant phase margin	Loop gain or phase	Gain and phase identification
8	P22	G. Vasu and G. W. Hiller (Authors are assignees.)	This patent supports P21 and is concerned solely with the computer which is used to compute gain (amplitude ratio) and phase shift from sinusoidal input and output signals. Conventional algebra of sinusoidal analysis is used in computer.			Gain and phase identification
9	P5	L. Kaufman (Sperry Rand Corp.)	Measures airframe gain—divides pitch angular acceleration by control surface deflection (i.e., $\ddot{\theta}/\delta$).	Constant bandwidth control system	Loop gain	Gain identification
10	P4	R. N. Bretoi (Minneapolis-Honeywell Regulator Co.)	Indirectly measures loop gain—uses induced limit cycle for gain identification.	1) Forward model 2) High-performance control system	Loop gain	Gain identification
11	P9	R. C. K. Lee and A. L. Ljungwe (Minneapolis-Honeywell Regulator Co.)	Indirectly measures loop gain—uses induced limit cycle for gain identification. System is the same in basic concept as P4 but varies in detail implementation.	1) Forward model 2) High-performance control system	Loop gain	Gain identification
12	P23	J. H. Lindahl (Minneapolis-Honeywell Regulator Co.)	This patent supports P4 with the implementation of the system described in P4 in a dual manner to assure system fail safety.	1) Forward model 2) High-performance control system	Loop gain	Gain identification

Table 2 *Continued*

System reference number	Patent or report reference number	Authors and assignees	Nature of identification	Method performance assurance	Control system adjustments	Adaptive system class (based on ID process)
13	P10	R. C. K. Lee (Minneapolis-Honeywell Regulator Co.)	Measures product of variable control system gain (K) and airframe gain ($\ddot{\theta}/\delta$)—uses pilot inputs for identification.	1) Forward model 2) High-performance control system	Loop gain	Gain identification
14	P11	R. C. Hendrick (Minneapolis-Honeywell Regulator Co.)	Indirectly measures control loop gain—equalizes dynamic energy in the vehicle short-period and gain crossover frequency bands.	1) Forward model 2) High-performance control system	Loop gain	Gain identification
15	P13	R. J. Brown and R. E. Andeen (Sperry Rand Corp.)	Indirectly measures control loop gain—equalizes dynamic energy in the vehicle short-period and gain crossover frequency bands. This patent is similar to P11. It has additional features of two AGC circuits to facilitate identification in variable gust environment and over the flight envelope.	1) Inverse model 2) Constant bandwidth system 3) Forward model	Loop gain	Gain identification
16	R10	General Electric Co. (F-111 airplane application)	Indirectly measures control loop gain—measures damping ratio of dynamic mode near gain crossover by means of bandpass filters and measurement of number of crossovers with respect to variable threshold. Use periodic pulse to assure sufficient excitation for identification.	1) Inverse model 2) Constant bandwidth system	Loop gain	Gain identification
17	R11	G. G. Anderson and J. A. Wolfe (Minneapolis-Honeywell Regulator Co.)	Indirectly measures airframe gain—measures vehicle aerodynamic derivative Z_α using linear accelerometer, rate gyro and angle of attack instrument for sensors. Shows relationships of Z_α to $M_{\delta E}$, $N_{\delta R}$, and $L_{\delta A}$ (i.e., to vehicle high-frequency gain on all three axes).	Limited variation in control system bandwidth	Loop gain adjustment at three discrete levels (gain adjusted discretely on pitch, roll and yaw axes)	Gain identification
18	P25	W. Moller (Bodenseewerk, Perkin-Elmer Co., Germany)	Indirectly measures control loop gain—measures magnitude of error signal.	Limited variation in control system bandwidth	Loop gain adjustment at two discrete levels (yaw axis)	Gain identification
19	P6	W. K. Taylor (International Business Machines Corp.)	Measures the variation in a performance index with parameter adjustment—varies control system parameters and derives the partial derivative of an index of performance with respect to parameter variation.	Optimizes performance index	Multiple parameter adjustment	Performance identification

Table 2 *Continued*

System reference number	Patent or report reference number	Authors and assignees	Nature of identification	Method performance assurance	Control system adjustments	Adaptive system class (based on ID process)
20	R14 and R15	R. P. Shipley, A. G. Engel, and J. W. Hung (North American Rockwell-Space Div. and Autonetics)	Identifies coefficients in vehicle differential Eqs. Uses a simulation of the vehicle eqns of motion to generate an error signal which is operated on to effect coefficient adjustment. Proof of the convergence of coefficients to correct values is given.	1) Forward model 2) Inverse model 3) Constant bandwidth system	Loop gain (identification permits other control system adjustments if considered desirable)	Vehicle coefficient identification
21	P18	J. Zaborsky et al. (McDonnell Aircraft Corp.)	Identifies coefficients in vehicle differential Eqs.	Minimizes performance index—uses performance index of integral of square of product of error times time (i.e., $\int te^2dt$). According to author, system is high performance (i.e., wide bandwidth).	Loop gain (identification permits other control system adjustments if considered desirable)	Vehicle coefficient identification
22	P3	C. F. White (author and assignee)	Identifies by comparison of control system error with that of ideal performance model. Test signal is used for identification.	1) Constant bandwidth 2) Constant phase margin	Loop gain and phase	Performance identification
23	P12	R. K. Smyth et al. (North American Rockwell-Autonetics)	Identifies by comparison of control system response (output) with that of performance model. A calibrated periodic test pulse is used for vehicle excitation. The system is mechanized using digital techniques.	Uniform system response to pulse	Loop gain or phase	Performance identification
24	P17	A. Under and G. H. Pfersch Jr. (The Bendix Corp.)	Identifies by analysis of transient response of control system error. Uses pilot input signals for system excitation and identification. Uses "predictor" Eqs. which inherently contain an "error performance model" which is used for identification of system performance and for system adjustment. The system is mechanized using a digital computer.	Control system "error signal" transient response	Loop gain	Performance identification
25	P18	W. A. Platt et al. (The Bendix Corp.)	This system is identical to P17 in concept. It varies only in detailed implementation.	Control system "error signal" transient response	Loop gain	Performance identification

Table 2 *Continued*

System reference number	Patent or report reference number	Authors and assignees	Nature of identification	Method performance assurance	Control System adjustments	Adaptive system class (based on ID process)
26	P14	J. J. Cattel et al. (Massachusetts Institute of Technology)	Identifies by analysis of a performance index. The performance index is to minimize the integral of of the squared error (i.e., $\int E^2dt$), where the error is the difference between the system output and an ideal system model. Uses nominal system transfer functions to evaluate the partial derivative of the performance index with parameter variations.	Forces system dynamic response to approximate ideal system model high-performance control system	Multiple parameter adjustment	Performance identification
27	P15	P. V. Osburn et al. (Massachusetts Institute of Technology)	These patents are identical in basic concept to P14. These patents differ from P14 only with respect to detail implementation of the patent and the patent claims which are made.	Forces system dynamic response to approximate ideal system model High-performance control system	Multiple parameter adjustment	Performance identification
28	P16	A. Kezar et al. (Massachusetts Institute of Technology)				

Simple Nonlinear

The simple nonlinear system is one which uses nonlinear computation on the control system error signal in order to achieve an adaptive capability. The system is also characterized by not having either an identification process, a performance index for performance evaluation, or variable parameter in the control system. Hence, it is referred to as simple nonlinear to distinguish it from other nonlinear systems which contain one or more of the basic adaptive functions.

Examples of simple nonlinear systems are systems 1 and 2 in Table 2. System 1 uses a computation method which is illustrated by the block diagram in Fig. 15.

For a perfect multiplier and divider, the block diagram algebra provides $\dot{\theta}_1\ (\epsilon/\dot{\theta}_0) = \epsilon$ or $\dot{\theta}_1 = \dot{\theta}_0$.

The system provides for a nonlinear gain increase in order to minimize the error signal. The system concept is simple and readily implemented. However, the system has several potential problem areas for which detailed analysis would be required to evaluate in a particular application. For example 1) the significance of bandwidth and threshold limits in the multiplier and divider circuits, 2) effect of high gain on system saturation and noise, 3) poor performance of system in the region of zero error, and 4) system stability.

A second technique of the simple nonlinear type is system 2. This system utilizes a nonlinear gain on attitude error such that the gain varies inversely with attitude rate. The principle is illustrated by the block diagram, Fig. 16.

The system operates on the principle that if the rate response is slow to a given command error, the loop gain is increased to speed up the response. If the rate is fast, the gain is reduced. The system thus acts to equalize the transient response at extreme vehicle flight conditions.

The simple nonlinear system is of interest in some missile applications where system complexity must be kept at a minimum, the minor reduction in sensitivity to vehicle dynamic changes is required.

Open-Loop Gain and/or Phase Identification

General

The majority of self-adaptive control systems which have been developed identify the open-loop control system gain and/or phase. This identification may either be performed by direct measurement or may consist of evaluating a parameter which is related to gain or phase.

The identification is invariably performed at a frequency above the vehicle short-period frequency. Consequently, bandpass filters are used to identify the gain or phase response in the proper frequency band.

Where oscillators are used in the direct measurement approach, the frequency of the oscillator may be held con-

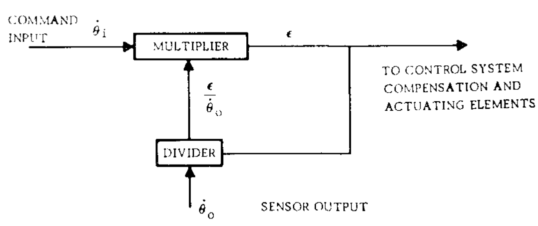


Fig. 15 One type of simple nonlinear adaptive system.

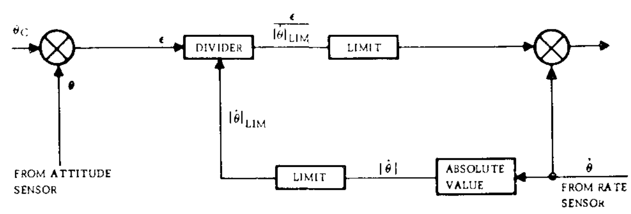


Fig. 16 Second type of simple nonlinear adaptive system.

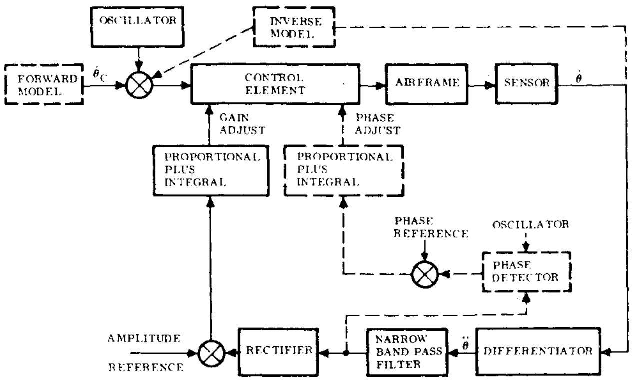


Fig. 17 One type of direct measuring gain and phase adaptive system.

stant at a preselected value, or adjusted in accordance with information derived from the identification.

After identification is made, the technique consists of adjusting gain to limit the system bandwidth variation, and adjusting phase to set the phase margin. In addition to controlling the gain crossover frequency and the phase margin at crossover, most systems provide for either a forward or inverse model in order to assure the required transient response to pilot commands. As noted, the forward model requires a fast response control system in order to achieve the desired performance objectives.

The inverse model requires a system with restricted bandwidth variation, but the system need not be a high gain fast response system. Also as noted, the use of the forward model results in modifying the vehicle characteristic response to gusts, whereas the inverse model does not.

Gain or phase adjustments are made to the control system in order to achieve a constant bandwidth and phase margin in accordance with the performance objectives. As will be noted, some systems use several discrete values of gain rather than using continuous gain adjustment. This results in reducing the bandwidth variation to an extent which is dependent on the number of discrete gain values used.

Generally, a gain or phase reference is used as a bias to the measurements made to permit adaptive loop gain or phase adjustment. Frequently, an integrator is used for phase or gain adjustment (from the reference) to improve the steady-state adaptive loop performance.

The particular systems which identify gain or phase and use these data to implement system adaptation are listed as systems 3–18 in Table 2. The identification of gain may not be direct, but may consist of measuring another system or airframe parameter which is related to gain. These systems will be reviewed with respect to their peculiarities of identification and operation.

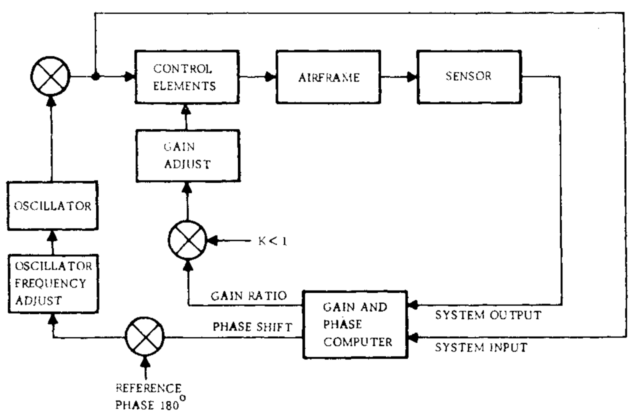


Fig. 18 Second type of direct measuring gain and phase adaptive system.

Direct measurement of gain or phase

Systems 3a, 4, 5, 7, 9, and 13 of Table 2 use identification techniques which may be classified as a direct measurement of system gain or phase. Systems 3a, 4, and 5 use an oscillator at a fixed frequency to excite the system to facilitate gain and phase measurements. The system functional operation is illustrated by the block diagram shown in Fig. 17. The differentiator, bandpass filter, and rectifier permit a measurement to be made of the output amplitude ($\ddot{\theta}$) at the oscillator frequency.

This output amplitude, when compared with a reference signal which is a function of the oscillator amplitude and the desired open-loop gain, provides an error signal which may be used for gain adjustment. System 5 has the additional feature of setting a phase margin in the system by comparing the output phase shift measured by the phase detector with a reference phase and using the difference for adjusting the system phase shift. This adaptive system approach may be used with either the forward or inverse model. System 3a uses the forward model, system 4 the inverse model, and system 5 shows neither.

Note that the oscillator represents noise to the pilot and the vehicle controls and consequently they must be subliminal (below pilot threshold) and yet above sensor and system threshold. It is indicated that a rate level of about 0.06°/sec will provide vehicle motion below pilot threshold flight conditions for the high-performance fighter considered.[13] Since a rate gyro threshold of 0.01°/sec is about the best to be achieved with conventional gyros, the amplitude must be set somewhat between these values. Note that with the system of Ref. 13 the pitch rate amplitude due to the oscillator is held substantially constant over the flight envelopes.

System 7 uses an oscillator for direct measurement of gain and phase. However, the manner of measurement and adjustment is slightly different than that described for system 5.

The principles of operation for system 7 are illustrated in the block diagram shown in Fig. 18. The system input and output signals are operated on by a computer which derives the open-loop gain ratio and phase shift using the theory of sinusoid analysis. As shown, the phase shift is compared with a reference phase of 180° to generate a phase error which is used to adjust the oscillator frequency so as to force the system at the frequency at which the open-loop phase shift is 180°. The gain ratio as determined by the computer is compared with a reference gain ratio, set to provide a specified gain margin. The error signal generated is used to adjust the gain to achieve the desired gain margin.

An alternate configuration of this system provides for the gain ratio to be used to adjust the oscillator frequency to the gain crossover frequency. The phase measurement in this case is used to adjust the control system gain to achieve a specified phase margin.

Systems 9 and 13 are quite similar in functional operation to systems 3a, 4, and 5. The primary difference is that systems 9 and 13 do not use an oscillator for system excitation

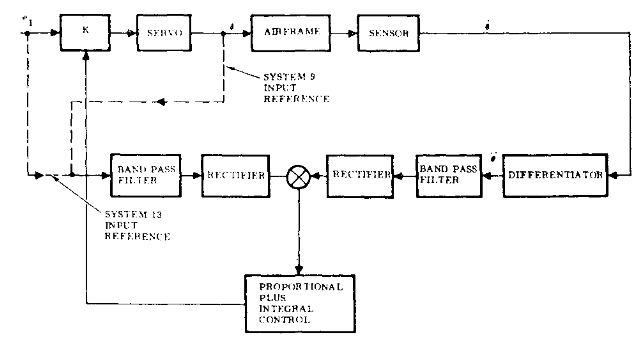


Fig. 19 Direct measuring gain adaptive system which identifies using pilot inputs for system excitation.

but depend on normal pilot inputs for identification. The block diagram for systems 9 and 13 is shown in Fig. 19. System 9 measures the ratio $\ddot{\theta}/\delta$ which is the airframe gain whereas system 13 measures the product of the variable adaptive system gain and airframe gain, i.e., $K(\ddot{\theta}/\delta)$.

Indirect measurement of gain or phase

There are numerous aircraft and control system parameters which are related to aircraft or control system gain in the region of gain crossover. Many adaptive control systems thus identify a parameter which relates to system gain and use this identification for achieving adaptive capability. Some of these systems are discussed below and the relationship which exists between system or airframe gain and the parameter used for identification noted. Systems of this type are 3b, 10–12, and 14–18 in Table 2.

Systems 10–12 derive information on control loop gain by means of an induced limit cycle. An intentional nonlinearity is introduced on the control loop. The gain is increased until a limit cycle of a predetermined amplitude is induced. The amplitude of the induced limit cycle indicates that the control system gain, including vehicle dynamics, is set at a given level. All of these systems have been designed with a forward model which requires a flat response characteristic in the basic control system. Hence, loop gain and adjustments are set to provide the required fast response control system.

A block diagram which illustrates the principles of operation of system 10 is shown in Fig. 20. The servo output is monitored at the frequency at which the limit cycle is expected to occur. The limit cycle amplitude is compared to a reference amplitude and the gain is increased if the limit cycle is below the reference and decreased if above the reference.

System 11 varies from system 10 in detail but not in the basic principles of identification. It is designed to reduce control surface chatter and system noise resulting from the limit cycle. System 12 illustrates an implementation of system 10 for dual channel operation in order to provide a "fail safe" system capability.

It may be noted that the limit cycle amplitude is controlled to a constant control surface amplitude. Since the airframe gain is a function of flight condition, this implies that the vehicle rate and acceleration amplitude (due to control surface limit cycle) varies with flight condition. This is in contrast with systems which use an oscillator and rate gyro for direct gain measurement. Such systems provide a vehicle angular rate amplitude which is constant over the flight envelope (and an acceleration amplitude which is substantially constant).

The control system design (as illustrated in the root locus plots shown in Figs. 10 and 11) has a fundamental characteristic that, as the gain is increased, the damping ratio of the closed-loop mode (near gain crossover) becomes more oscillatory. Some adaptive techniques use this characteristic for gain identification.

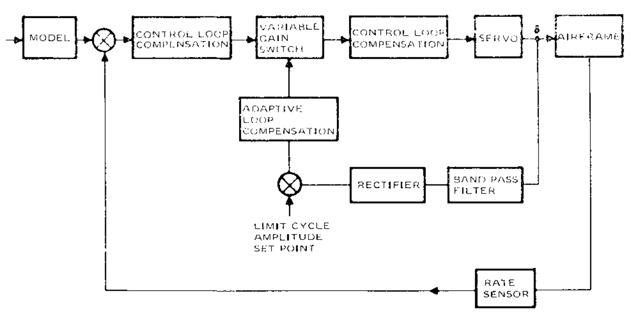


Fig. 20 Indirect gain measuring adaptive system—utilizes a limit cycle.

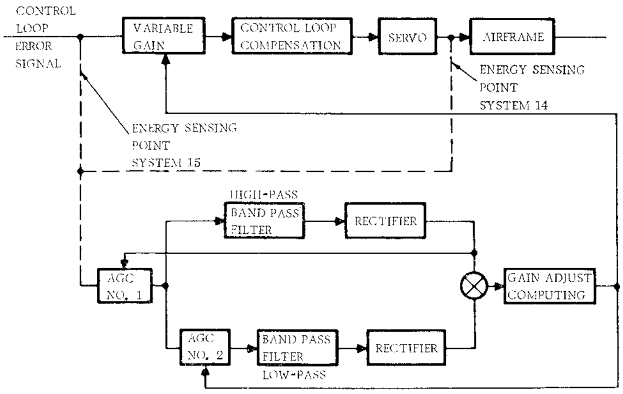


Fig. 21 Indirect gain measuring adaptive system—measures control system energy.

Systems 14 and 15 measure the dynamic energy of the system in the low- and high-frequency bands in a manner illustrated by the block diagram shown in Fig. 21. The signal, which is monitored, is passed through high- and low-bandpass filters in parallel. The output signals of the filters are compared and the difference signal operated on to adjust the control loop variable gain. System 14 monitors control surface deflection and system 15 monitors control system error. Both systems depend on natural inputs (i.e., pilot and gusts) for control system activity in order to facilitate identification.

System 15 uses two AGC (automatic gain control) circuits within the adaptive loop. The first of these (AGC No. 1) is used to adjust the gain as a function of the level of energy in the control system (i.e., to compensate for wide variations in atmospheric turbulence for example). The circuit labeled AGC No. 2 is used to adjust the relative gains between the high and low pass circuits. The need for the latter gain control arises from the fact that there is generally more energy in the system at high frequency for the high dynamic pressure conditions. The adaptive loop gain adjuster, which is a measure of the airframe gain and the dynamic pressure, is used for adjustment of AGC No. 2.

System 3b is similar to systems 14 and 15 in that it measures the energy through a high pass filter and turns the gain down if the energy is in excess of a preset level. The gain is biased so that if the energy decreases, the gain is increased.

In effect, the technique sets the high-frequency gain which, in turn, establishes the damping ratio of the dynamic mode near gain crossover. The mechanization of system 3b was given with respect to directional control. The signal monitored for this system was the output of a lateral accelerometer sensor.

It has already been noted that if a direct measure of control loop gain is made at a frequency near gain crossover, and the loop gain is adjusted so that this gain is held constant and

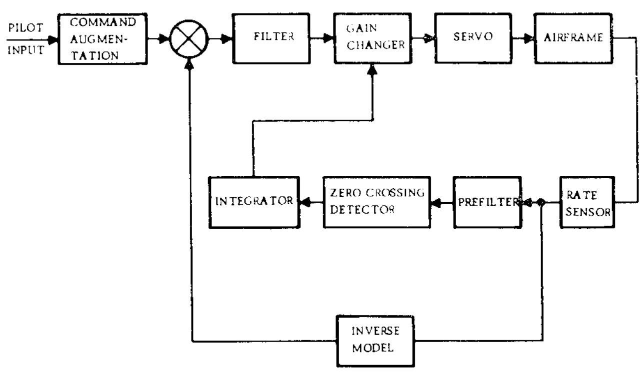


Fig. 22 Indirect gain measuring adaptive system—measures damping ratio of dynamic mode at gain crossover.

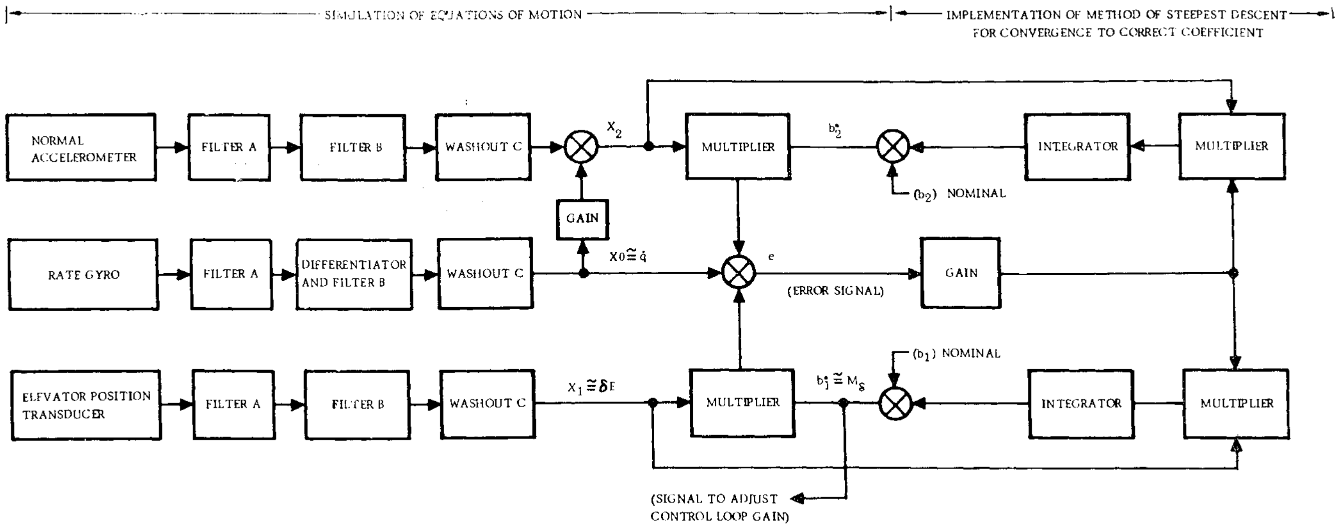


Fig. 23 Mechanization for adaptive control.

independent of flight condition, the damping ratio of the closed-loop mode near crossover is substantially constant. This is illustrated in the root locus plots (Fig. 14) for two extreme flight conditions wherein loop gain was held constant at 30 rps through adaptive loop gain adjustment. As may be observed in the figure, the damping ratio of the high-frequency mode is set to about 0.3 for each flight condition.

Hence, if a means is devised for measuring the damping ratio of the mode and the adaptive control system controls to a constant damping ratio, the performance achieved is nominally equivalent to controlling to a constant gain.

This adaptive technique is currently being used on the F-111 airplane on both the pitch and roll axes. The block diagram illustrating the principles of operation of the system is shown in Fig. 22. This system is, to the writer's knowledge, the only adaptive (variable parameter) flight control system which is in production for operational use.

The damping sensor has a bandpass filter located in a frequency band centered in 14 rps. The sensor measures the number of crossovers of vehicle rate motion about a given threshold. If three crossovers occur, no gain adjustment is made. With more than three crossovers, the system gain is reduced; whereas if there are less than three crossovers, the system gain is increased.

Problems observed in early testing of the system, and subsequently corrected were noted in Ref. 10. 1) It was found desirable to adjust the threshold for the damping sensor as a function of the noise level generated by air turbulence. If this were not done, excessive air turbulence would tend to turn the gain down. 2) Friction in the linkage between the damper sensors and the control surface actuators produces an apparent open-loop condition within the control loop. To overcome this phenomenon, special bearing installations were required in the primary control system. 3) Flow forces and friction within the surface actuator control valves and rate limiting of the actuators themselves can drastically change the actuator servo dynamics and the adaptive mode frequency. The frequency shift was sufficient under some conditions to make the adaptive concept impractical. To correct this, the horizontal tail actuator valves were servo boosted, and the rate flow capacity increased. 4) It was necessary to introduce, at regular intervals, a subliminal impulse (below pilot threshold) to assure proper identification during calm air level flight conditions. If this were not provided, experience indicated that the gain would tend to increase and result in a lightly damped adaptive mode which would ring when the aircraft was maneuvered.

It may be noted that problems 1 and 3 result from using a damping sensor for identification. If an oscillator of set frequency were used for measuring loop gain, variations in actuator or loop dynamics would not affect the identification process. The requirement of a periodic pulse for vehicle excitation is probably indicative of the fact that identification techniques which are based solely on natural inputs (pilot or gust) are doomed to failure under calm air, level flight conditions. This maladjustment may not occur often but is unacceptable even if it occurs occasionally.

Both the impulse and the oscillator for vehicle excitation must be subliminal to avoid potential pilot objection. The requirement for a high-resolution control system (Item 2), including the connections to the primary control system, is to be expected for a system which must transmit low-level signals for purposes of identification. Another indirect approach to measuring vehicle gain (M_δ) is presented in Ref. 11. The technique employed therein is to measure the aerodynamic deviation Z_α where Z_α is the rate of change of vehicle normal force with vehicle angle of attack. From the longitudinal equations of motion for the aerospace vehicle it may be shown that $N_z - l\dot{q} = -Z_\alpha\alpha - Z_\delta\delta$ where N_z is the linear acceleration normal to the vehicle and measured by an accelerometer, l is the distance which the accelerometer is forward of the center of gravity of the vehicle, $\dot{q}$ is vehicle pitch angular acceleration, $Z_\alpha = (\partial Z/\partial\alpha)/m$ and $Z_\delta = (\partial Z/\partial\delta)/m$, Z is normal force, α is angle of attack, m is vehicle mass, and δ is control surface deflection.

The previous equation may be written as $[N_z - l \text{ sq}]$ [Bandpass filter] $= -Z_\alpha[\alpha + (Z_\delta/Z_\alpha)\delta]$ [Bandpass filter]. Where [Bandpass filter], on the left and right side of the equation, are identical bandpass filters centered with respect to the vehicle short-period frequency band.

Analysis of Z_δ/Z_α (Ref. 11) has indicated that the ratio of these derivatives remains relatively constant over the flight envelope. Hence, since N_z, q, α, and δ are measured values of normal acceleration, pitch rate, angle of attack, and control surface deflection, the equation is solvable for Z_α.

In order for this identification of Z_α to be significant, it must be shown that Z_α is proportional to M_δ. The referenced report has investigated various aircraft and determined that over the flight envelope, the airframe gain factors for pitch roll and gain axis (i.e., $M_{\delta E}$, $L_{\delta A}$, and $N_{\delta R}$) are approximately proportional to Z_α. The analysis has indicated that identification of $M_{\delta E}$, $L_{\delta A}$, or $N_{\delta R}$ in this indirect fashion will result in identification errors no greater than about 50%. While a precise determination of airframe gain would permit a constant open-loop gain to be attained (as the vehicle gain varies over the range of 50–1), this magnitude of error in identification will result in an open-loop gain variation of 1.5–1.0, which is acceptable for most applications.

The system, as proposed in Ref. 11, uses the identification of Z_α to set the gains in the pitch, roll, and gain control loops at three discrete levels for each axis. Thus, rather than providing continuous gain adaptive loop adjustment, and essentially constant bandwidth, loop gain variation, and bandwidth are essentially reduced by a factor of 3.0 by this means. The intent of the design using discrete gain levels was to simplify the system implementation over a system with continuous gain adjustment.

Coefficient Identification

Various means have been described for identifying the gain and phase response of the vehicle or open-loop control system on a continuous basis. Other techniques have been investigated which are somewhat more general, in that they permit evaluation of the coefficients in the differential equations which define the dynamics of the aerospace vehicle. Two such techniques are systems 20 and 21. Detailed implementation of system 20 is described in Refs. 14 and 15. The technique of parameters identification used in this system may be described briefly as follows: The vehicle differential equations of motion are used to derive an error signal. A gradient technique utilizes the error signal to continuously modify the coefficients until the equations of motion are satisfied. Sufficiently rapid convergence by this technique has been demonstrated to provide a practical means for adaptive control. The block diagram in Fig. 23 illustrates one of several mechanizations which can be used.

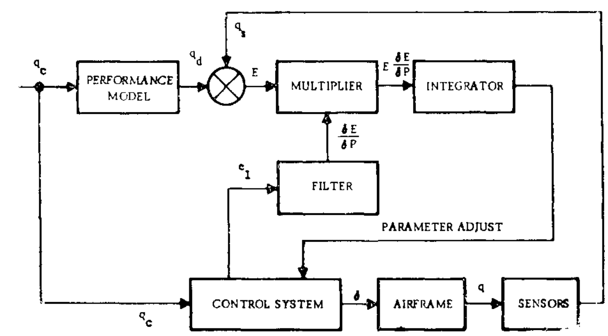
This mechanization is based on the following form of the lift and pitching moment equations: $\dot{q} = M_q q + M_\alpha \alpha + M_{\alpha E}\alpha_E$ and $N_Z = (U_0/57.3g) Z_\alpha \alpha$.

The normal acceleration at the aircraft center of gravity N_Z is related to the normal acceleration at the location of the accelerometer N_{ZP} by the relation $N_Z = N_{ZP} + (lx/57.3\, g)\,\dot{q}$.

Combining equations, after making an approximation for the term in Mq, which is a minor term in the equation, it is obtained $\dot{q} = (57.3\, g/U_0)(M_\alpha/Z_\alpha)(N_{ZP} + C_1\dot{q} + C_2 q) + M_{\delta E}\delta E$ or $X_0 = b_1X_1 + b_2X_2$, where $X_0 \triangleq \dot{q}$; $b_1 \triangleq M_{\delta E}$; $X_1 \triangleq \delta_E$; $b_2 \triangleq (57.3\, g/U_0)(M_\alpha/Z_\alpha)$; $X_2 \triangleq N_{ZP} + C_1\dot{q} + C_2 q$. An error signal may be derived as $C_0 = X_0 - b_1{}^* X_1 - b_2{}^* X_2$ where $b_1{}^*$ and $b_2{}^*$ are current estimates of the coefficients b_1 and b_2.

The coefficients could be adjusted manually simply by feeding the error into an rms voltmeter and adjusting each coefficient in turn to obtain a minimum. After several cycles of adjustment of each coefficient a null could be achieved. The method shown in Fig. 23 is automatic and uses a steepest descent technique for which the convergence is demonstrated by Lyapunov's second method in Ref. 15.

Once the coefficients in the differential equations have been identified, adaptive parameter adjustments may be made



$$\frac{\partial}{\partial P}\left(\int E^2\, dt\right) = 0$$

Fig. 24 Performance identification adaptive system—minimizes integral squared error.

to compensate for vehicle variations. For example, the control system gain may be adjusted to compensate for changes in vehicle gain (i.e., M_δ) as was done for the systems which merely identified gain changes. In addition, a pole may be adjusted in the adaptive system to correspond to the zero in the pitch transfer function which is determined by the value of Z_w. Or, if it were considered desirable, a pair of complex zeros may be used to cancel the measured airframe poles and a separate set of control system poles located in the s plane to achieve an ideal response.

Thus, this sophisticated identification permits a more refined adaptive control system to be designed. However, as analysis presented herein has indicated, it would appear that acceptable augmentation control system response characteristics may be achieved without the use of the sophistication which the vehicle coefficient identification process permits.

Performance Identification

The previously described adaptive control systems, which identify the open-loop control system gain and phase characteristics, or the coefficients in the vehicle differential equations, operate on the principle that, having performed the identification, the necessary adjustments to the control loop transfer characteristics can be made which will result in the desired closed-loop system performance characteristics.

There is another class of adaptive control systems which does not attempt to specifically identify the variations in open-loop control system or vehicle dynamics characteristics which have occurred. Instead, the identification is made on the basis of the closed-loop system performance. For these systems, identification and performance evaluation are part of a single process. Some of the techniques which fit into this class of adaptive control system are listed in Table 2 as systems 19 and 22–28, and are discussed below.

Derivative of performance index

A method of adapting a control system is to adjust the system on a continuous basis in order to optimize a performance index. This may consist of minimizing or maximizing this reference performance index with respect to variations in the parameters selected for adjustment.

System 19 of Table 2 is an adaptive system of this type. The system described is generalized and does not attempt to define a performance index for any specific application. The system is concerned with defining how system adjustments can be made to achieve a minimum or maximum of any performance index selected.

The system requires that a perturbation be introduced in the parameter to be adjusted, and the partial derivative of the performance index with respect to the parameter variation measured. The parameters to be evaluated are adjusted one at a time on a continuous basis, with sufficient time interval allowed between adjustments to properly evaluate the effect of the parameter perturbation on the system dynamic characteristics. This technique has several significant limitations in its use for aerospace vehicle applications 1) The parameter perturbation size must be sufficiently large to minimize the noise effects of air turbulence and pilot commands in identification of the value of the partial derivative. However, if it is too large, it may affect vehicle control or be disturbing to the pilot. 2) The identification process may not be sufficiently fast to cope with rapidly changing vehicle dynamics. 3) It is difficult to select a performance index that will assure good response to pilot commands, acceptable vehicle response to gusts, and reasonable gain levels which do not cause system noise or saturation problems.

Performance model—minimal of integral squared error

A self-adaptive system which is related to the partial derivative test approach described above was developed by Massa-

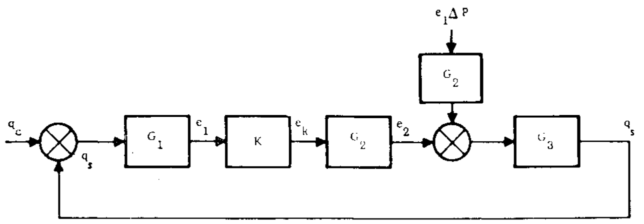


Fig. 25 Method of evaluation of partial derivative of error (*E*) with respect to parameter variation (*P*).

chusetts Institute of Technology (systems 26–28). The approach consists of minimizing the integral squared error (ISE) where the error is developed by comparing the system output with the output of reference performance model. This method removes system limitations 1 and 2, as noted for the preceding system. The performance index used is based on a desired response of the vehicle to pilot commands, and does not directly include vehicle response to gust. Furthermore, the system appears to be basically a high-gain system, and may present practical problems with respect to noise and saturation.

A block diagram of the system is shown in Fig. 24; a performance model is used to generate the desired vehicle response to a pilot command. The error signal (E) between the desired and sensed vehicle response is operated on to generate a parameter (P) adjustment which minimizes the integral of the squared error; i.e., at the minimum.

Since the limits of integration are independent of P and since, for a real system, the integral of the derivative of the function exists, the differentiation may be carried out under the integral sign. Hence, $(\partial/\partial P) \int (E^2 \, dt) = 2 \int E(\partial E/\partial P) \, dt = 0$.

The derivative $\partial E/\partial P$ which was evaluated in system 19 by test, is evaluated here by consideration of the transfer functions of the nominal system. The manner in which $\partial E/\partial P$ may be evaluated can be seen by reference to a typical control system block diagram (Fig. 25).

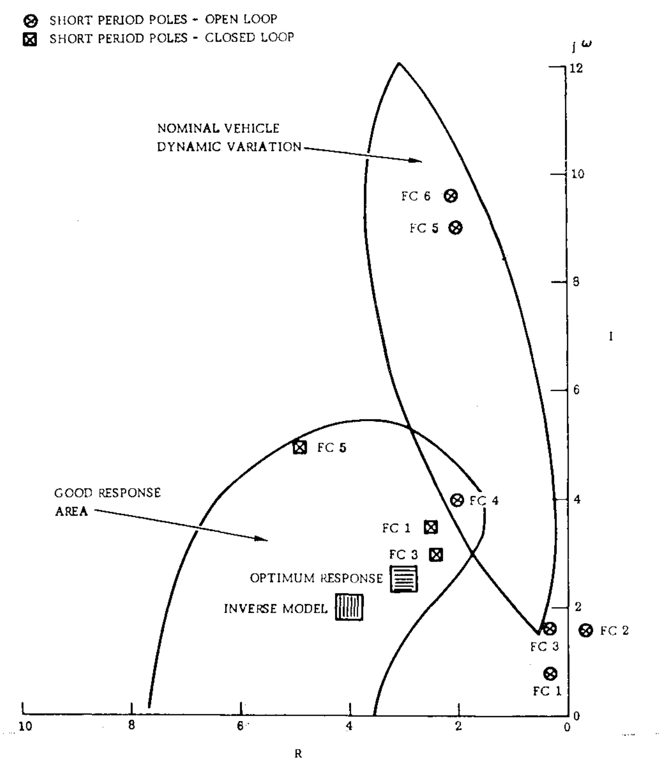


Fig. 26 Complex plane plot of vehicle short-period open-loop and closed-loop system poles.

The parameter adjustment is considered as a separate input $e_1 \Delta K$ to the block diagram such that it would have the same effect as a delta adjustment to K.

Then

$$\frac{\partial E}{\partial P} \equiv \frac{\partial E}{\partial K} = \frac{\partial(q_d - q_s)}{\partial K} = \frac{\partial q_s}{\partial K} = \frac{G_2 G_5}{1 + K G_1 G_2 G_3} e_1$$

which is determined directly from block diagram algebra by evaluating the delta change in q_s resulting from the ΔK adjustment. For the case cited, $G_2G_5/(1 + KG_1g_2G_3)$ would represent the "filter" on e_1 in Fig. 25.

A simplification of the method to facilitate implementation uses only the sign of $\partial E/\partial P$ without the magnitude; i.e., $\int E$ [sign $(\partial E/\partial P)$] dt is used for the parameter adjustment.

The system uses normal pilot commands for identification of system performance. Consequently, in calm air, it could have drift problems for lack of adequate excitation, similar to the problem experienced with the F-111 aircraft adaptive system, unless a periodic excitation were used.

Performance model of system response to pulse

System 23 is an adaptive system which employs a performance model of the desired system response to a calibrated pulse input. The system uses a periodic subliminal pulse for system excitation. The vehicle output response to the pulse is compared with a predetermined optimum response which is stored in the computer. The error signal generated is used for system gain or phase adjustment. The system is designed for implementation by means of a digital computer.

It should be noted that, with this system, both pilot commands and gust disturbances represent extraneous inputs and noise in the identification process. Since the level of the pulse must be kept below pilot threshold so not to disturb the vehicle or the pilot, the noise problem could be serious.

Inherent performance model of system error transient response

A self adaptive control method which uses a predictor located in series in the control system is described as systems 24 and 25. The predictor inherently has built into it a performance model of the transient response of the control system error.

The system implementation uses digital computation. System development was carried to the point of flight evaluation. The system tested, in addition to the predictor, used a forward model requiring a high-performance basic control loop with the associated limitations.

Table 3 Aircraft pitch rate transfer functions

Flight case	Altitude, ft	Mach	θ/δ_e
1	0	0.161	$\dfrac{-1.14\,(s + 0.415)}{s^2 + 0.604s + 0.7074}$
2	30,000	0.6	$\dfrac{-5.31\,(s + 0.478)}{s^2 - 0.692s + 2.585}$
3	30,000	0.6	$\dfrac{-5.31\,(s + 0.478)}{s^2 + 0.692s + 2.585}$
4	0	0.7	$\dfrac{-22.97\,(s + 0.761)}{s^2 + 2.088s + 20.27}$
5	0	1.2	$\dfrac{-70\,(s + 1.27)}{s^2 + 4s + 85}$
6	0	1.4	$\dfrac{-84\,(s + 1.408)}{s^2 + 4.32s + 96.64}$

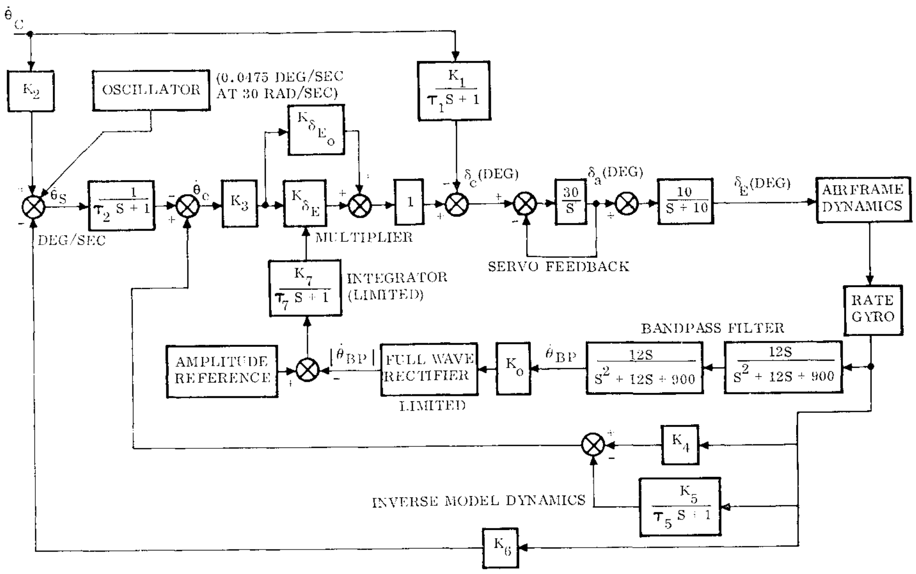


Fig. 27 Detailed mechanization of direct gain measuring adaptive control system.

Performance model of system error response

System 22 in Table 2 is a self-adaptive system which uses a model of the desired system performance to generate an identification signal. The identification signal is derived from a comparison of the control system error with the error in the model system. This system varies from the MIT system in that a comparison of errors is made rather than a comparison of output signals. In addition, the MIT system uses a performance index for setting the parameter values, whereas system 22 uses the error signal directly for gain or phase adjustment. This system suffers from both a clear performance identification and a clearly valid adjustment procedure.

Design Application

General

To illustrate the control system performance which can be achieved using the adaptive control techniques which have been described, the detail design features and test results of a system developed by North American Rockwell, Autonetics Division, for the U.S. Air Force Flight Dynamics Laboratory[13] is presented below.

The system is a direct gain identification type which uses an oscillator for system excitation, and an inverse model for short-period mode control. The system has all of the basic control system design features described for system P-8 in Table 2.

Requirements

The performance criteria used for system design are the same criteria previously described. The control system selected was a rate type of system for the pitch axis, and could be used as an inner control loop to an acceleration type system for which the transient response requirements are illustrated (Refer to Fig. 29). The objective of the design was to provide a uniform rate response system for a supersonic fighter type aircraft over the vehicle's flight envelope.

The structural coupling problem was not investigated in the design. It was assumed that this would be handled by 1) location of sensor instruments; 2) through bandwidth limitation provided by means of the adaptive control system; and 3) by appropriate additional signal filtering at frequencies above the control band.

Vehicle Dynamic Variation

The vehicle selected for the design example was a high-performance fighter airplane. Five flight conditions were selected as representing extremes in the vehicle flight envelope. A sixth flight condition (FC 3 in the example) was arbitrarily created as an unstable condition to test the system capability for identifying and properly correcting for a situation which was outside the anticipated dynamics of the vehicle.

The pitch rate transfer functions for these six flight conditions are shown in Table 3. The short-period vehicle modes are plotted in the complex s plane in Fig. 26, and compared with the nominal vehicle dynamics variations previously discussed. The extent of the dynamic variations are similar, although it would probably have been more satisfying to have checked additional flight conditions if the time in the program had permitted. The unstable flight condition which was artificial was quite significantly out of the standard vehicle envelope. (Other data shown in Fig. 26 relating to closed-loop performance are discussed in subsequent paragraphs.)

System Description

The adaptive system principles used are the same as those shown in Fig. 17 for the gain adjustment portion of the

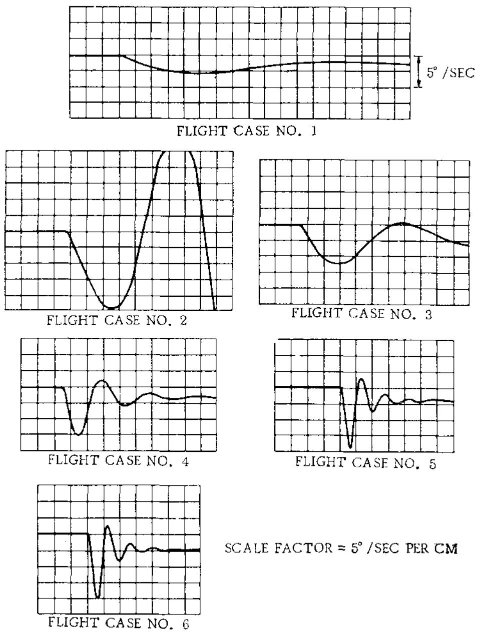


Fig. 28 Transient response of open-loop vehicle to pitch rate command of 2°/sec.

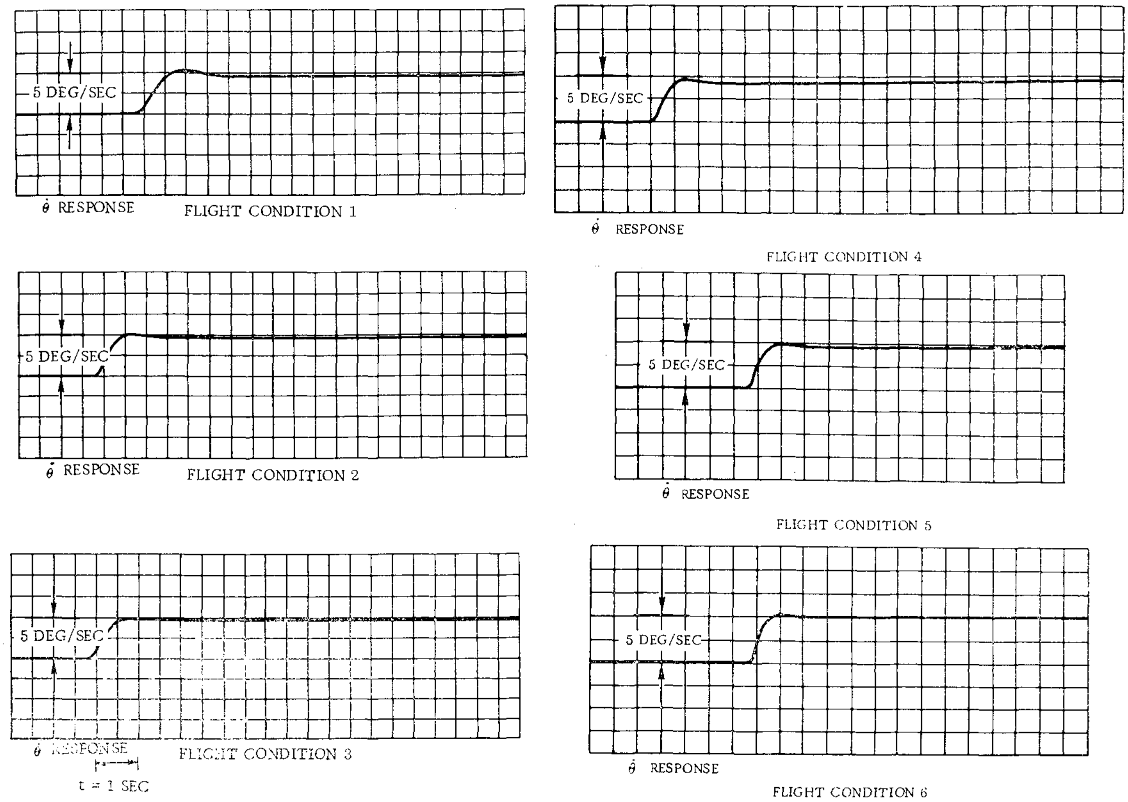


Fig. 29 Transient response of vehicle with adaptive control system to pitch rate command of 5°/sec.

system. The inverse model is used for short-period mode control. The details of the system implementation are illustrated in Fig. 27. It may be observed that the inverse model is mechanized using two feedback paths which are summed at different points in the control loop. This is a convenience for circuit implementation which does not alter the control equations. The adaptive control loop gain adjustment is a pure integrator which is limited to restrict the range of gain adjustment. An open-loop/closed-loop technique discussed in text books but not used too frequently in practice, assists in providing a faster rise time for system response to pilot commands. This technique consists of having the command signal $\dot{\theta}_c$ introduced at the normal location (i.e., compared with measured vehicle pitch rate) but also injected downstream in the control system at the input to the servos.

System Performance

The adaptive system described provided relatively uniform control system transient response and satisfied performance requirements. The effect of the inverse model in controlling the location of the vehicle short-period poles in closed loop is illustrated by Fig. 26. As shown, the three flight conditions for which the root locus was constructed are within the acceptable performance boundary.

The transient response to a step command of 2°/sec for the open-loop airframe is shown in Fig. 28. The closed-loop transient responses for the same flight conditions are shown in Fig. 29. The conformity of the closed-loop response for all flight conditions is evident. This uniformity may also be seen from the transient response characteristics which are listed in Table 4.

Electronic Equipment

The electronic portion of the system was mechanized and fabricated using solid-state analog integrated circuits employing a square wave carrier. The system developed, in addition to being adaptive, was also triple redundant using a unique redundancy concept developed by Autonetics and referred to as TRISAFE. A photograph of the equipment which, because of its redundancy, contains three normal pitch channels of computation, is shown in Fig. 30. The electronics weight was about six pounds, not including the case (which was unusually heavy and not really designed for airborne use). The self-adaptive control features represented approximately 50% of the complexity and system weight. Thus, the electronics for an equivalent fixed gain system would have been about $\frac{1}{2}$ of the weight and cost.

Summary

As noted in Ref. 13, the advantages and disadvantages of this adaptive approach are 1) Positive gain control is maintained at all time. There is no gain limit cycle, no gain error due to lapses of identification inputs, and no

Table 4 System transient response characteristics

Flight case	Rise time, sec	Overshoot, %	Damping ratio	Steady-state error, %	$K_{\delta E}$
1	1.0	13	0.6	4.0	4.0 limited
2	0.7	10	0.7	3.1	2.6
3	0.6	6	0.7	3.1	2.6
4	0.8	5	0.7	5.1	0.7
5	0.8	7	0.7	3.3	0.06
6	0.5	10	0.7	0.3	0.05

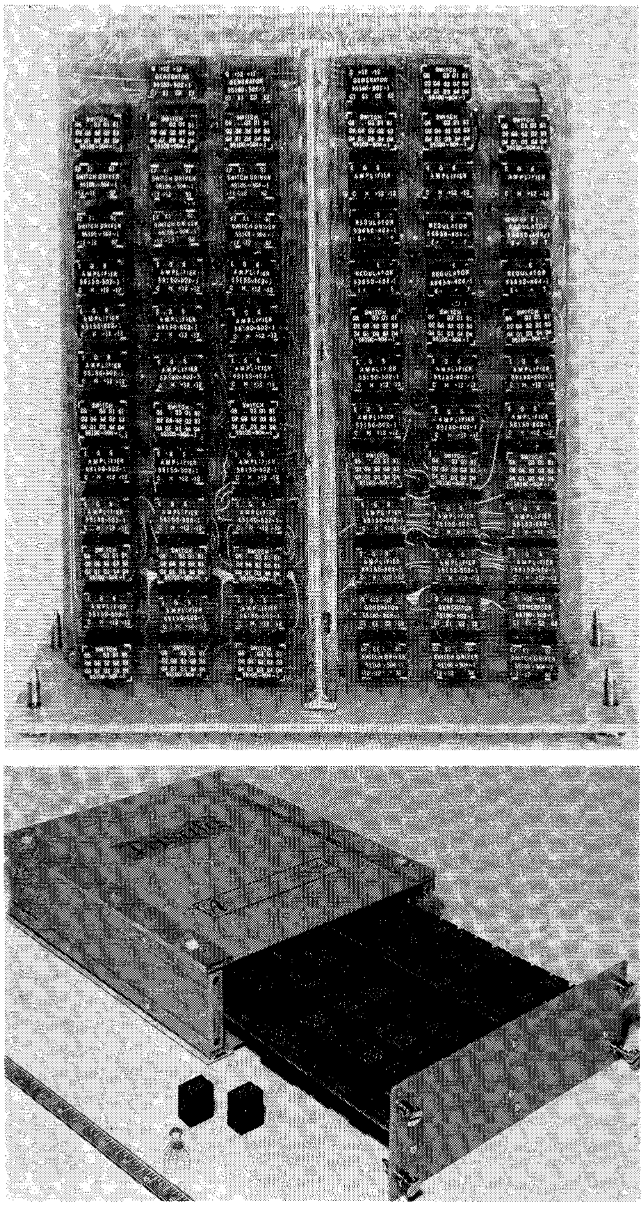


Fig. 30 Photograph of laboratory model of TRISAFE (triple redundant) adaptive control system.

significant gain error due to mismatch of oscillator frequency and bandpass. 2) High gain is not inherently required. 3) Mechanization is relatively simple. 4) Potential exists for the sensing of phase information at the oscillator frequency and for further parameter adjustment.

Possible disadvantages are 1) In order to maintain control over the gain, a relatively high-level test signal is required. 2) If bending frequencies (structural) closely approach control frequencies, it may be difficult to select an oscillator frequency high enough to exclude control energy and yet low enough to exclude bending energy.

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Patents

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P3 Variable Bandwidth Servosystem, Patent No. 2,990,504, Filed 9-10-58, Granted 6-27-61, Author: C. F. White, Assignee: C. F. White.

P4 Automatic Control Apparatus, Patent No. 3,057,584, Filed 3-1-60, Granted 10-9-62, Author: R. N. Bretoi, Assignee: Minneapolis Honeywell Regulator Co.

P5 Flight Control System, Patent No. 3,073,554, Filed 6-7-61, Granted 1-15-63, Author: L. Kaufman, Assignee: Sperry Rand Corp.

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