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STRUCTURAL FLIGHT LOAD TESTING, CALIBRATION AND ANALYSIS

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Abstract

In this paper structural flight load testing is reported. The calibration procedure including strain gauge bridge selection to obtain flight loads is described. There are several evaluation methods for short and long flight periods to check design loads for static and fatigue criteria. The Maximum Likelihood method is used to investigate aerodynamic coefficients. Counting procedures are used for statistical purposes.

I. Introduction

Load measurements serve to check for adequate dimensioning of static and dynamic load cases as well as establishing critical loads, which are not adequately covered by relevant specifications. Such measurements are a requirement of military and civilian specifications and regulations. In the US, for example, MIL-A-8871 requires for certification purposes structural flight testing in addition to static and dynamic ground testing. Prior to the completion of static tests 8o% of the limit loads are demonstrated in flight, upon completion of such tests a 1oo% flight demonstration has taken place. Of late, FAR Part 25§ 25.3o1 requires such flight load tests for commercial aircraft. These tests close a logical circle, which, depending on the progress of the dimensioning and / or certification phase consists of: model testing, determination of input data for calculation and dimensioning procedures, calculation of sectional loads, static and dynamic tests and finally, control in form of flight testing incorporating the determination of the stationary and dynamic behaviour of the overall aircraft and its components.

For checking of the assumed collectives for the fatigue test, long-time in-service measurements are taken. The measured data (component loads and accelerations) are assessed on the basis of statistical procedures and thus, the actual load collectives derived. These results are taken as a basis for a comparison with and/or the required correction of the assumed load collectives.

An analysis of these load measurements in a positive case allows interference in frequently very costly major tests to achieve economical corrections. In the opposite case, i.e. with a prevailing negative results, the analysis facilitates the elimination of weaknesses prior to commencement of production.

Additionally, information is available which will provide more exact flight computer, simulator, control unit and flight control system inputs, this acquiring relevancy for direct lift control, manoeuvre load control, gust reduction etc.

These load measurements are based on strain gauges installed in the aircraft. The gauges permit precise load measurements when they are sensibly arranged and calibrated.

The incorporation of data describing the flight condition (movement parameters) enables an analysis of load portions.

II. Calibration procedure and strain gauge bridge selection

Calibrated strain gauges are commonly used to obtain flight loads. They are installed at those places of the structure, which are assumed to show linear relationship towards loading. As a rule shear bridges are bonded to spar webs, bending bridges onto spar flanges or stringers and torsion bridges onto the skin. The load calibration will be performed by applying discrete loads in a grid pattern over the surface. Strain outputs $\mu = \frac{\delta}{\delta_{cal}}$ as a nondimensional gauge response due to load will be recorded. So a load equation can be developed in the following form:

$$L = [\mu_1 \quad \mu_2 \quad \mu_3 \cdots \mu_j] \begin{Bmatrix} \beta_{11} \\ \beta_{12} \\ \beta_{13} \\ \vdots \\ \beta_{1j} \end{Bmatrix}$$

L is the load

β is the influence coefficient

In general form the calibration procedure is described in NACA Report 1178.(1)

A rectangular matrix system is generated, whose load vectors may alternately be shear, bending or torsion. The following example contains n loads and j strain bridge outputs.

$$\begin{Bmatrix} L_1 \\ L_2 \\ L_3 \\ \vdots \\ L_n \end{Bmatrix} = \begin{Vmatrix} \mu_{11} & \mu_{12} & \mu_{13} \cdots \mu_{1j} \\ \mu_{21} & \mu_{22} & \mu_{23} \cdots \mu_{2j} \\ \mu_{31} & \mu_{32} & \mu_{33} \cdots \mu_{3j} \\ \vdots & \vdots & \vdots \quad \vdots \\ \mu_{n1} & \mu_{n2} & \mu_{n3} \cdots \mu_{nj} \end{Vmatrix} \begin{Bmatrix} \beta_{11} \\ \beta_{12} \\ \beta_{13} \\ \vdots \\ \beta_{1j} \end{Bmatrix}$$

The solution of this overdetermined equation system $n > j$ is conditioned by the nonlinearity of measured values, i.e. this equation system is solved for $\{\beta_{1j}\}$, according to the method of least squares. Thus an influence coefficient is derived for each bridge.

Retroactively, a control vector for each calibrated load of the measured load can be calculated from solution β . For this, the difference between applied and measured calibration load is derived as follows:

$$\{\varepsilon_v\} = \{L\} - \{L'\}$$

Thus, the following probable error of the load vector results:

$$P.E.(L) = 0.6745 \sqrt{\frac{\Sigma \varepsilon_v^2}{n-(j+1)}}$$

Error estimation of the influence coefficients for each bridge are achieved by using terms (variances) of the main diagonal of the following matrix:

$$\begin{bmatrix} m_{11} & m_{12} & m_{13} & \cdots & m_{1j} \\ m_{21} & m_{22} & m_{23} & \cdots & m_{2j} \\ m_{31} & m_{32} & m_{33} & \cdots & m_{3j} \\ \vdots & \vdots & \vdots & & \vdots \\ m_{j1} & m_{j2} & m_{j3} & \cdots & m_{jj} \end{bmatrix} = \left[\| \mu_{nj} \|^T \| \mu_{nj} \| \right]^{-1}$$

From this, the deviations of the β- values are obtained:

$$\begin{Bmatrix} \text{P.E.}(\beta_{11}) \\ \text{P.E.}(\beta_{12}) \\ \text{P.E.}(\beta_{13}) \\ \vdots \\ \text{P.E.}(\beta_{1j}) \end{Bmatrix} = \text{P.E.}(L) \begin{Bmatrix} \sqrt{m_{11}} \\ \sqrt{m_{22}} \\ \sqrt{m_{33}} \\ \vdots \\ \sqrt{m_{jj}} \end{Bmatrix}$$

P.E. (β) must be understood as a scatter value of the coefficients. Thus redundant and irrelevant bridges can easily be detected and sorted out. However, during the frequently time - critical calibration phase it would be too troublesome, to manually prepare a new combination of matrices subsequent to the solution of the first equation system and the first P.E.(β). This will never be optimal.

The evaluation of the calibration will be done by computer program automatically:

All bridges are always incorporated into the matrix to determine β- values and error - outputs. Subsequently the program generates the quotient $\frac{\text{BETA}}{\text{P.E. (BETA)}}$. Thus sorting out the worst response column, in which this quotient holds the smallest value. In doing so, the first "bad" bridge is cancelled and the procedure automatically commences with a new, smaller measuring value matrix. This procedure is applied for all load vectors until only one bridge is left over.

The error of the load is by no means at a minimum when incorporating all bridges. On the contrary it rises upon sorting out of a few measuring value columns and rises again when a small number of bridges prevails. (See figure 1)

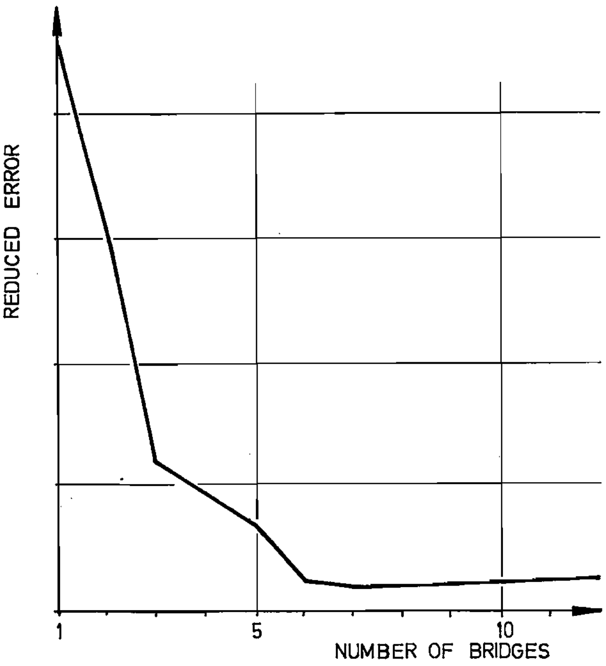

FIGURE 1. ERROR DUE TO BRIDGE ELIMINATION

When selecting the bridges the decision should rather be made in favour of a small number of bridges with a reasonably acceptable error, since most measuring points are no longer accessible after failure. Even in case of a bridge failure new combinations which can be calculated beforehand can be prepared.

So the optimal bridge selection can be carried out in regard to load error and number of strain gauge bridges. The realization can be done either in generating electrical bridge summation circuits or by recording single bridge responses and using computer to add those signals to obtain pure load measurements.

Electrical combinations have been successfully used in flight load testing of military aircrafts TRANSALL C16o, VAK 191B, TORNADO undercarriages and in the civilian aircraft VFW 614.

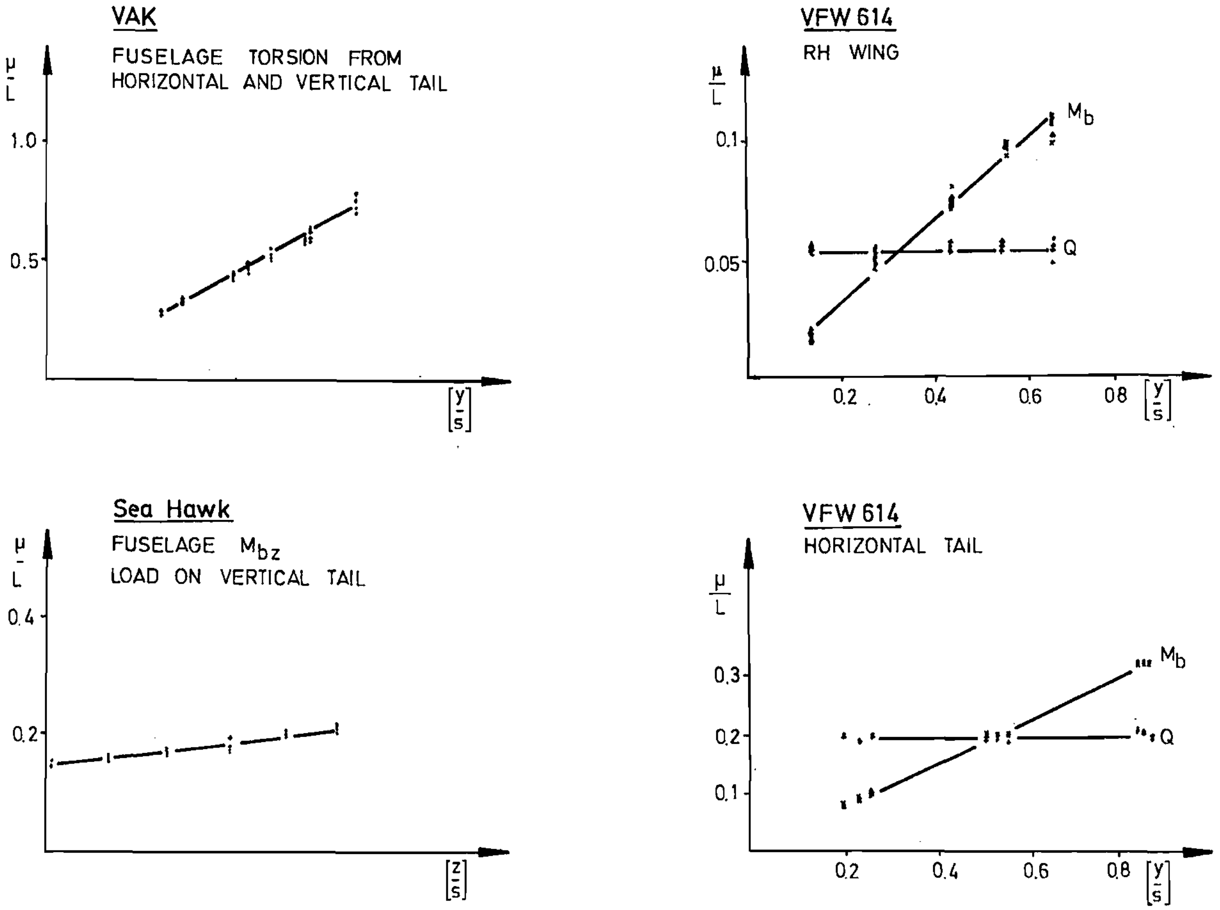

FIGURE 2. INFLUENCE COEFFICIENT PLOTS

III. Survey of Evaluation Techniques

In order to check assumptions and tests relating to the structure, evaluations must be carried out for the entire frequency range ($0 \leq f \leq 5000$ Hz). In the case in question, a distinction is made between dynamic and static problems.

To permit an assessment of service life issues on the basis of flight tests, it is imperative that long-term measurements be performed with definite load parameters. Such long-term measurements must cover a period of at least one year to fulfill certain statistical safeties and to make allowance for seasonal meteorological influences. Moreover, the data are to be acquired in scheduled service in order to obtain information on characteristics typical for service conditions.

Statistical counting methods are employed to evaluate such a large quantity of data. Detailed descriptions of this are given in the references . (10), (11)

Standard methods worth mentioning:

- range pair
- level crossing
- peak counting

These are one-dimensional methods

describing the measured parameter as a function of the frequency.

Two-dimensional methods are also applied, showing the dependence of two parameters as a function of the frequency. (Example: connection between vertical tail load and the roll moment at the horizontal tail or the landing gear loads in forward and lateral direction).

Long-term measurements involve a minor test scope with reference to the life of an aircraft. Extreme value distributions are used for extrapolation of the evaluated spectra with regard to the life. Unfortunately, such evaluations of scheduled flights are only available very late, so that it is necessary to perform such investigations beforehand during flight tests. Many ground taxi runs and ferry flights to test locations can be used for this purpose.

In point of fact, the result of long-term measurements serves to check out the loads assumed for fatigue. Corrective measures can be introduced into the demonstration calculations and tests. Application to aircraft variants is very helpful.

From the point of view of timing, the check of static demonstrations is more favourable. Structural flight tests should commence early on in the flight testing phase. Corresponding regulations give definitions of the flight conditions with which the load level is checked. In contrast to the civil regulations FAR 25 § 3o1, the MIL specification A-8871 gives a very clear definition of the structural flight tests, detailing the components and locations at which loads should be measured. This specification gives basic values of the flight spectrum (e.g. Mach number, altitude combinations) as well as defined abrupt manoeuvres with the aim of reaching a predetermined load level. Prior to completion of the static laboratory tests, 8o% of the limit load and after the test 1oo% of the limit load must be demonstrated in flight. Extrapolation to the limit load should here take place at an early stage.

The aim is to disturb the dynamic system, i.e. the aircraft in such a way that a clear answer can be evaluated. An abrupt short manoeuvre is very unsuitable for this. Although a high load level is reached for a short time, the dynamic informatory content is small. Control surface inputs which excite the aircraft to such an extent that both the rigid-body motion and phygoids are clear, are more favourable. Extensive examinations are detailed in references (8). We have decided in favour of the multi-stage signal that can be controlled by the pilots.

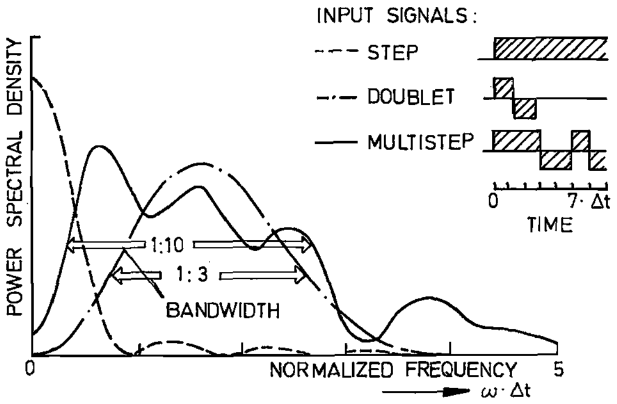

FIGURE 3. FREQUENCY DOMAIN COMPARISON OF VARISOUS INPUT SIGNALS

see Ref.8

After some practice, the pilots were able to apply the control surface inputs so that the performance spectrum of the disturbance is satisfactorily complete.

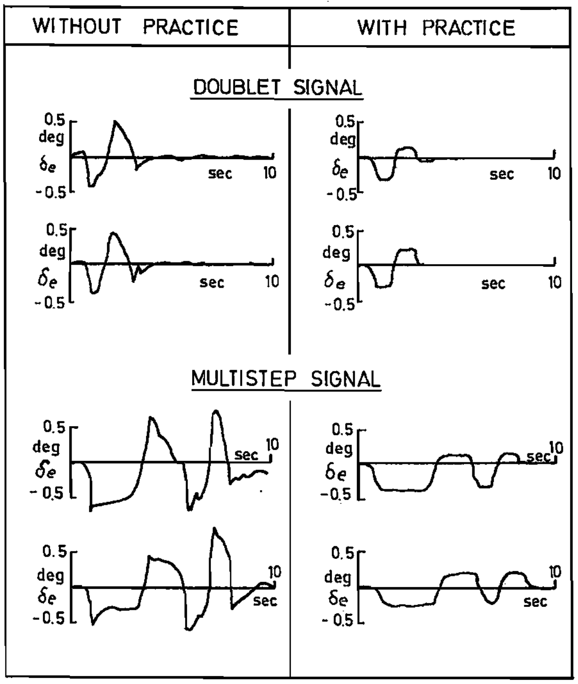

FIGURE 4. PILOT FLOWN INPUT SIGNALS
see Ref. 8

Aerodynamic coefficients obtained by way of wind tunnel measurements and empirical procedures serve as a basis for dimensioning in the design phase. An important step in the certification phase during flight testing is to check these values.

In the past, so-called digital matching procedures were used by us to recalculate the flight load parameters. The measurement and calculation were adapted by manually manipulating the aerodynamic coefficients entered into the mathematical model. This procedure is uneconomical because it requires much time and requires considerable experience on the part of the user.

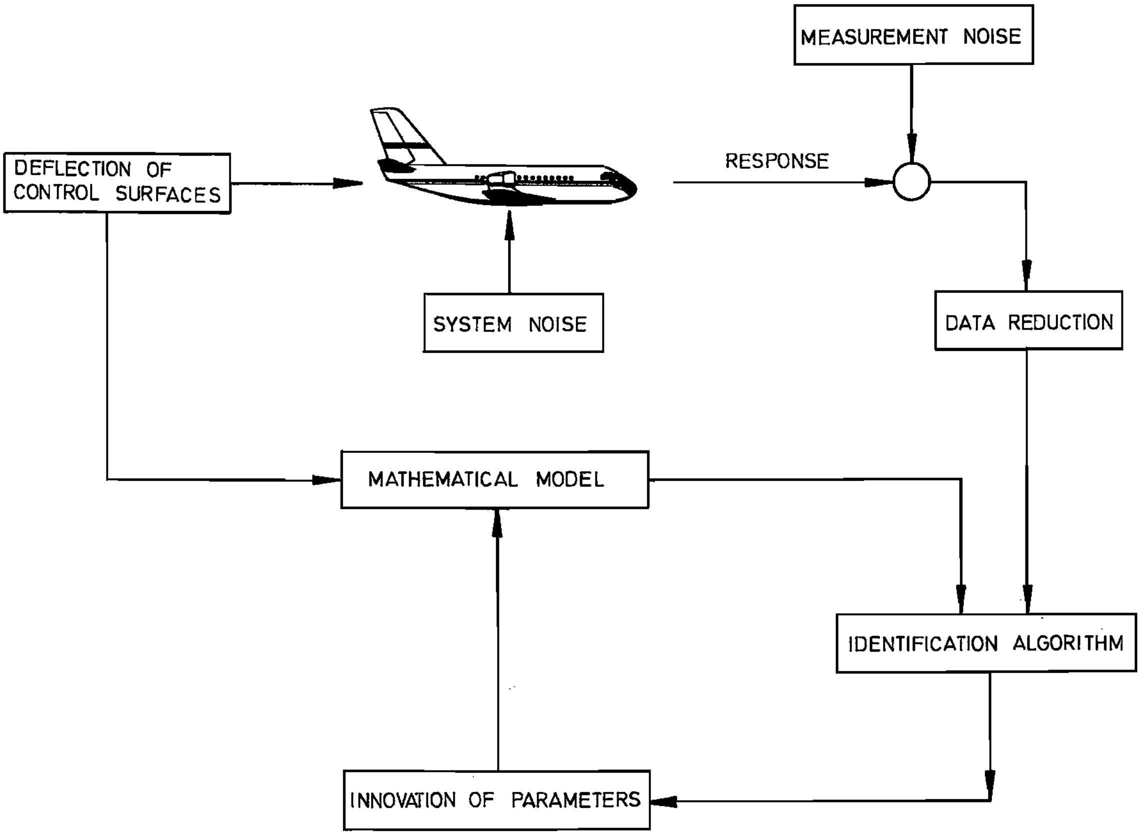

FIGURE 5. FLOAT CHART OF PARAMETERIDENTIFICATION

Recently, the Maximum Likelihood procedure has been used. Details will, however, not be given on this procedure here as references provide an extensive description of it (7).

Based on the good cooperation with the DFVLR (German aerospace test institute) in Braunschweig and Oberpfaffenhofen, we have adopted computer programs which are used with and without Kalman Filters. With these it is possible to eliminate both measurement and system noise.

These programs have been adapted to our purposes. In other words, we have added component load equations to the mathematical model of the movement equations. Parameter identification was successfully achieved on the strength of the component load measurements:

- horizontal tail load
- wing load
- vertical tail load

The following gives an example of the system of initial magnitudes for longitudinal movement; the horizontal tail load is:

$$\begin{bmatrix} a_z \\ \alpha_M \\ q_M \\ L_{HT} \end{bmatrix} = \left[\begin{array}{cc|c} V\cdot Z\alpha & 0 & \\ 1 & 0 & \underline{0} \\ 0 & 1 & \\ g_1\, Z\alpha_H & g_1\, Zy_H & \end{array}\right] \cdot \left[\begin{array}{c} \alpha \\ q \\ \hline w_g \\ \end{array}\right] + \begin{bmatrix} 0 \\ 0 \\ 0 \\ g_1\cdot Z_{\eta H} \end{bmatrix} \cdot \begin{bmatrix} \eta \end{bmatrix} + \begin{bmatrix} W_{az} \\ W_\alpha \\ W_q \\ W_{L_{HT}} \end{bmatrix}$$

The wing is:

$$\begin{bmatrix} a_x \\ a_z \\ \alpha \\ q \\ \vartheta \\ L_W \end{bmatrix} = \left[\begin{array}{cccc|c} X_u & X_\alpha & 0 & 0 & \\ V_\infty\cdot Z_u & V_\infty\cdot Z\alpha & 0 & 0 & \\ 0 & 1 & 0 & 0 & \\ 0 & 0 & 1 & 0 & \underline{0} \\ 0 & 0 & 0 & 1 & \\ & g_1\cdot Z\alpha_F & g_1\cdot Zy_F & 0 & \end{array}\right] \cdot \left[\begin{array}{c} u \\ \alpha \\ q \\ \vartheta \\ \hline u_g \\ w_g \end{array}\right] + \begin{bmatrix} X\eta & 0 \\ V_\infty\cdot Zh & 0 \\ 0 & 0 \\ 0 & 0 \\ 0 & 0 \\ Z_{\eta F} & 0 \end{bmatrix} \cdot \begin{bmatrix} \eta \\ \alpha^2 \end{bmatrix} + \begin{bmatrix} W_{ax} \\ W_{az} \\ W_\alpha \\ W_q \\ W_\vartheta \\ W_{L_W} \end{bmatrix}$$

For lateral movement, taking the vertical tail into account:

$$\begin{bmatrix} a_y \\ \beta \\ p \\ r \\ L_{VT} \end{bmatrix} = \left[\begin{array}{cccc|c} -V_\infty\cdot Y\beta & -V_\infty Y_x & g & 0 & \\ 1 & 0 & 0 & 0 & \\ 0 & 1 & 0 & 0 & \underline{0} \\ 0 & 0 & 1 & 0 & \\ g_3Y\beta_s & g_3Y_{xs} & g_3Y_{zs} & 0 & \end{array}\right] \cdot \left[\begin{array}{c} \beta \\ p \\ r \\ \varphi \\ \hline vg \end{array}\right] + \begin{bmatrix} -V_\infty\cdot Y & -V_\infty\cdot Y & 0 \\ 0 & 0 & 0 \\ 0 & 0 & 0 \\ 0 & 0 & 0 \\ g_3Y_{\xi s} & g_3Y_{\xi s} & 0 \end{bmatrix} \cdot \begin{bmatrix} \zeta \\ \xi \\ 1 \end{bmatrix} + \begin{bmatrix} W_{ay} \\ W_\beta \\ W_p \\ W_r \\ W_{L_{VT}} \end{bmatrix}$$

Figures 6, 7, 8, show the adaptation of flight measurement and calculation. The wind tunnel parameters always served as initial values for the calculation. The result, particularly that of the components, was fairly good.

Generally speaking, it can be said that coefficients applicable to the overall aircraft can also be determined better by including load measurements.

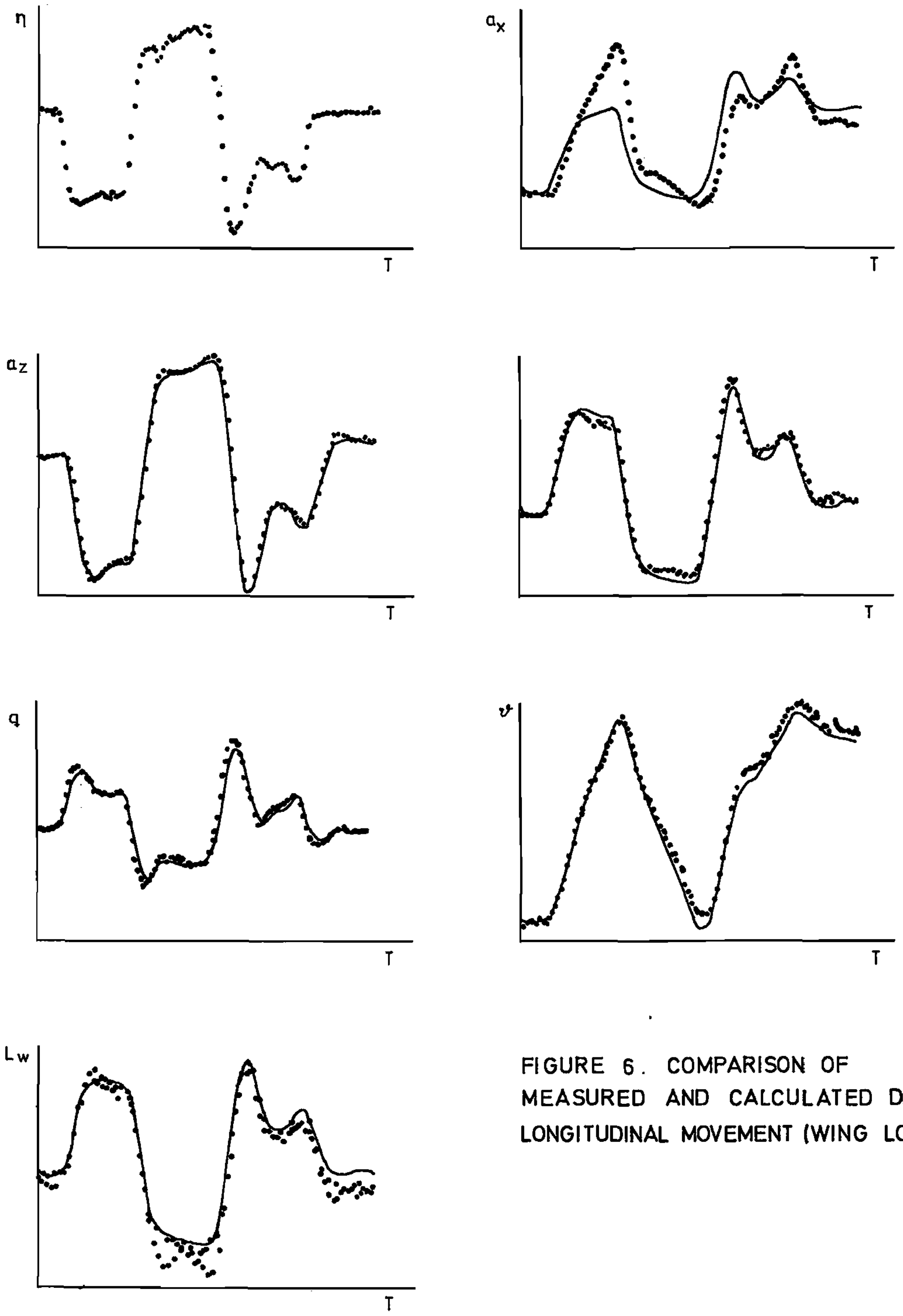

FIGURE 6. COMPARISON OF MEASURED AND CALCULATED DATA LONGITUDINAL MOVEMENT (WING LOAD)

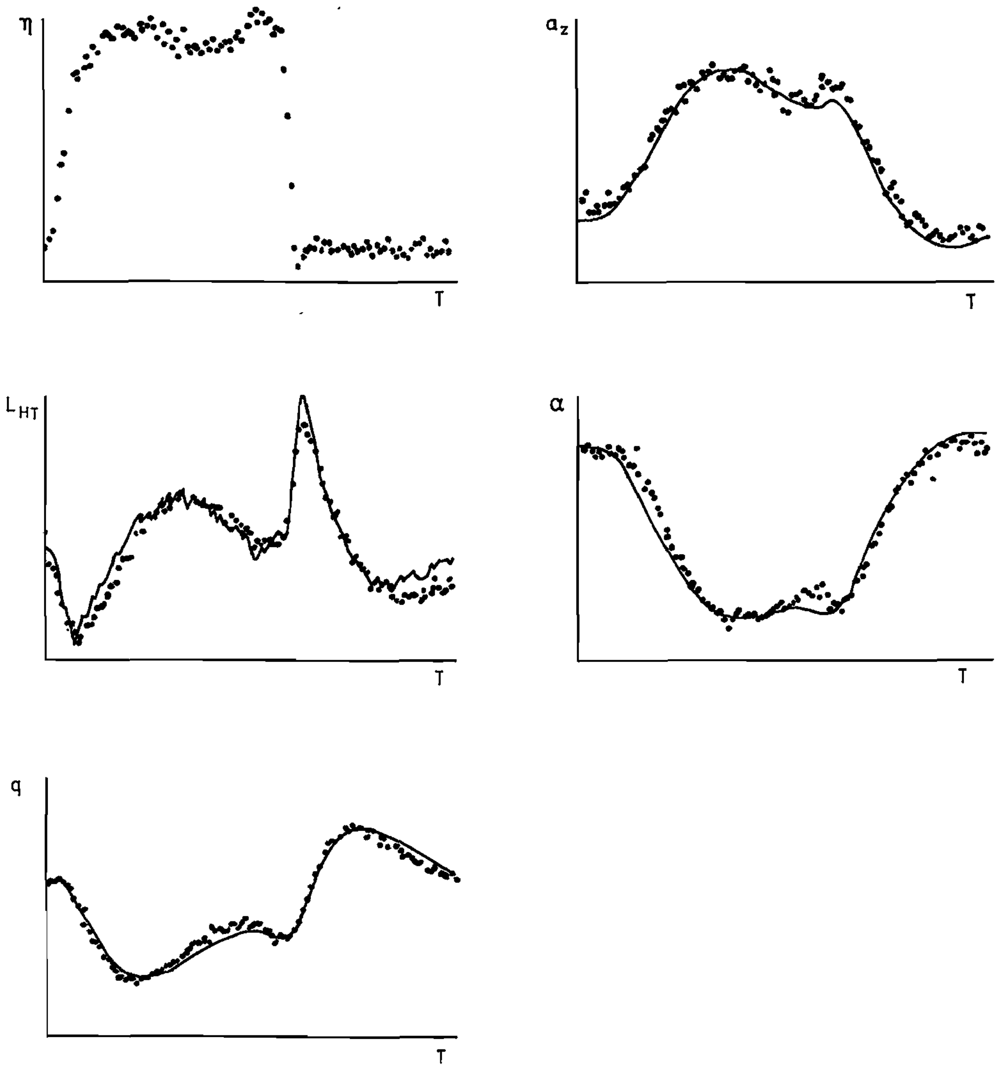

FIGURE 7. COMPARISON OF MEASURED AND CALCULATED DATA LONGITUDINAL MOVEMENT (HORIZONTAL TAIL LOAD)

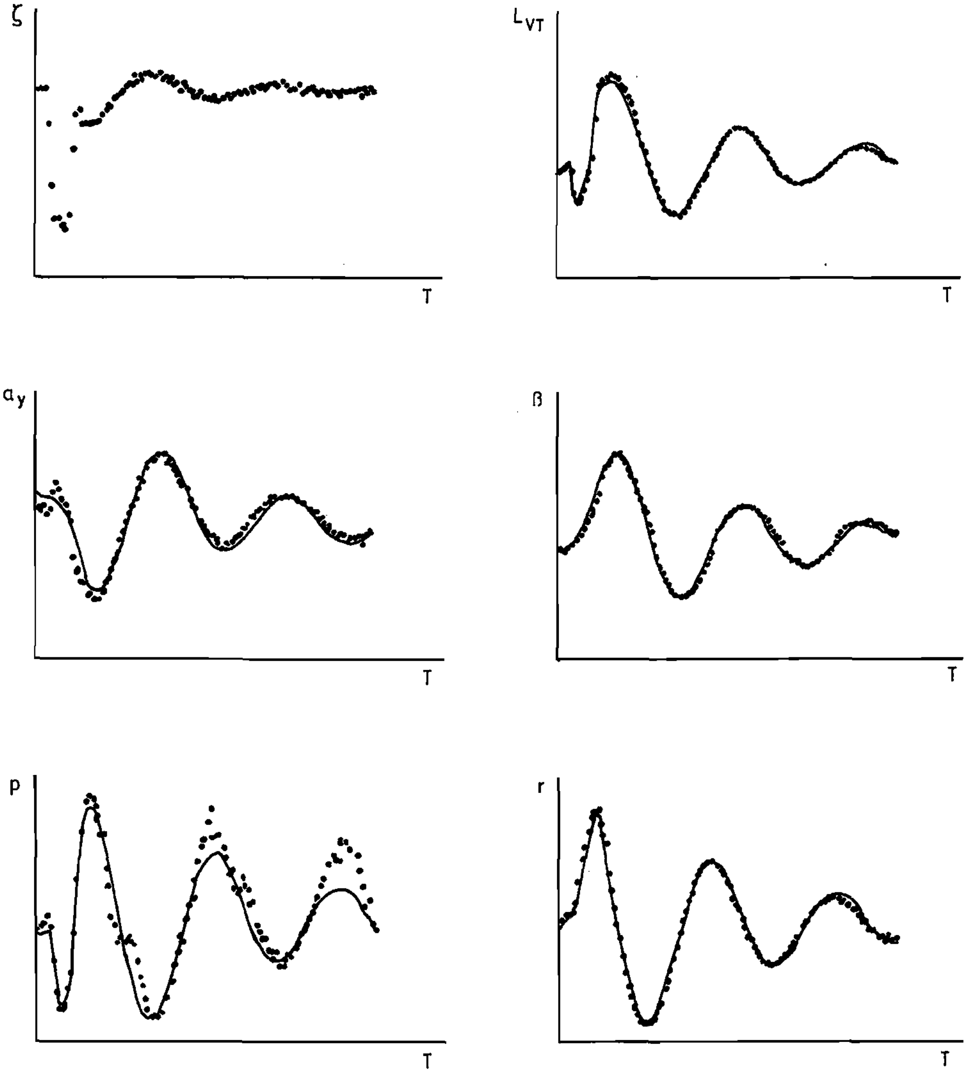

FIGURE 8. COMPARISON OF MEASURED AND CALCULATED DATA LATERAL MOVEMENT (VERTICAL TAIL LOAD)

IV Conclusion

Structural flight load testing is an important part of the certification of an aircraft. This report describes calibration of strain gauges and bridge selection and evaluation of flight parameters, for fatigue and static test problems.

For aerodynamic parameter identification an optimum input is necessary. Maximum Likelihood with Kalman Filter is a well-known, efficient method for parameter identification.

V. References

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