RECENT ADAPTIVE CONTROL WORK AT THE GENERAL ELECTRIC COMPANY

BY M. F. Marx

The adaptive control work at the General Electric Company was initiated late in 1954 under a program known as the Pilot-Airplane Link System (PAL). Since this time, a flight test program on a B-25 airplane and extensive computer simulations were carried out. Some of the results of this work were made public at the AIEE computer symposium held at Atlantic City in the Fall of 1957. The present discussion summarizes the PAL program and some of the more recent work carried out.

PERFORMANCE CRITERIA

The basic criteria on performance were adopted quite early in the program. It is required that the system be nontailoring and provide for invariant response.

The first requirement of being nontailoring is quite evident. The system should be able to adjust its feedback parameters without the assistance of external information such as air data programs require. The system should be able to adapt itself to the particular situation on hand. It is implied here that the changes to which it must adapt itself can result from changes in configuration or flight condition.

The second requirement of being capable of invariant response stems from the fact that the desired response can be specified within rather narrow limits. Invariant inner loop response greatly simplifies any additional outer loop design.

In order to indicate the various adaptive control techniques considered, the airplane pitch rate control is examined. Figure 1 presents the airplane short period plus actuator configuration which is typical of most aircraft control applications.

The airplane gain, time constant, frequency and damping are assumed variable.

In any analysis of a pitch rate control, it is important to include the dynamics of both actuators since they are the limiting items in determining the system feedback gain. The systems were first analyzed on a linear basis. If feasibility existed, the analysis was extended to include such actuator nonlinearities as deadband, saturation and variable gain.

PILOT AIRPLANE LINK APPROACH

The PAL system, which was flight tested in a B-25 airplane, was primarily a normal acceleration command system (stick force per "g" constant). The flight test program demonstrated the adaptive control behavior and the feel and handling characteristics of the normal acceleration command system. Pilot opinion indicated the maneuver limiting features and maneuverability as excellent. Subsequent NASA work has substantiated this observation.

Basically the PAL System is a multiplier-divider arrangement as indicated in Figure 2.

The divider output, δ_D, is equal to $\epsilon/K_n n$. Also since $\epsilon = F_S \ \delta_D$, $F_S = K_n n$ or the stick force per "g" is constant.

Further insight as to the dynamic operation of this system can be obtained by considering the system mechanization. This is presented in Figure 3.

Examination of the integrator input indicates that $\epsilon = \delta_D K_n n$ from which $\delta_D = \epsilon/k_n n$. In short, the integrator output is the divider output.

Noting that the two inputs to the integrator (normal acceleration and stick force) are both multiplied by the divider output, δ_D, a simplication follows by omitting this multiplication. The resulting system is presented in Figure 4.

As it stands, the system shown in Figure 4 is not satisfactory due to conditions existing at zero stick force. Under these conditions there is no feedback whatsoever. In addition, the integrator can cause the multiplier to assume any position. The actuation can be remedied by adding the constant voltage, K_1, as shown in Figure 5.

As shown, under conditions of zero input force, the integrator will run so as to drive the acceleration, n, to zero. Since the airplane is now trimmed, it follows that the multiplier gain, K_x, times the voltage, K_1, commands trim elevator deflection. For the case of linear pitching moment curves, a value of $F_S = K_1$, will result in approximately a 2 "g" maneuver. Hence, the multiplier is positioned approximately in its correct position prior to maneuver entry. The quantity, c, produces a direct feed term around the integrator to help stabilize the system.

The block diagram of the system shown in Figure 6 indicates the form of the transfer function used for analysis. For simplicity, conditions existing at trim are not included.

The important things to notice here are the terms in the feedback. The feedback gain and lead are proportional to the command and response. Thus, the response is amplitude sensitive. Furthermore, negative commands of sufficient size can result in positive feedback and consequent instability.

Of these difficulties, the polarity sensitivity is felt most serious since it limits application to the pitch channel. The B-25 tests and computer simulation of high performance airplanes support this conclusion.

Since this approach did not satisfy fully thé nontailoring and invariant response criteria, the system was replaced by the frequency sensitive servo technique.

FREQUENCY SENSITIVE SERVO

The use of pitch rate as the basic controlled variable results in better system integration than some of the other variables. The PAL program indicated the necessity of being able to vary the system dynamics suitably through the manipulation of a single parameter. Attainment of this objective, incidentally, considerably simplifies the gain changing problem by present air data programs.

The feedback configuration achieving this goal is shown in Figure 7.

This block diagram can be rewritten as Figure 8.

The root locus plot of this system has the characteristic form shown in Figure 9.

The two locii of interest are the ones starting at the airplane and parallel actuator poles. The blocks indicate the desired operating ranges. It can be shown that, for the feedback configuration shown, a value of open loop gain exists such that operation is within the desired blocks. Figures 10 and 11 present root locus plots for a high performance airplane operating at extreme flight conditions.

The system possesses sufficient gains to enable adequate cancellation of the low frequency root associated with the integrator by the airplane path time constant. For comparison purposes transient response data for the uncontrolled airplane, airplane with damper and integration, and the recommended system are presented in Figure 12. The same step pitch rate command and angle of attack disturbance were applied in all instances.

Satisfactory invariant response is achieved for all conditions investigated.Invariant response results only when the frequency of the feedback term is greater than the highest airplane frequency. In this manner, the locus enters the zeros from the same direction.

The frequency sensitive servo is a device designed to control the system gain so that the closed loop poles lie within the regions indicated in Figure 9. The operation of the device depends on the fact that the frequency of the various modes depend upon the open loop gain. Hence, if frequency errors are used to control system gain, the operation within the blocks of Figure 9 should be achieved.

The frequency detector first investigated consisted of single lead and lag network combination as shown in Figure 13.

The time constant of the networks are set at the inverse of the desired operating frequency. If the system frequency is too high the lead network output will be greater than the lag network output. Polarity is set so that the integrator reduces gain. The reverse operation takes place if the gain is too low. The depolarizers are necessary in order to accept error signals of either polarity.

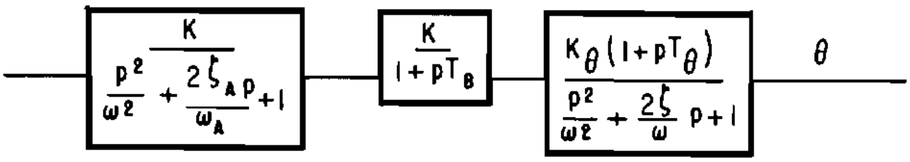
The frequency servo approach has been evaluated for the control of a simple quadratic device and found to be able to establish the correct feedback gain during one transient.

Figure 14 indicates the dynamic behavior of the frequency sensitive servo. The first traces present the response to a step input for the case where the multiplier had been positioned correctly prior to the transient. The results indicate a stable system and no sensitivity to input polarity. The third trace was prepared for the condition of zero gain at time zero. The large output from the lag network indicates that the gain increased transiently as desired. Thus it is seen that the system is capable of establishing the correct gain during a single transient.

Extension to the control of the airplane mode for the case on hand has led to difficulty due to the low frequency closed loop pole caused by the integrator. For cases where the required open loop gain is low, this pole results in low frequency components in the error which the frequency servo interprets as resulting from insufficient gain. Consequently, successive commands progressively increase the system gain until the actuator roots become oscillatory.

The frequency servo is presently being applied to control the frequency of the actuator loop directly. There are several benefits to be derived from this procedure. The low frequency components in the response due to the integrator can be filtered. This will alleviate the difficulty experienced in progressive gain increase following successive commands. The other advantage to be obtained by monitoring the actuator mode is that compensation for the effects of structural feedback is provided since the closed loop frequency is controlled in the presence of the structural feedback terms.

It has been shown that the system transient response can be made essentially invariant through the control of one variable. The only tailoring required is to ascertain that the feedback frequency is higher than the highest airplane natural frequency and that the range of the multiplier is adequate. The system is capable of self adjustment during transients without requiring special testing of input transients or steady state forcing. The system response is essentially linear with command inputs, i.e., it is not polarity or amplitude sensitive. The system techniques described can be applied to all channels without special provisions. The mechanizations are simple, employing continuous signal information thus eliminating switching or sampling requirements.



ω_A = actuator resonant frequency = 50 radians/second

ζ_A = actuator damping ratio = 0.7

T_B = power actuator time constant = 1/15 second

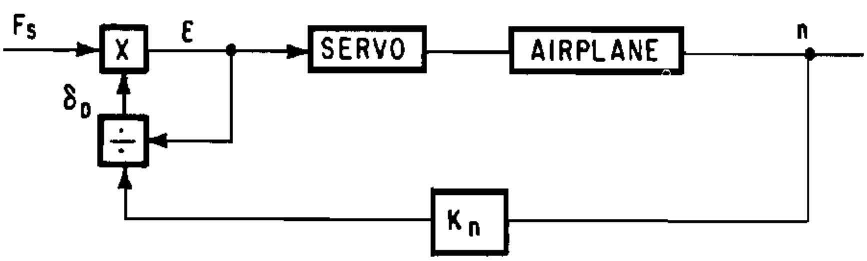
T_θ = airplane path time constant

ω = airplane short period resonant frequency

ζ = airplane short period damping ratio

K_θ = airplane short period gain.

Figure 1.



F_S = Force command

n = normal acceleration

δ_D = divider output

ϵ = servo input

K_n = accelerometer gradient

Figure 2.

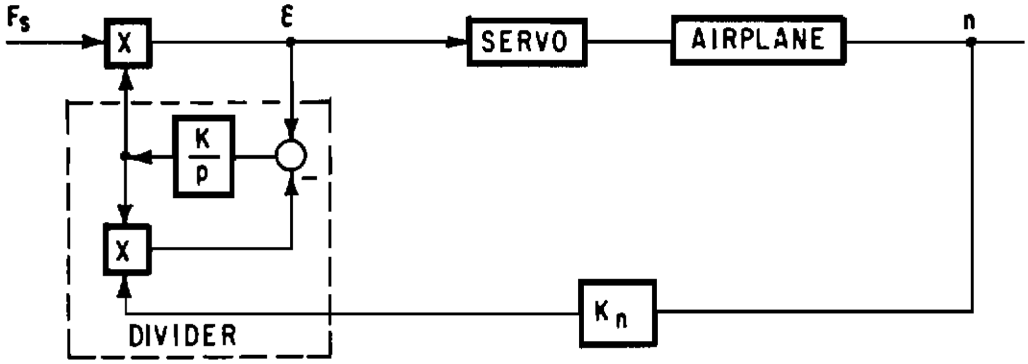


Figure 3.

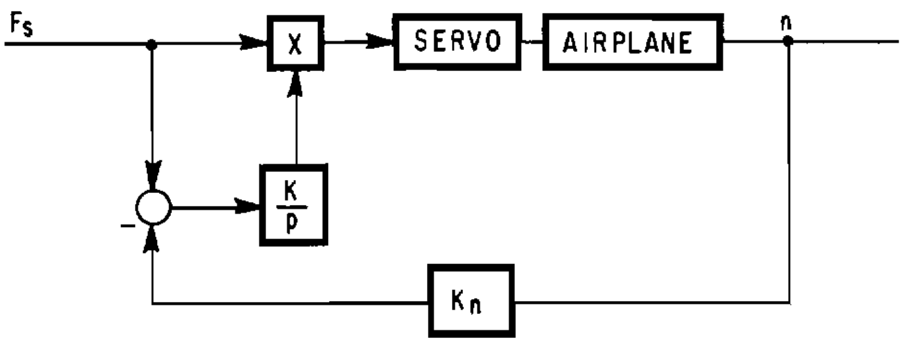


Figure 4.

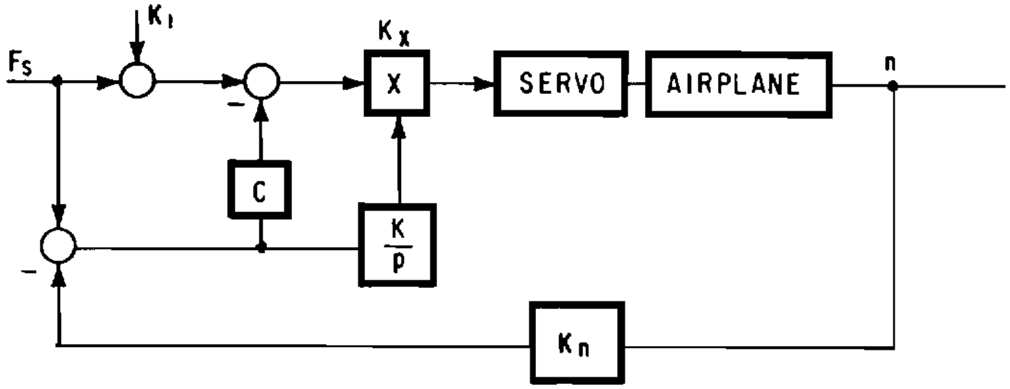


Figure 5.

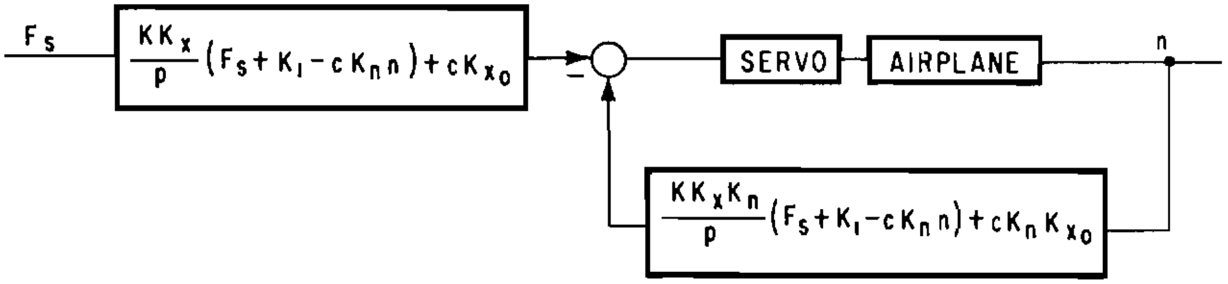
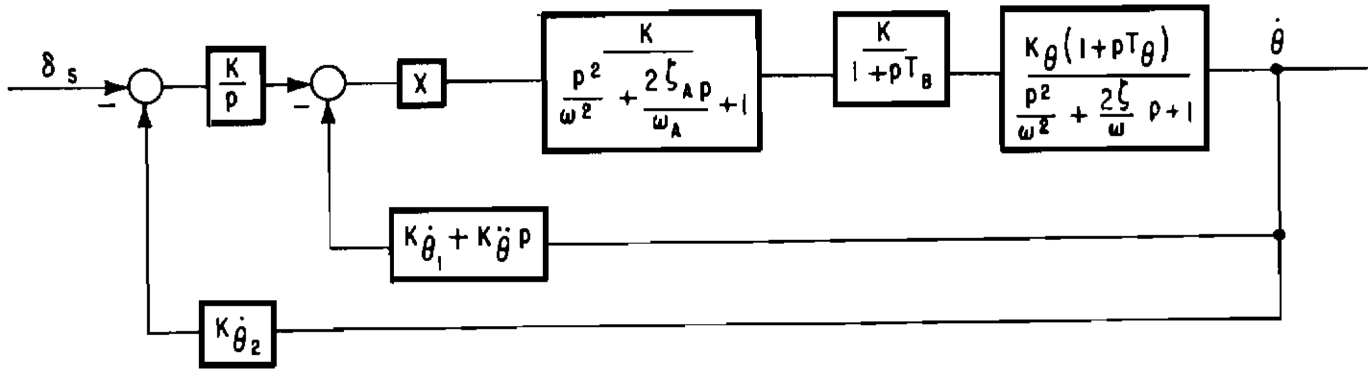


Figure 6.



δ_s = stick displacement

ω_A = parallel actuator resonant frequency

ζ_A = parallel actuator damping ratio

ω = airplane short period frequency

ζ = airplane short period damping ratio

K_θ = airplane short gain

$K_{\dot{\theta}1}$ and $K_{\dot{\theta}2}$ = rate gyro gradients

$K_{\ddot{\theta}}$ = angular accelerometer gradient.

Figure 7.

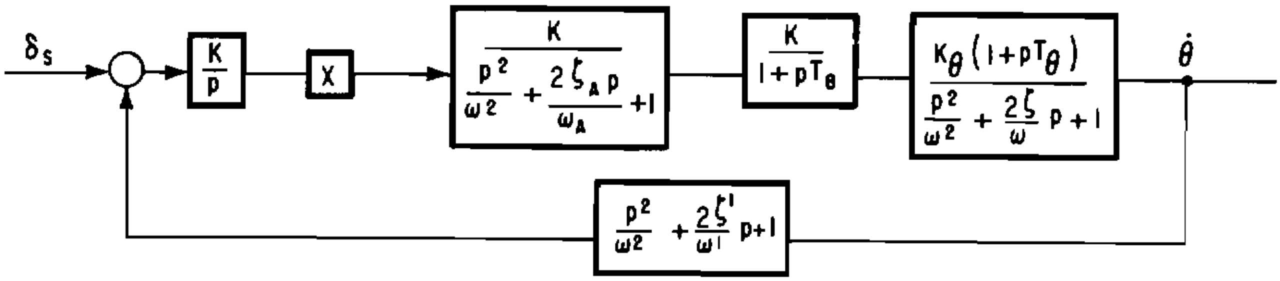


Figure 8.

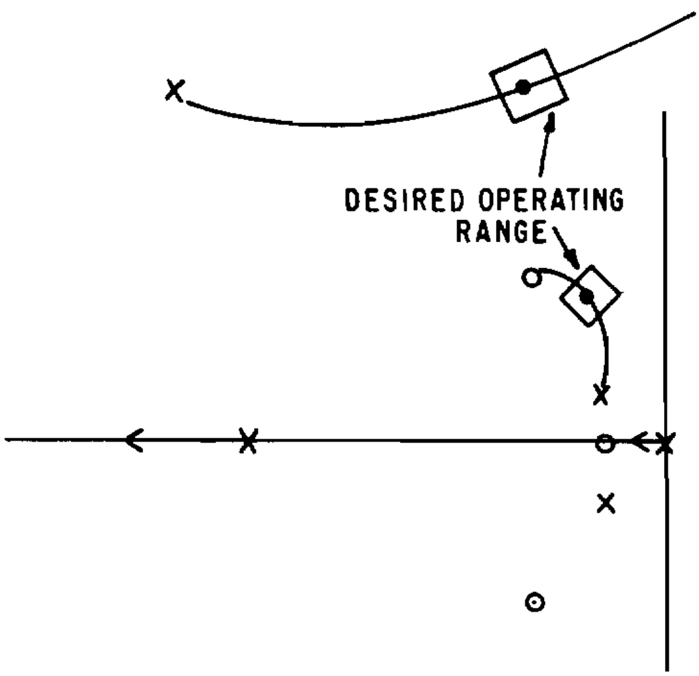


Figure 9.

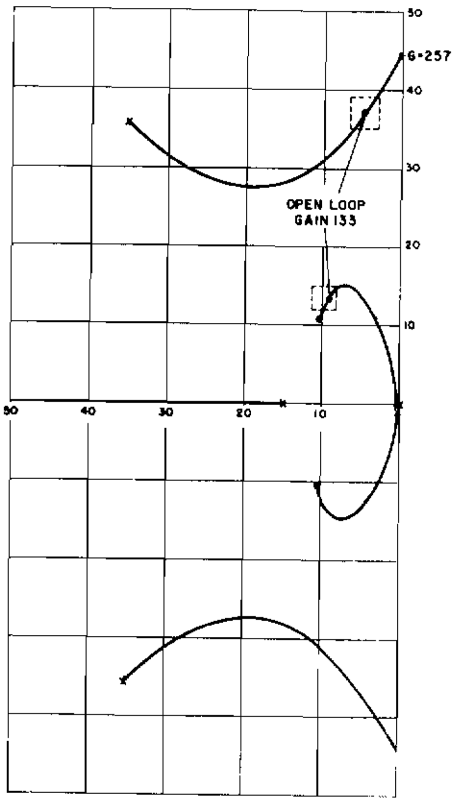


Figure 10. Linear System Having Complex Feedback

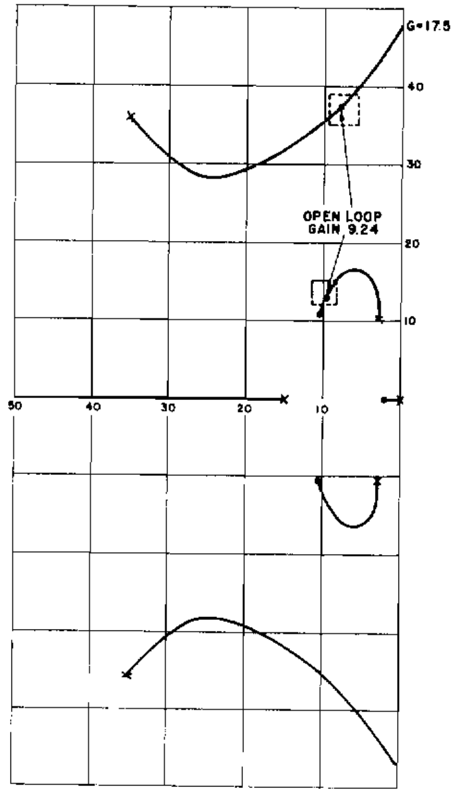
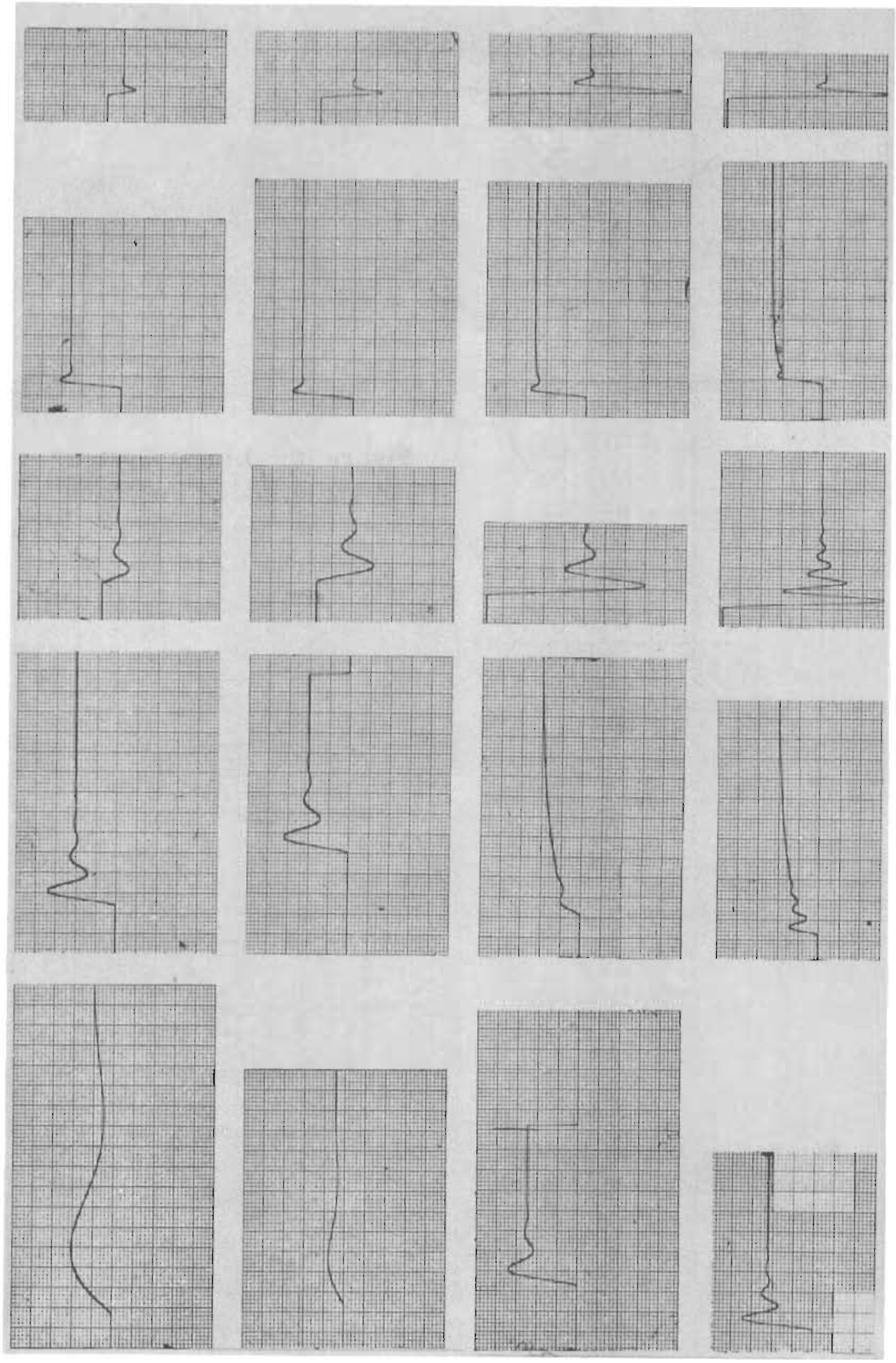


Figure 11. Linear System Having Complex Feedback

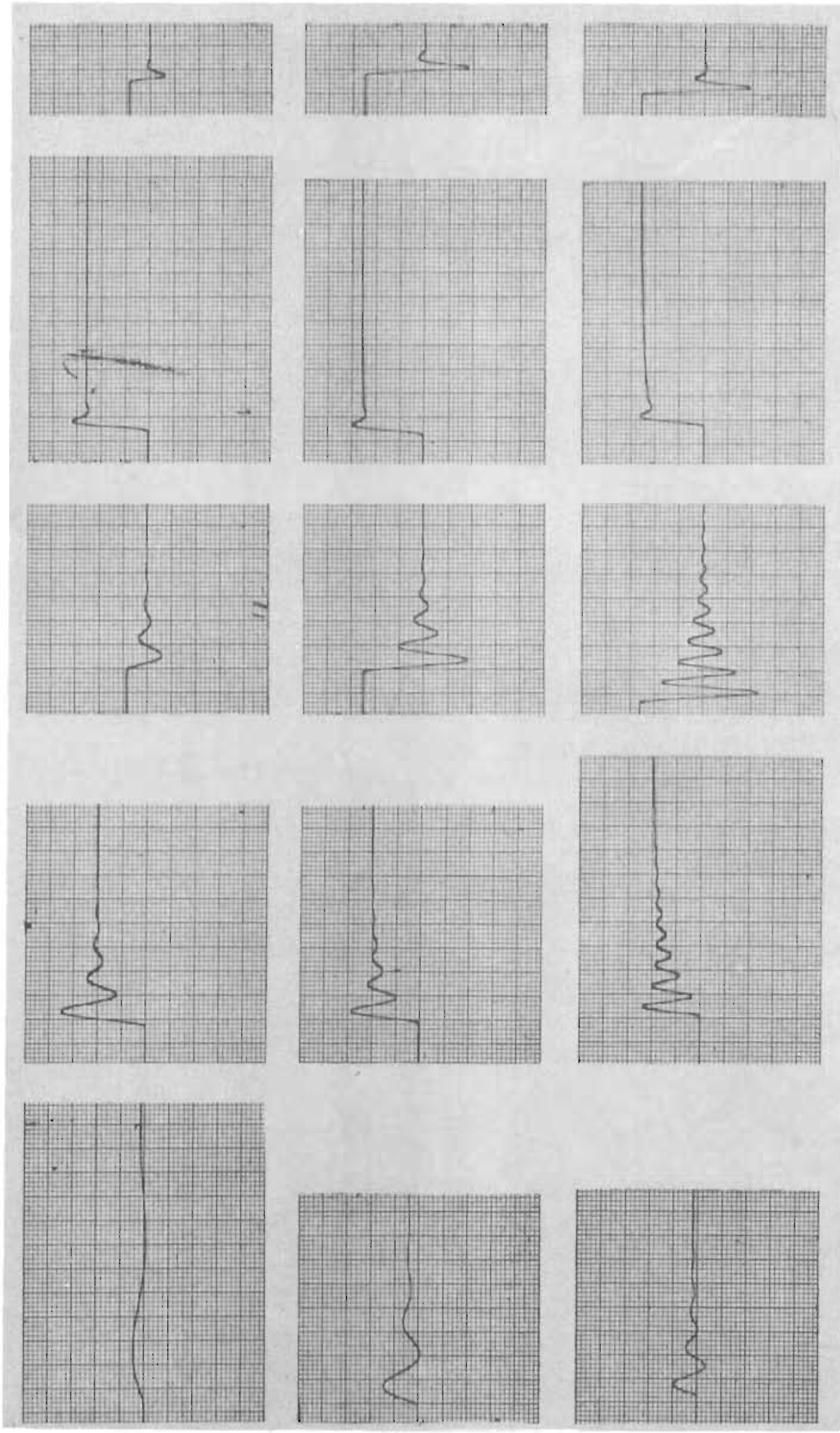


Uncontrolled Airplane

Damper and Integration

Adaptive System

Figure 12. Comparison of Transient Response Data (Part 1)



Mach. = .4
20,000'

Mach. = .95
20,000'

Mach. = 1.5
20,000'

Figure 12. Comparison of Transient Response Data (Part 2)

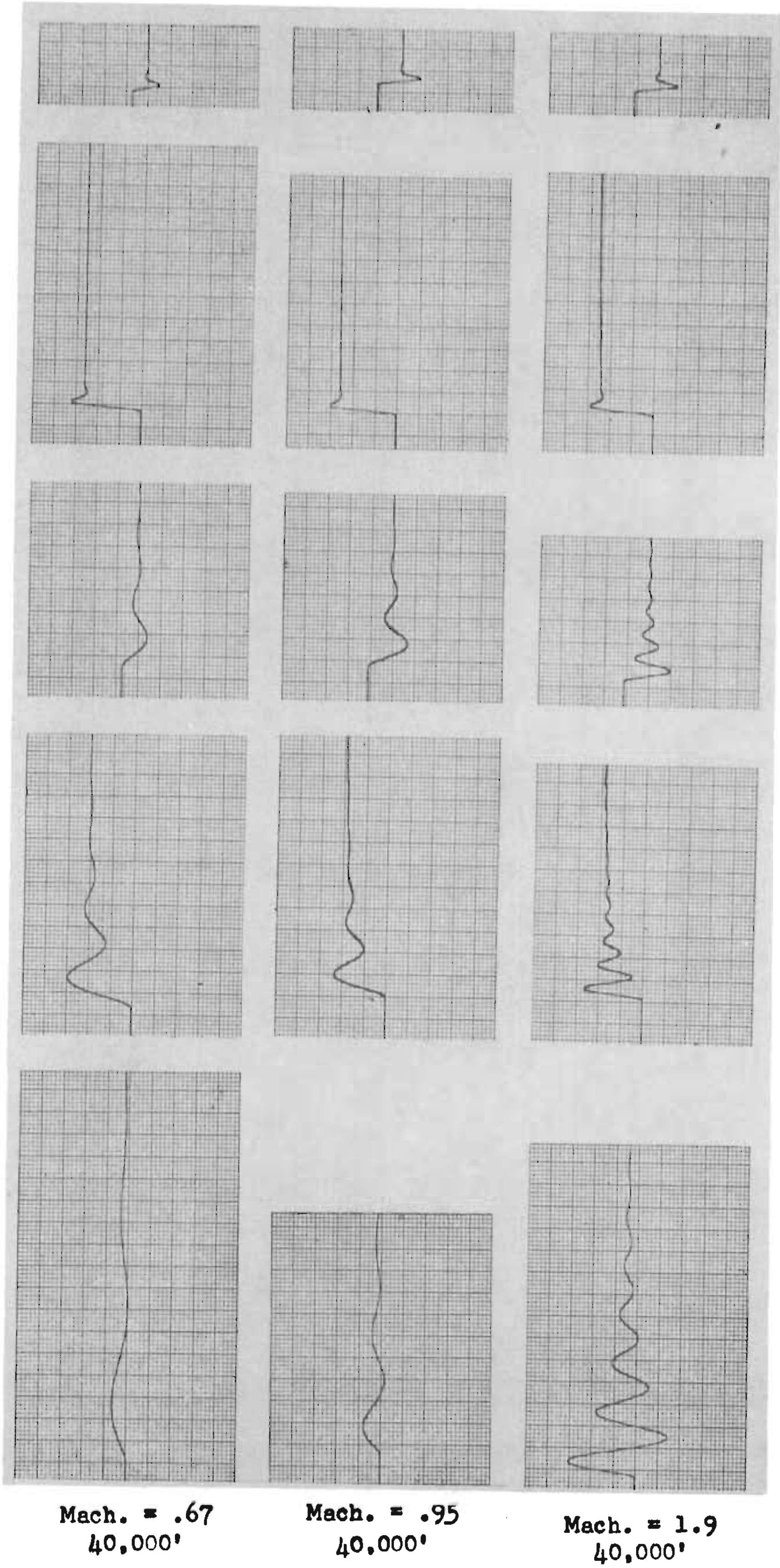


Figure 12. Comparison of Transient Response Data (Part 3)

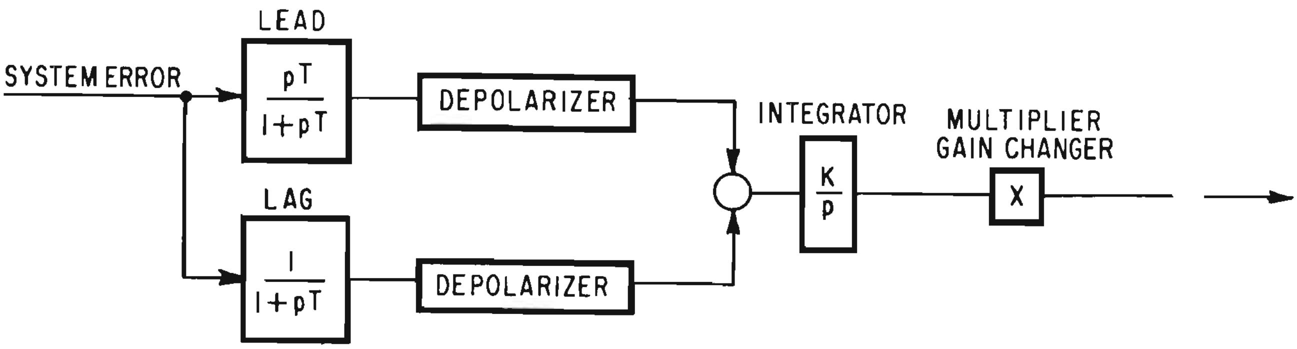


Figure 13.

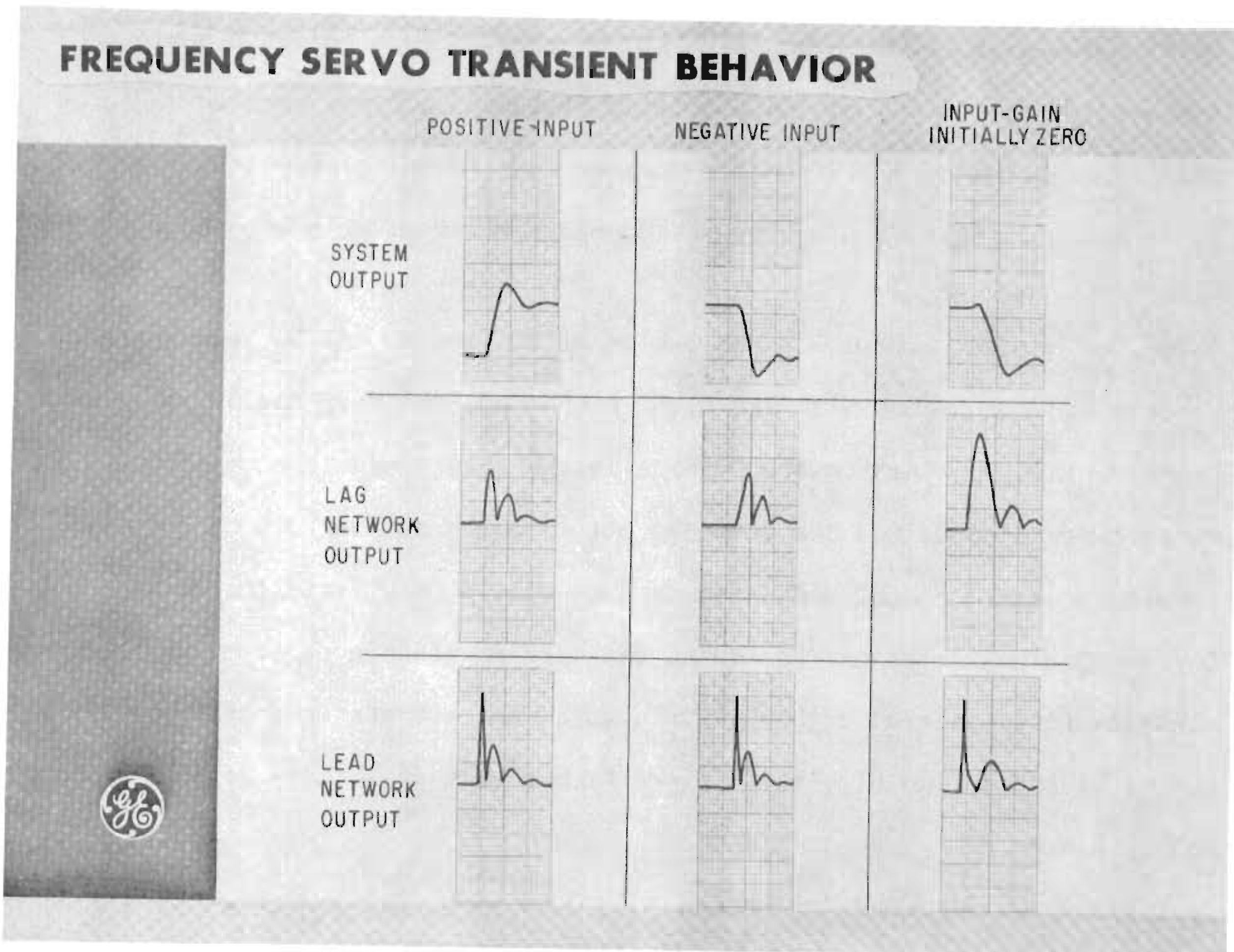


Figure 14.