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ESTIMATION OF NEUTRAL AND MANEUVER POINTS OF AIRCRAFT BY DYNAMIC MANEUVERS

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Abstract

A new flight test technique, based on aircraft parameter estimation methods, is proposed **to** simultaneously determine the neutral and maneuver point of aircraft. The new procedure is derived by relating the neutral point and maneuver point of an aircraft **to** key short period parameters $\mathbf{M}_{\alpha}$ and short period natural frequency ω_n^2 respectively. The new flight test method results in substantial savings in flight test time compared to conventional methods. The method is more accurate since only inertial sensor data (pitch rate and normal acceleration) is used in the estimation procedure.

Introduction

The Neutral Point N_0 and Maneuver Point N_m are important longitudinal stability parameters which critically determine the Aft CG limit of an aircraft. Since these parameters are a function of speed. angle of attack, external store configuration, control surface deployment (slats) etc., ex-ensive flight tests are conducted to accurately determine these critical stability parameters. Existing methods based on steady state trim flights turn out to **be** time consuming and are error prone due to the results being dependent on air data and aircraft weight data. In this paper an alternative flight test methodology, based on dynamic maneuvers followed by modern aircraft parameter estimation analysis methodology, is proposed to determine N_0 and N_m simultaneously. This results in substantial reduction in flight test time. Further the estimation of the stability parameters are independent of air data, mass or inertia data of the aircraft and depend only on the accuracy of CG position of the aircraft and the accuracy of inertial sensors (pitch rate and normal acceleration).

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Definitions

The Neutral Point , $\mathbf{N_0}$ is defined as the CG position for which, in straight and level flight conditions (1-g),

$$\frac{dC_m}{dC_L} = 0 \text{ or equivalently } \frac{d\delta_e}{dC_L} = 0 \qquad (1)$$

where C_m is the moment coefficient, C_L is the Lift Coefficient and δ_e is the elevator position. The distance between N_0 and actual CG position $(N_0 - \bar{x}_{CG})$ is called the **static margin.** N_0 and $\bar{x}_{CG}$ are defined as a percentage of an aircraft reference length, typically the mean aerodynamic chord (mac) denoted by $\bar{c}$.

The Maneuver Point, $\mathbf{N_m}$ is defined as the CG position at which, under steady **pull-up** maneuvers, (in which the velocity and angle of attack (α) are held constant)

$$\frac{dC_m}{dC_L} = 0 \text{ or equivalently } \frac{d\delta_e}{dn} = 0 \qquad (2)$$

where "n" is the load factor, defined as the ratio of Lift **to** Weight. N_m is again defined as a percentdge of $\bar{c}$. The distance$(N_m - \bar{x}_{CG})$ is called the **manoeuvre margin.** It should be pointed out that under accelerated flight condition additional stability accrues due **to pitch** rate damping and thus N_m is invariably aft of N_0

Conventional **Flight** Tests **To** **Determine** N_0 and N_m

Determination of N_0 by flight tests[1] is usually done by measuring elevator angle for trim in steady flight, at a number of air speeds for different CG positions. For each reference C_L and CG position the slope $\frac{d\delta_e}{dC_L}$ is computed. Then

N_0 is determined graphically by noting the *CG* position where $\frac{d\delta_e}{dC_L} = 0$.

Maneuver point N_m is determined from flight tests by analysing data from pull-up maneuvers[1]. The pilot sets up a shallow dive at the speed and power called for in the flight condition of interest. He then pulls back on the stick and attempts to hold a steady predetermined "g" (load factor) on his accelerometer. If he is skilful, stick position and accelerations are momentarily steady with the desired airspeed holding about constant. In this technique, the elevator position for trim is really being used as an indicator of C_m, and n is, of course, an indicator of C_L, and so that the CG position where $\frac{d\delta_e}{dn} = 0$ implies $\frac{dC_m}{dC_L} = 0$.

This experiment is repeated for several load factors and different *CG* positions. The slope $\frac{d\delta_e}{dn}$ for each CG position is computed. Then N_m is graphically determined by noting the *CG* position where $\frac{d\delta_e}{dn} \equiv 0$.

Proposed method for Estimating N_0 and N_m

Consider the short period perturbation dynamics of an aircraft about a trim condition represented as a time invariant linear system in state variable form:

$$\dot{x} = Ax + Bu; \quad y = Cx \tag{3}$$

where

$$x = [\alpha \; q]' \; ; \; u = [\delta_e] \; ; \; Y = [\alpha \; q \; N_z]';$$

are the state, control and output vectors respectively with prime (') indicating transpose operator and α - angle of attack, q - pitch rate, δ_e - elevator deflection, N_z - Normal Acceleration at *CG*. The respective matrices are given by

$$A = \begin{bmatrix} \frac{Z_\alpha}{U_0} & 1 \\ M_\alpha & M_q \end{bmatrix}; \quad B = \begin{bmatrix} 0 \\ M_{\delta_e} \end{bmatrix}; \quad C = \begin{bmatrix} 1 & 0 \\ 0 & 1 \\ \frac{Z_\alpha}{g} & 0 \end{bmatrix}$$

where Z_α, M_α, and M_q are the aircraft dimensional stability derivatives. It is to be noted that M_q definition above includes the effect of $M_{\dot{\alpha}}$. M_{δ_e} is the dimensional control derivative. U_0 is the trim longitudinal velocity. g is the gravitation constant. The short period mode of the aircraft is given by the characteristic polynomial

$$s^2 - \left(\frac{Z_\alpha}{U_0} + M_q\right)s + \left(\frac{Z_\alpha}{U_0} M_q - M_\alpha\right) = 0 \tag{4}$$

where s is the laplace operator. This is in the form of a standard second order system with a characteristic polynomial

$$s^2 + 2\xi\omega_n s + \omega_n^2 = 0 \tag{5}$$

where ξ is the short period damping factor and ω_n is the short period natural frequency. From eqns (4) and (5) the relationship of the short period damping and natural frequency in terms of the dimensional stability derivatives are readily derived.

It is now possible to establish relationship between the static stability conditions given by eqns (1) and (2) with those of eqn (4). From eqn (1) we have

$$\frac{dC_m}{dC_L} = \frac{C_{m_\alpha}}{C_{L_\alpha}}$$

The stability condition of eqn (1) implies that the nondimensional derivative $C_{m_\alpha} = 0$ and from eqn (4), this means that the dimensional derivative $M_\alpha = \frac{\bar{q} S \bar{c} C_{m_\alpha}}{I_y}$ is zero, where $\bar{q}$ is the dynamic pressure, S is the aircraft reference area (wing) and I_y is the pitch inertia. The stability condition of eqn (2), namely maneuver point, indicates the CG location at which aircraft stability is lost under maneuvering conditions, and this implies one of the roots of eqn (4) is zero at that limiting CG position or

$$\left(\frac{Z_\alpha}{U_0} M_q - M_\alpha\right) = \omega_n^2 = 0$$

The above analysis shows that if the elements of the "A" matrix of eqn (3) are determined by flight tests. using parameter estimation techniques. then the Neutral Point N_0 is given by the location of CG where M_α vanishes and the Maneuver point N_m is given by the location of CG where ω_n^2 is zero.

From a practical flight test perspective this result has significant merits namely; i) Computation of N_0 is not dependent upon mass (as in the classical method) or inertia of the aircraft. ii) For determination of N_m the dynamic maneuvers required to perform parameter estimation analysis are far simpler and require less flight test time compared to the classical method. and iii) Since N_0 and N_m estimates are derived from a knowledge of the short period natural frequency and damping, accurate determination of ω_n and ξ from flight trajectory is possible, especially using only inertial sensors (q and N_z) where as in classical methods the results are dependent on air data sensors (to compute C_L) which are difficult to calibrate accurately and weight data which can only be estimated at the reference flight test point.

It is shown. in the nest section. how aircraft parameter estimation techniques can be used to compute the elements of the A. B and C matrices of eqn (3) and consequently N_0 and N_m. The proposed method, of course. is valid provided the estimation algorithm yields unique values for the elements of the "A" matris in eqn (3). By noting the number of free parameters in matrices A. B and C and correlating with the number of poles and zeros and gains to be simultaneously estimated in the $\frac{\alpha}{\delta_e}$, $\frac{q}{\delta_e}$ and $\frac{N_z}{\delta_e}$ transfer functions this uniqueness can be established.

Aircraft Parameter Estimation Method

Figure 1. gives the basis of the aircraft parameter estimation method. The aircraft is perturbed from its trim condition by applying pilot inputs to the control surface. Flight trajectories of specified aircraft response variables y (for example α, q and N_z) along with the control inputs (δ_e) are recorded. A mathematical model of the aircraft dynamics is postulated in the form of eqn (3) and the mathematical model responses y_m (α_m, q_m and N_{z_m}) are generated using the same pilot input. The error between the model response and the actual aircraft response $(y_m - y)$ is iteratively reduced by progressively modifying the model parameters (A. B and C matrices of eqn (3) till the error $e = (y_m - y)$ is reduced below a specified threshold. The converged parameters of the "A" matrix yield the desired parameters M_α and ω_n required to estimate N_0 and N_m. Many algorithms esist to perform the above estimation procedure. In this report an algorithm developed in Ref. 2. which uses the maximum likelihood estimation (MLE) criterion is used. The algorithm enjoys excellent statistical properties and also estimates the standard deviations in the estimated parameters which establish the confidence level of the parameter estimates.

Flight Test Method using MLE to Estimate N_0 and N_m

The aircraft is trimmed for straight and level flight at different C_L (different speeds). A doublet pulse (a bidirectional pulse) input is given to the elevator. The doublet input ensures that the phugoid mode is suppressed and only the short period mode is excited. This permits the use of a short period mode approximation of the aircraft dynamics as given in eqn (3). The experiment is repeated for different CG locations. Using the MLE algorithm. M_α and ω_n are computed for each CG location. Using graphical procedures, as in the conventional method. the

CG locations at which M_α vanishes (Neutral point N_0) and ω_n^2 vanishes (Maneuver point N_m) are determined.

Simulation validation of the new Flight Test Method

In this section the validity of the proposed method is established using a six-degree-of-freedom (DOF) non-linear simulation of a generic high performance fighter aircraft. A special purpose software called "A Linearising Link Software" (ALLS)[3] is used to generate the conventional flight test procedure data for computing N_0 and N_m. Using the six DOF non-linear aircraft simulation, the MLE flight test method is simulated to derive estimates of N_0 and N_m by the proposed method and the results are compared.

The "ALLS" Software

The "ALLS" software[3] was originally developed to derive linear perturbation aircraft models from a six DOF non-linear simulation. The basic principle used in the software is to define appropriate "TRIM conditions mathematically and iteratively manipulate the control settings of the six DOF aircraft model (throttle, elevator, aileron, rudder etc.,) until the defined "TRIM" aircraft state is achieved. For example, if the aircraft is to be trimmed for straight and level flight at a reference altitude and speed, the "TRIM" criterion is that all the translation and rotational accelerations must be zero and using an optimisation algorithm the "ALLS" procedure computes the control settings to achieve this condition. In the case of the pull-up maneuver, the control settings are computed such that the specified load factor (n) is achieved at the reference speed and altitude. This trim state results in a non-zero steady state pitch rate and constant angle of attack and speed conditions, which is exactly the trim condition the pilot attempts to achieve in pull-up flight tests (as desired earlier). Thus using the "ALLS" software all the data that will be required to compute N_0 and N_m using the conventional flight testing method can be derived. Further the "ALLS" software also generates linear perturbation models, about the trim state, in the form of eqn (3), which can be used as "TRUTH" models to validate the MLE derived models.

Simulation Results

Using the "ALLS" software the conventional method flight testing data are generated. Fig. 2 shows the trim curves for straight and level flight (I-g) conditions (plot of C_L vs δ_e). The trim curves are generated for four CG locations covering a range of C_L values. Notice that the trim curves are not linear with respect to C_L and thus the slope $\frac{d\delta_e}{dC_L}$ is a function of the reference C_L at which the neutral point is to be determined. Accordingly the local slopes are computed for two reference C_L values, namely 0.093 and 0.18. Fig. 3 shows the trim curves for the pull-up maneuver as a function of load factor. The trim curves are generated for four CG condition. The slope $\frac{d\delta_e}{dn}$ for the two reference C_L conditions can be computed from this figure.

The aircraft is initially trimmed at a reference $C_L = 0.18$. A doublet input is given to the elevator and parameter estimation experiments are conducted. Using the six DOF non-linear simulation of the aircraft. the angle of attack. pitch rate and normal acceleration trajectories for this input is generated. Using this trajectory data, the MLE estimation procedure is invoked by postulating a mathematical model as in eqn (3) to estimate the elements of the A, B and C matrices. Further using the "ALLS" software, linear perturbation model for the reference C_L (entries of A, B and C matrices) are also generated. Assuming that the "ALLS" model is the "TRUTH" model, Table 1 establishes the achievable accuracy of the parameter estimation procedure.

It is seen from the table that the match between ALLS and MLE values for the parameters of interest namely. M_α and ω_n are quite satisfactory. This simulation experiment validates that the parameter estimation technique yields accurate values of the critical parameters required in the estimation of N_0 and N_m.

Fig. 4 compares the MLE method and the classical method for estimating N_0. Excellent agreement is seen between both the methods at the two reference C_L conditions. Fig. 5 shows the comparison of MLE and classical method to predict the Maneuver point. The agreement for $C_L = 0.093$ is very good. However there is a small discrepancy for $C_L = 0.18$ (1.5 percent $\bar{c}$). A closer look at this difference reveals that in the classical method, appreciable C_L excursions are required to generate the required load factors (1 to 2g - Fig. 3) in the pull-up maneuver. Thus the non-linear C_L vs δ_e (as in Fig. 2) comes into picture and the measured slope is no longer a local slope. This results in a slight error in estimation of N_m. However the MLE method does not have this limitation because the MLE maneuver used to generate the trajectory data is essentially a small perturbation around the reference C_L.

Conclusions

A new flight test and analysis method, based on system theoretic concepts, to estimate aircraft longitudinal static and dynamic stability. in terms of neutral and maneuver points, is proposed. It is shown that modern parameter estimation techniques can be effectively used to compute these stability parameters. Since the stability information is estracted from the short period dynamic response of the aircraft, substantial flight test time reduction results when compared to the conventional steady state flight test procedures. Since the proposed method does not use air data information or Mass/Inertia data, the resulting estimates of the neutral and maneuver points are generally more accurate.

References

[1] Edward Seckel., "Stability and Control of Airplanes and Helicopters," Academic Press, 1964.

[2] Jategaonkar, R.V., and Plaetschke, E., "Maximum Likelihood Estimation of parameters in Non-linear flight mechanics systems," 7th IFAC Symposium on Identification & Systems parameter estimation, York U.K. July 1985.

[3] Rekha, R., and Padma Madhuranath., "ALLS - A Linearising Link Software - Links Flight Simulation to Control Law Design," NAL PD FC 9007. May 1990.

TABLE 1. COMPARISON OF ALLS AND MLE METHODS FOR COMPUTING STABILITY DERIVATIVES

	CG=0.25$\bar{c}$		CG=0.265$\bar{c}$		CG=0.28$\bar{c}$	
PARAMETER	ALLS	MLE	ALLS	MLE	ALLS	MLE
$\frac{Z_\alpha}{U_0}$	-0.81	-0.69 (0.71)#	-0.81	-0.73 (0.69)	-0.81	-0.70 (0.42)
M_α	-9.89	-9.84 (0.11)	-7.81	-7.99 (0.15)	-5.78	-5.96 (0.09)
M_q	-1.26	-1.40 (0.62)	-1.26	-1.34 (0.70)	-1.26	-1.36 (0.32)
ω_n (rad/sec)	3.30	3.29	2.97	2.99	2.61	2.63

Percent Standard deviation

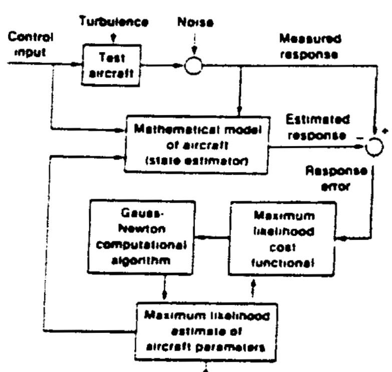

Fig 1. Maximum Likelihood Estimation Procedure

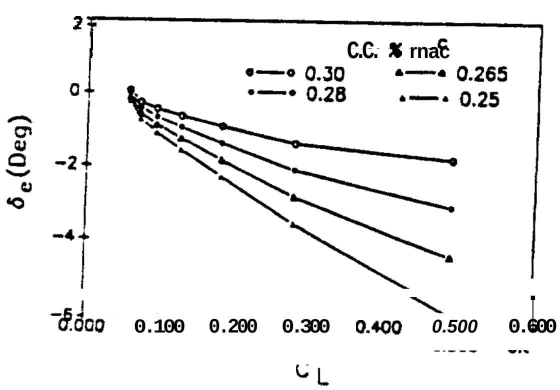

Fig 2. Level Flight Trim Curves

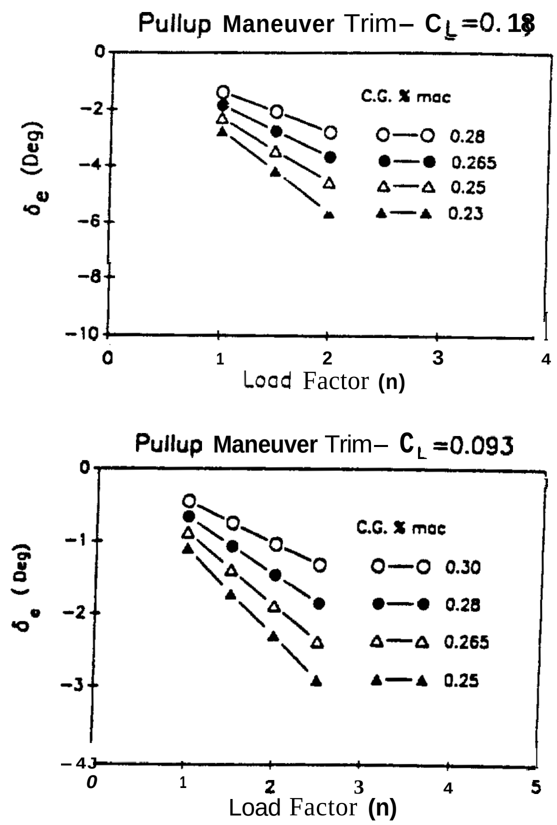

Fig 3. Pull UP Maneuver Trim

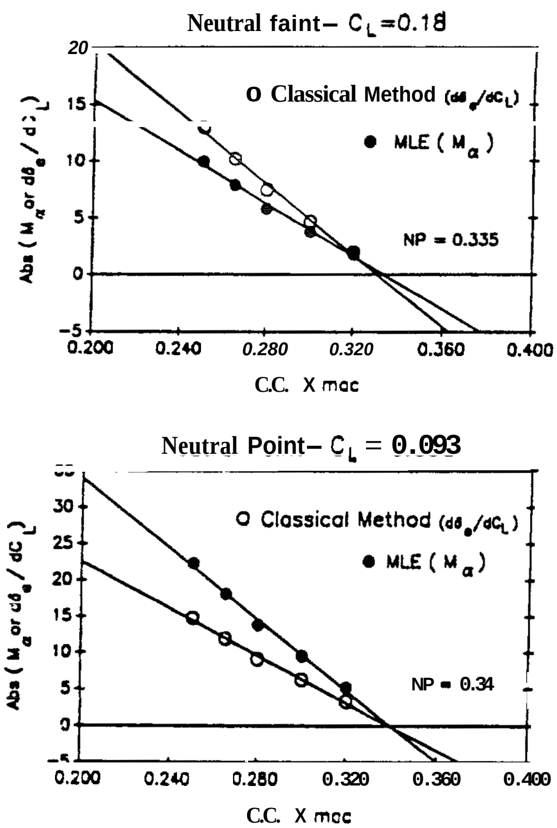

Fig 4. Comparison of Classical and MLE Methods – Neutral Paint

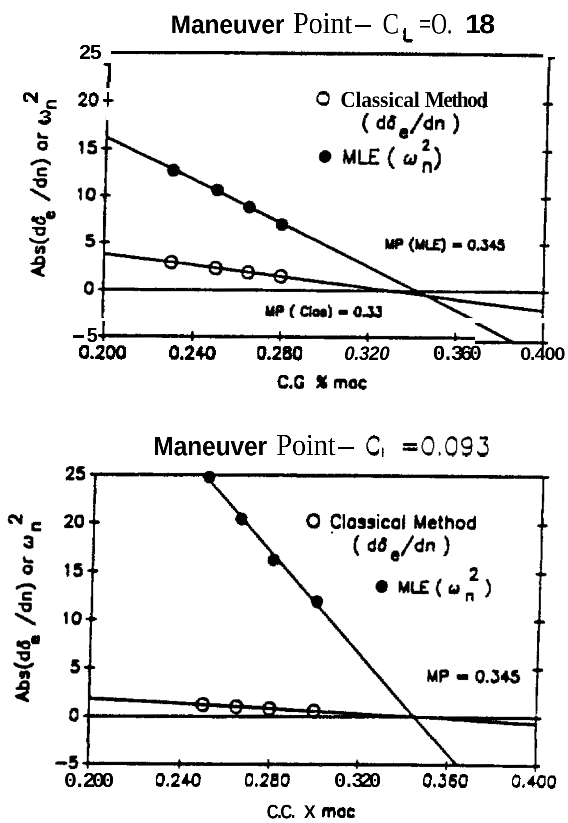

Fig 5. Comparison of Classical and MLE Methods – Maneuver Paint

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