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A STABILITY AUGMENTATION SYSTEM WHICH COVERS THE COMPLETE FLIGHT ENVELOPE FOR A F-4C AIRCRAFT WITHOUT GAIN SCHEDULING

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Abstract

For a McDonnell-Douglas F-4c aircraft a robust, fixed gain controller is designed, which provides satisfying handling qualities of the longitudinal motion of the aircraft over the complete flight range without gain scheduling. Robustness is achieved in the sense of covering large parameter variations and providing good gain and phase margins. Only low control rates and low feedback gains are involved. The results are obtained by application of a performance vector optimization design method which allows to take care of a great many of different design objectives simultaneously and in a highly systematic fashion.

I. Introduction

High performance aircrafts, such as a McDonnell-Douglas F-4c Phantom, can operate over a wide range of flight conditions. Because the dynamical behaviour of such an aircraft undergoes significant changes, when the aircraft changes its operating point, a suitable control and stability augmentation system always has to be capable of accommodating very large parameter variations.

The most experienced control method being used in present flight control systems is to apply a linear (dynamic) feedforward plus feedback controller structure, and to schedule the controller coefficients in a preprogrammed fashion according to the flight condition, i.e. as a function of the altitude, dynamic pressure, flap positions, etc.. Such a gain scheduling control design has proven very successful in practice. But it requires designing the controller coefficients for each individual flight condition, designing a suitable scheduling law, and implementing the sensors upon which the scheduling is to be based. Therefore the complexity of such a control system, both from the design and the realization point of view, increases the more the larger the number of quantities is with respect to which the gains have to be scheduled. But for the realization of flight control systems reliability as well as simplicity and ease of implementation are most important.

From a theoretical point of view, adaptive control methods appear to be ideally suited for solving the parameter variation problem. Attempts in this direction have been conducted within the "NASA Advanced Control Law Program for the F-8 Digital Fly-by-Wire Aircraft" as reported in [1]. However, the net result of these experiments regarding both the performance of control and the complexity of the controller was not convincing, if compared to gain scheduling control (see also the final discussion in [1], p. 806). Although much progress has been made in the theory of adaptive control since then, the inherent assumptions and nonlinearities associated with adaptive controllers raise a number of essential questions, which have not yet been resolved satisfactorily and which may so far complicate adaptive flight control applications [2], [3].

While intensive work has been devoted to finding new and more sophisticated controller structures so as to improve the control result, much less work has been done in the direction of exploring in more detail the potentials of relatively simple linear controller structures *without* gain scheduling. In [4] such a fixed gain controller was designed for the longitudinal motion of a fighter aircraft. This controller applied pitch rate and normal acceleration feedback and did provide satisfactory flying qualities over the complete flight range.

However, results such as [4] were heavily based on the physics of the aircraft and on the feeling of the design expert, how to choose two or three physically relevant controller gains. Therefore the design could, for example, not be automated to the selection of a larger number of controller gains and to the design of feedback compensation networks containing many coefficients. This may have been one of the reasons why it was concluded in [4] that normal acceleration feedback could not be substituted by dynamic pitch rate feedback compensation. Furthermore, a more general technique for generating such design results was complicated by the fact, that in such a practical design the number of different design objectives to be considered simultaneously is very large. In both respects the situation has changed since the development of systematic multi-objective design methods (e.g. [5] - [7]), which make use of efficient optimization techniques.

The present paper is concerned with the application of such a multi-objective design method for a realistic design of a stability augmentation control system for the longitudinal motion of an F-4c aircraft. The controller is a fixed gain controller, which gives acceptable flying qualities throughout the flight range of the aircraft without gain scheduling. In particular, the controller does not use normal acceleration feedbach (which is less desirable), but only applies pitch rate feedback in conjunction with dynamic feedback and feedforward compensation. Two alternatives designs using the same controller structure but placing emphasis on the time responses of the pitch rate and the normal acceleration respectively, were carried out. Here only the first one is presented. For details of both designs, see [12]. The robustness achieved in these designs concern insensitivity with respect to large parameter variations, low feedback gains, and acceptable gain and phase margins.

The work reported herein is part of an advanced technology program for future developments. The feasibility of the control design technique has been flight tested under real world effects on a small aircraft [8]. Flight tests on a high performance aircraft are being planned.

II. The Uncontrolled Aircraft and Basic Control Requirements

This paper considers the longitudinal motion of a McDonnell/Douglas F-4c fighter aircraft, as described mathematically and in numerical detail in [9], [10]. This high performance aircraft can operate over a wide range of flight conditions. Fig. 1 shows the flight envelope, where four extremal flight conditions and a "nominal" one have been selected to be considered in the design of a suitable stability augmentation system. Since the flight conditions are subject to significant changes, parameter variation effects will play an important role in the control design.

According to [9], the longitudinal motion of the aircraft is modelled by a linearized third order system of the form

$$\frac{d}{dt}\begin{bmatrix}\dot{\Theta}\\ \alpha\\ \eta\end{bmatrix} = \begin{bmatrix}a_{11} & a_{12} & b_1\\ a_{21} & a_{22} & b_2\\ 0 & 0 & -20\end{bmatrix}\begin{bmatrix}\dot{\Theta}\\ \alpha\\ \eta\end{bmatrix} + \begin{bmatrix}0\\ 0\\ 20\end{bmatrix}\eta_c\ ,$$

i.e. a second order short period motion description of the aircraft plus a first order actuator system. The state variables are $\dot{\Theta}$ ≙ pitch rate, α ≙ (incremental) angle of attack, η ≙ (incremental) elevator deflection and the input variable is η_c ≙ (incremental) elevator command. All increments are from trimmed flight conditions. The output variables of interest are $\dot{\Theta}$, α and the incremental normal acceleration of the center of gravity of the aircraft. The aircraft data a_{ij}, b_i are given in Table 1 for the above five flight conditions.

The behaviour of the uncontrolled aircraft is characterized by a conjugate complex pair of eigenvalues, which vary significantly from flight condition to flight condition. The step responses of the uncontrolled aircraft range from being very slow and well damped in flight condition 1 to being very fast and extremely low damped in flight condition 4 as is shown in Fig. 3.

It is intuitively clear that the uncontrolled aircraft would be extremely difficult to handle over the entire flight envelope. Therefore a stability augmentation system i.e. a suitable feedback control system, is necessary to improve the handling qualities. Improving the handling qualities essentially means to modify the damping and the speed of the response of the aircraft as it is desired. In addition, a disturbance regulation effect is desirable. Quantitatively the handling quality requirements are specified in [11].

III. Structure of the Controller

The variables which could be measured and used for feedback in the stability augmentation system, are angle of attack, normal acceleration and pitch rate.

Angle of attack sensing requires an external vane or probe for implementation. Such sensors are of lower reliability and are therefore not used in most stability augmentation systems. In place of angle of attack the normal acceleration is frequently used, because it is easier to measure and is very closely related (almost proportional) to the angle of attack. However, acceleration measurement is sensitive to noise, structural vibrations etc., and will therefore not be used here.

Relative to that, pitch rate sensing is of highest reliability and quality. In addition, pitch rate is more sensitive to changes of flight conditions than angle of attack and normal acceleration, which may be advantageous for sensitivity reduction effects. Pitch rate is the variable of interest having the least delay, and angle of attack and normal acceleration can be approximated by passing $\dot{\Theta}$ through a lag compensator. For these reasons pitch rate was chosen to be the only state variable to be measured and used for feedback.

Having made the decision that only pitch rate is to be fed back, we have a single-input single-output control problem. We choose a third order compensator control structure, which is depicted in Fig. 2. Here k_1, k_2, k_3 are the compensator time constants, $k_4, \ldots, k_7$ are the feedback gains, and $k_8, \ldots, k_{10}$ are feedforward compensator gains. The values of all of the controller coefficients $\underline{k} = [k_1, \ldots, k_{10}]^T$ are subject to the subsequent design. It is emphasized, that the controller coefficients $\underline{k}$ are to be designed as a fixed set of gains, which provides satisfactory control for all flight conditions.

The motivation for choosing this controller structure is as follows. One of the first order lag compensators is to provide approximate integral feedback, i.e. approximate feedback of the pitch attitude Θ. On the one hand this is to improve the disturbance rejection at low frequencies, and on the other hand it is to damp the phygoid motion of the aircraft, which is not modelled and which can be interpreted as a low frequency disturbance. The remaining two first order lag compensators can be interpreted as approximate state reconstructors for α and η (note, that asymptotically exact state reconstruction is not possible because of the parameter variations), or alternatively, as a finite bandwidth approximation of $\ddot{\Theta}$, i.e. the pitch acceleration.

IV. Design Technique and Design Results

For the design a synthesis technique based on the optimization of a vector performance index was used [7]. This technique allows to handle a great number of design objectives in a highly systematic fashion. The details of the design are reported in [12].

The controller coefficients of the final design are given in Table 2. It is seen that the feedback gains are relatively low apart from k_7 which corresponds to the approximate integral feedback filter and therefore is active at low frequencies only. Fig. 3 shows the step response of the controlled aircraft. In all five flight conditions the pitch rate behaves close to the desired response, i.e. shows essentially the same characteristics with a sliding time scale. The response of the controlled aircraft to a step disturbance, which is added to the elevator command η_c, is depicted in Fig. 4. Compared to the uncontrolled case, it is seen that the pitch rate is well damped and regulated.

The damping ratio and the natural frequency of the short period motion conforms with specifications given in [11] for all flight conditions. Equally requirements concerning gain and phase margins are performed for all flight conditions. From practical experience this allows to conclude acceptable toleration of real world effects, which are not contained in the mathematical model of the aircraft, such as nonlinearities, neglected time constants and actuator deficiencies.

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flight condition	a_{11}	a_{12}	a_{21}	a_{22}	b_1	b_2
1	-0.4615	- 0.3693	0.9792	-0.4535	- 1.459	-0.0290
2	-3.126	-72.08	1.0	-2.112	-63.48	-0.2098
3	-0.4436	- 1.803	0.9866	-0.2978	- 4.989	-0.0411
4	-0.3718	-42.75	0.9997	-0.484	-17.72	-0.0419
5	-0.7210	- 7.834	0.9990	-0.5503	-11.44	-0.05649

TABLE 1 Data of the McDonnel/Douglas F-4c "Phantom"

		$\dot{\Theta}$-Design
time constants	k_1	0.86
	k_2	0.12
	k_3	3.12
feedback gains	k_4	1.08
	k_5	-0.85
	k_6	0.53
	k_7	7.01
feedforward gains	k_8	-2.63
	k_9	3.65
	k_{10}	12.77

TABLE 2 Controller Coefficients

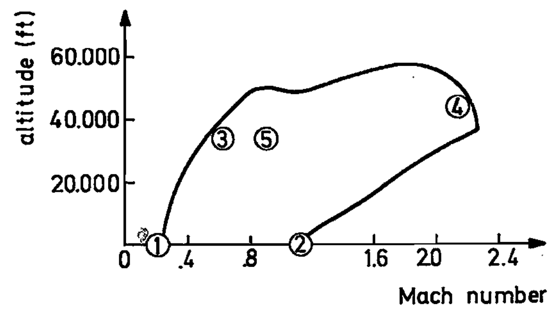

FIGURE 1 Flight envelope of the McDonnel/Douglas F-4c "Phantom"

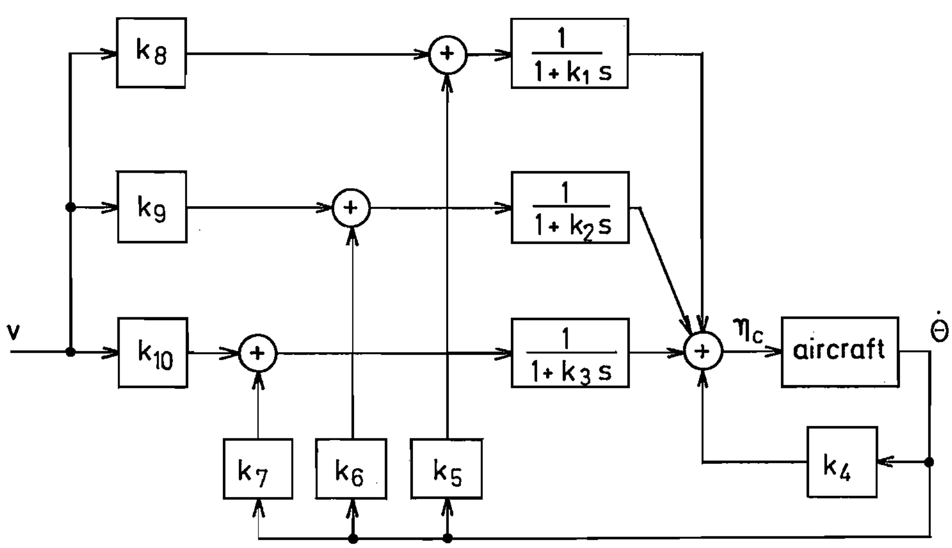

FIGURE 2 Structure of the controller

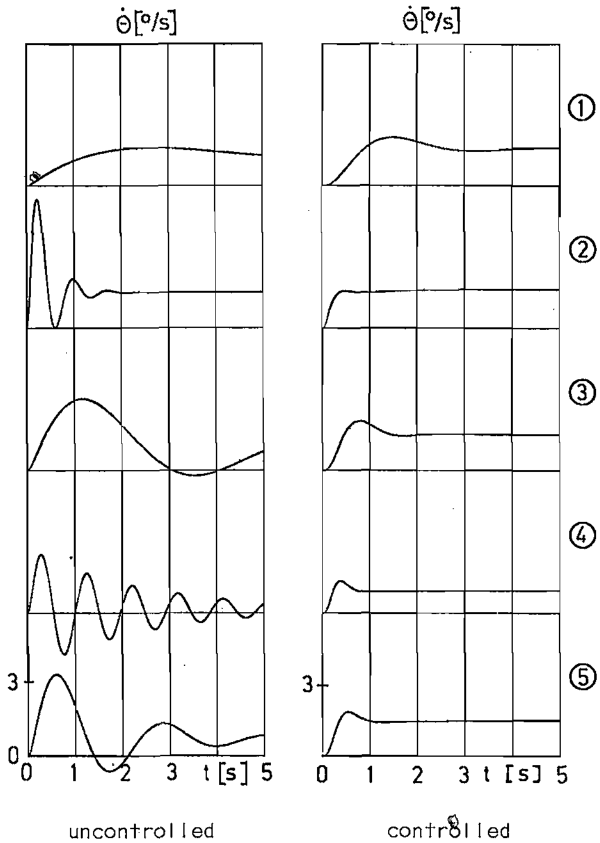

FIGURE 3 Step responses of the aircraft (flight condition 1 - 5)

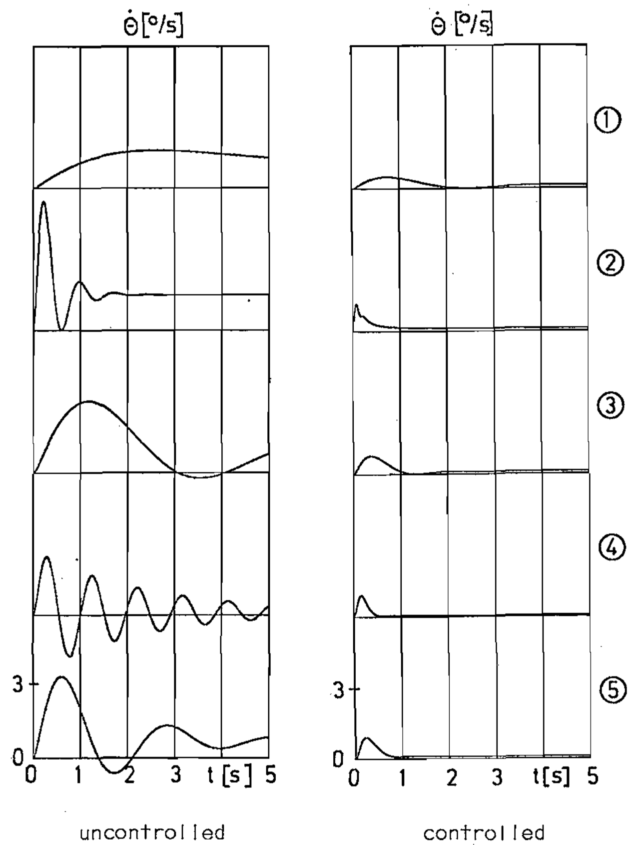

FIGURE 4 Response to a step disturbance (flight condition 1 - 5)