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PHASE II FLIGHT SIMULATOR MATHEMATICAL MODEL AND DATA-PACKAGE, BASED ON FLIGHT TEST AND SIMULATION TECHNIQUES

by

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Abstract

The Dutch Government Civil Aviation School (RLS) operated six Cessna Citation 500 aircraft in the final stage of the training of civil aviation pilot students. In the spring of 1986 RLS decided to purchase a phase II approved flight simulator to transfer parts of the training from flight to the ground. As a result of this only three aircraft would be necessary for actual flight training. However, because the aircraft was developed in the late sixties no mathematical model and data package were available with the required accuracy for a phase II flight simulator. Therefore RLS contracted the National Aerospace Laboratory (NLR) and the Faculty of Aerospace Engineering of Delft University of Technology (DUT) to install an accurate instrumentation system in one of the RLS Cessna Citation 500 aircraft, to execute a flight test programme and to process and analyse the resulting flight test data in order to generate the required mathematical simulation models and corresponding data.

In order to acquire the necessary information in the relatively short period of time available for the execution of the flight tests and the analysis, use was made intensively of dynamic flight test techniques in relation with computer data processing.

The mathematical models to be identified must give an adequate description of the aerodynamic forces and moments, the engine characteristics, the flight control system and the landing gear characteristics. In order to evaluate and test the generated models, both off-line and on-line (pilot in the loop) simulations were performed on the computer and moving base flight simulator owned and operated by the Stability and Control Group of the Faculty of Aerospace Engineering of DUT. Here also the comparisons were made, the so-called proof of match, between the measured flight test time histories and the computed model responses.

In the paper a survey is presented of the employed instrumentation system, the flight test programme, the data processing and corresponding parameter identification and the synthesis of the various models.

1. Introduction

It is a well-known fact, that flight hours are expensive and in-flight training imposes certain restrictions due to safety considerations. As a consequence already at an early stage tools were developed to shift at least parts of the training to the ground. As a result of the growth of technical skills this tool finally emerged into what is now called the ground based flight simulator. Due to the tremendous advances in computer technology both with respect to memory capacity and computation speed, flight training relies more and more upon this device. However, with the increase of tasks also the demands with respect to fidelity become more severe.

Present flight simulators consist of very complex systems;

1. Digital computers
2. Fully instrumented cockpit, including control loading system
3. Visual system
4. Motion system
5. Model software describing the various mathematical models of the simulated aircraft
6. System software interacting the various systems.

The fidelity of the complete flight simulator not only depends on the fidelity of the individual systems, but also on the integration of all parts. In concreto this means, that the flight simulator can be considered as a chain of which all links (systems) must be equally weighted.

The model software represents a very important link in this chain. Furthermore it is the least subjective part, because off-line analysis can be performed using the same real-time simulation software. The characteristics of the vehicle are laid down in various submodels embedded modular within the simulation programme. The most important submodels, related directly to the aircraft, are models with respect to:

1. Aerodynamics
2. Engine
3. Flight control system
4. Landing gear
5. Atmosphere, wind and gust
6. Navigation.

At present most mathematical models of transport aircraft are based on windtunnel measurements, theoretical analysis, data of engine manufacturers, updated with flight test results.

In the course of time the FAA has established various requirements for correlating simulator aerodynamic responses to airplane data. These requirements are legislated through Advisory Circular AC 120-40A (Ref. 1).

In the early days of simulator evaluations, an FAA pilot would subjectively evaluate a simulator by flying it and give his comments on the handling characteristics. Because each pilot will interpret these characteristics to his own standard, tuning may lead easily to very different flight simulators for the same type of aircraft.

Therefore in order to define more objective criteria, in 1980 the Advanced Simulation Plan was established as a joint effort of FAA, NASA, ALPA, airlines and simulator manufacturers. This

Fig. 1 Cessna Citation (PH-CTA) in flight

plan offered three major levels of approval defined as Phase I, II and III. If, for example, a flight simulator has fulfilled phase II requirements it is allowed to let pilots receive an airplane type rating without ever flying the aircraft.

Apart from the requirements for the various levels of approval the FAA also specifies a so-called Acceptance Test Guide (ATG), containing all kinds of tests with respect to aerodynamics, engines, systems and ground handling. This document is drafted by the simulator manufacturer in cooperation with the operator as a guarantee to the flight simulator customer. Parts of this ATG (so-called Proof of Match (POM) data) are evaluated by the FAA to award a certain level of approval.

The Cessna Citation 500 executive jet aircraft is operated by the Dutch Government Civil Aviation School (RLS) in the final stage of civil aviation pilot training.

In the spring of 1986 the RLS decided to purchase a flight simulator for the aircraft, which should have a Phase II approval. This made it possible to reduce the fleet from six to three aircraft.

Because the Citation 500 was developed in the late sixties no mathematical model and data package was available, which was of such quality that it could be used to obtain Phase II approval. Therefore RLS contracted NLR and DUT to install an accurate instrumentation system in one of the Citation 500 aircraft (PH-CTA) (Fig. 1), execute a flight test programme, analyse the data, evaluate the a priori mathematical models of the aerodynamics, engine, flight control system and landing gear and finally generate the

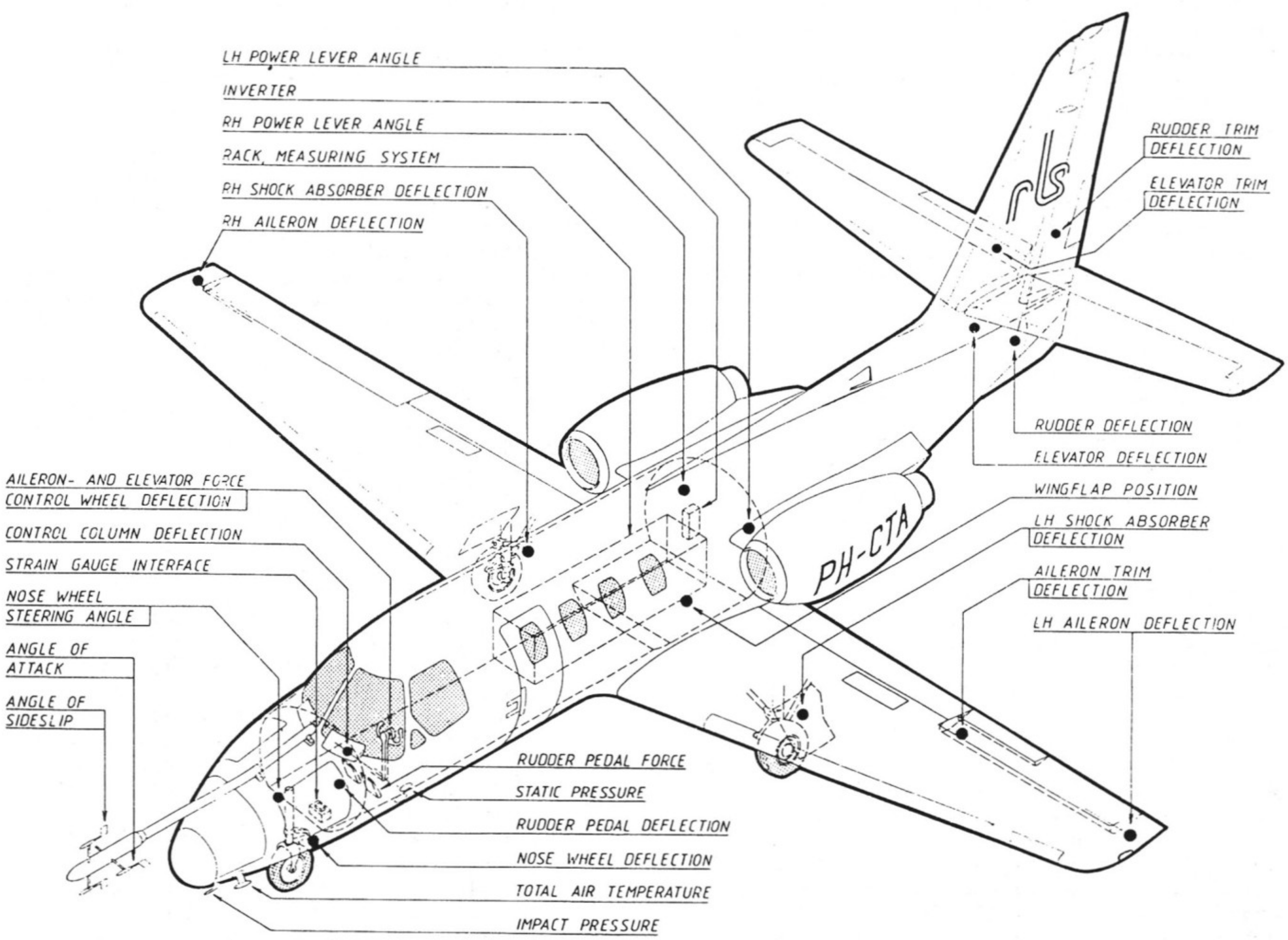

Fig. 2 Sensors in test aircraft

necessary data for these models, based on the results of the flight tests.

Before the flight test programme a priori models were developed based on literature (Ref. 2), comparable aircraft and engineering judgement.

In the paper the instrumentation system is discussed in section 2. Section 3 contains a short survey of the executed flight test programme, whereas section 4 yields the description of the data processing. In section 5 as an example the data analysis and modelling is described for the aerodynamics valid for the normal flight envelope. Finally in section 6 results are shown of some proof of match recordings.

2. Instrumentation

The instrumentation system required for flight tests incorporating dynamic manoeuvres must be more accurate than usually is employed with conventional methods. However, todays commercially available instrumentation systems can fulfill these requirements.
Five more or less independent sensor systems can be distinguished:

1. Inertial measurement system
2. Air-data measurement system and vanes to measure the aerodynamic angles
3. Transducers for measuring engine parameters
4. Transducers for measuring control surfaces and trim deflections
5. Transducers for the measurement of control forces and deflections and gear parameters such as shock absorber deflections and nosewheel steering angle.

Figure 2 shows the positions of the various sensors in the aircraft.

The inertial measurements are performed with a strapped-down Honeywell laser-gyro Inertial Reference System (IRS). For the measurements of static pressure and impact pressure the standard aircraft pitot tube and static sources have been used. However, accurate values of these pressures were obtained by very accurate Garrett barometric transducers, that were held at a constant temperature.

With respect to the engine parameter system, where possible standard on-board systems and transducers were employed. Also for the measurement of fuel flow, fuel quantity, stick shaker, radar altitude and events like gear up/down and speedbrakes retracted/extended use was made of existing instrumentation in the aircraft.

The transducers required to measure the control surface and trim tab positions and the shock absorber deflections were positioned as close as possible to the parameter to be measured.

As can be noted from figure 2 the angle of attack and slip angle vanes were mounted on a special designed boom, which was placed on top of the nose compartment. This design was chosen from a construction point of view and to attenuate the vane position errors.

Apart from the above-mentioned instrumentation three accelerometers were installed in the cockpit in order to measure vibration and buffet levels at the flight deck.

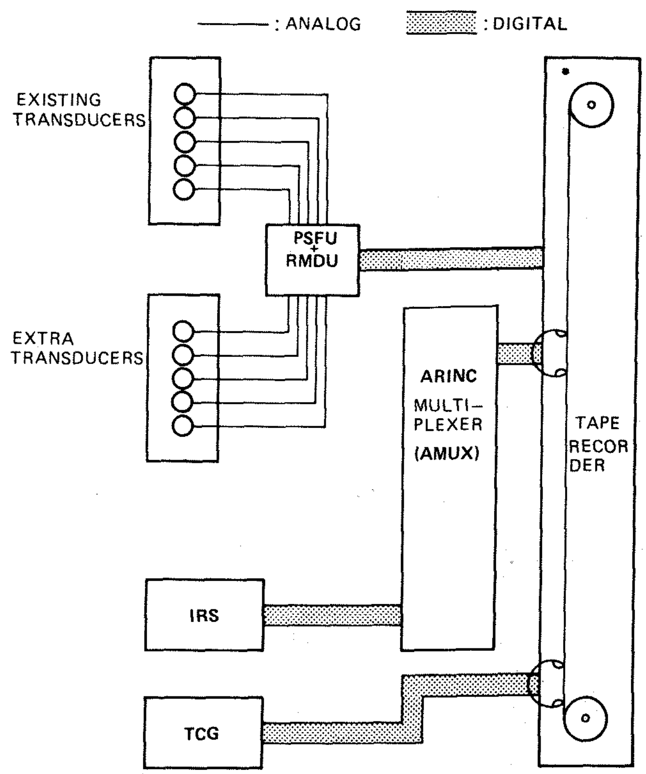

Fig. 3 Instrumentation system

The analog transducer outputs were digitized by means of a Pre-Sample Filter Unit (PSFU) to correct for aliasing and a standard NLR Remote Multiplexer Digitizer Unit (RMDU). The IRS output is converted to ARINC format by means of an ARINC multiplexer (AMUX) and just as the RMDU signals stored on a 14 track taperecorder. A reference time signal from a time code generator also is recorded on a separate track. This signal is used to synchronize all data on the various tracks to the same time grid, which is a requisite for the recording of dynamic manoeuvres. In figure 3 a schematic view is presented of the recording process.

Different sampling rates are employed for the various parameters dependent on the frequency contents. As a result of this, parameters were recorded with a sample rate varying from 2 Hz to 50 Hz, the accelerometers in the cockpit were sampled with 256 Hz.

Figure 4 shows the IRS unit, signal conditioning, operating panel and data recorder mounted in special racks in the aircraft.

Before the start of the actual flight test programme, an instrumentation checkout flight was performed in which the various system were tested and during which a number of manoeuvres were executed. During this test flight and also during the flight test programme the instrumentation functioned without significant problems. At a regular basis calibration checks were performed in order to safeguard the functioning of the instrumentation system. The smooth operation contributed significantly to the ability to execute the flight tests within the tight time schedule.

The flight test programme was executed in the period between January 28 until April 4, 1987. In total approximately 52 hours were flown.

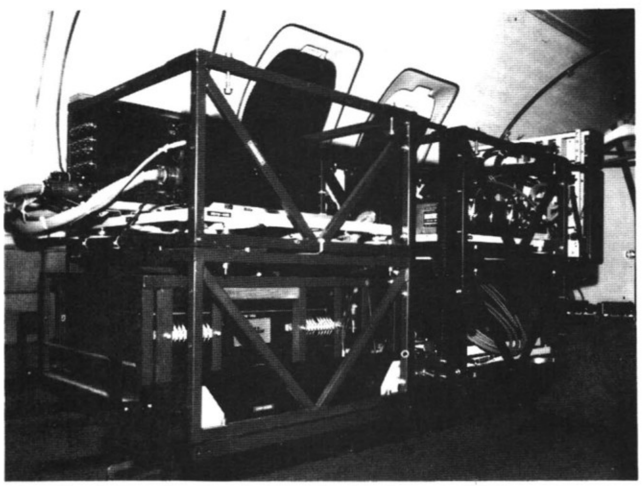

Fig. 4 Instrumentation in test aircraft

Finally it must be mentioned, that two flights were devoted to the determination of the position error correction as a function of configuration setting, speed and altitude. The corrections were determined by comparison of data obtained from the NLR Metro research aircraft and data obtained from the Cessna Citation aircraft. The measurements were executed during formation flights of the two aircraft. Hereby the Metro functioned as a pacer equipped with a tail cone for static pressure and a special pitot tube for impact pressure measurements.

3. Flight Test Programme

The flight test programme was drafted with two different objectives in mind:

A. Test flights to obtain data for the evaluation of the mathematical flight simulation model.
B. Test flights in order to fulfill a number of requirements of the simulator manufacturer and the FAA. These are the already mentioned ATG and POM requirements.

Five topics had to be covered by the flight test programme with respect to the mathematical modelling viz.:

1. Aerodynamics
2. Engine dynamics
3. Flight control system
4. Aircraft performance and handling on the ground
5. Flight deck cues, such as the levels of sound, vibration and buffeting present in certain conditions.

Obviously, a comprehensive programme would be necessary to acquire the data for the modelling of the topics mentioned under label 1 to 4. The only way to perform this challenging task successfully within the limited time available to execute the flight tests, was the ample use of dynamic flight test techniques in combination with a high accuracy instrumentation system. These new techniques comprising measurements in quasi-steady and non-stationary flight (NSM) have been developed by DUT and NLR to reduce the valuable test time while maintaining the same fidelity of the results (Ref. 3).

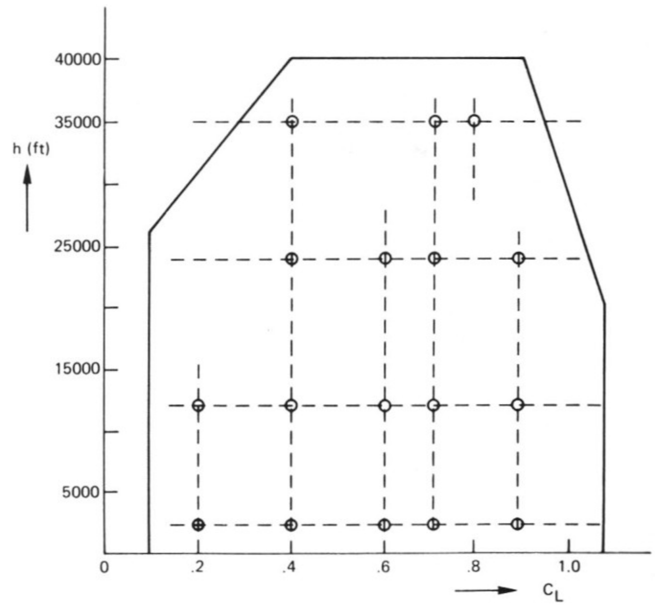

Fig. 5 Testpoints within flight envelope

Taking into account the configuration considered a grid of altitudes and speeds was placed upon the flight envelope of interest (Ref. 4). This is schematically shown in figure 5. This resulted in a set of testpoints, labelled by a particular configuration, centre of mass, altitude and lift coefficient, at which a train of specific manoeuvres was executed. This train of manoeuvres tailored to the various objectives (performance, stability and control, FCS, etc.) is described hereafter. The sequence of the manoeuvres lasted not more than 13 minutes per testpoint. Per altitude a maximum of 5 testpoints was selected at more or less equal angle of attack interspaces.

The tests for the aerodynamic modelling can be split up into three parts. Measurements are required with respect to:

1. Performance model
2. Stability and control model
3. Stall, ground effect and buffet model.

As already has been mentioned the train of manoeuvres contains manoeuvres useful for the determination of the performance model. This performance model can be split up in a symmetric and asymmetric part. The latter comprises mainly steady flight conditions in which a sideslip angle is present. The following "manoeuvres" can be distinguished:

1. Quasi-steady rectilinear horizontal reference conditions of the test points and the angle of attack excursions of the dynamic manoeuvres, that are initiated from these conditions.

2. Quasi-stationary flight conditions, during which for each axis separately the appropriate trim tab deflection slowly is increased and decreased. At the same time the steady reference condition in maintained by means of the corresponding elevator, aileron or rudder deflection. The manoeuvre lasts as long as the control forces are considered acceptable to the pilot. In fact here an exchange between trim tab and control effectivity takes place.

3. Asymmetric quasi-stationary manoeuvres. These are nominally rectilinear sideslipping flights with "relatively slowly" varying slip angle, roll angle and heading. Positive and negative slip angle excursions are required.

4. Quasi-steady wind-up turn manoeuvre. In nominally horizontal flight and constant airspeed the roll angle is slowly increased and decreased to approximately 60°. Both left and right turns are executed.

When both the longitudinal and lateral performance models are available, the "performance" envelope of the aircraft has been covered. This means, that each steady state condition within the flight envelope, characterized by configuration, centre of mass, speed and altitude can be computed including the required angle of attack, sideslip angle, thrust-setting and trim deflections for moment control. However, in this model no terms are present yet, describing the aerodynamic effects, when deviations are made from this performance model. In particular these deviations determine the flying qualitites of the aircraft.

Therefore it is necessary to add terms in the longitudinal and lateral performance models, so that the stability and control characteristics are described accurately for the flight envelope of interest. This results in the addition of coefficients to the force and moment coefficients. The force models in this aspect are less critical than the moment models, because it is a well-known fact, that these determine to a large extent the flying qualitites. In order to be able to evaluate this part of the mathematical model the following manoeuvres were selected.

1. Symmetric non-stationary manoeuvres. Hereby the aircraft was excited manually by means of rectangular shaped elevator doublets, varying in amplitude and time.
2. Asymmetric non-stationary manoeuvres. Also here rectangular shaped aileron or rudder doublets, manually applied, were used, varying in amplitude and time.

The arguments to use these type of inputs rather than more "optimal" inputs were of a practical nature. Firstly the aircraft has no automatic flight control system, which could be used to implement these optimal signals. Furthermore because of the tight time schedule it would not be possible to tailor these signals throughout the flight envelope. Also budget constraints excluded the installation of special equipment, which could artificially manipulate the controls. Because the inputs had to be performed manually and several pilots participated in the flight test programme, it was decided, that only a rather simple input signal was suitable. The block type input was chosen because it is capable of exciting the aircraft over a rather large frequency range.

Ground effect measurements were executed for three configuration settings with landing gear down. For a number of preselected heights varying from 2 to 10 m above the ground rectilinear flights were executed at constant airspeed and height. During the run small excitations were evoked by means of elevator, aileron and rudder. Both the steady parts of the runs and the excitations can be used to evaluate the ground effect.

Both static and dynamic effects can be represented as increments models e.g. as a function of height above the ground superimposed on the coefficients valid for free flight. In the flight test programme also landings were included in which the final parts of the landing could be used for the evaluation of the ground effect. Therefore the aircraft was landed hands-off (as far as possible) at a number of configuration settings and at various constant sink rates.

Stalls were performed for four different configurations at approximately 12000 ft using different entry techniques.

Because it is not a Phase II requirement, no specific tests were planned to determine buffet phenomena. However, during stall and manoeuvres, in particular during some of the elevator doublets with large amplitude, these effects were encountered and logged on the test cards.

One test flight was dedicated to special tests with respect to the engine dynamic responses. The tests included throttle chops at several altitudes and speeds, throttle slams from idle to maximum continuous, small throttle steps, in-flight engine shut-downs and starts at several points in the flight envelope. Finally also constant power ratings were recorded on the ground and video recordings of the engine instruments were made of the engine start-up on the ground.

For the evaluation of the flight control model no specific manoeuvres were planned, because during all manoeuvres performed for aerodynamic modelling also the control forces and control wheel and pedal displacements were recorded. However, on the ground at rest control column and wheel sweeps have been performed. These measurements give information with respect to the dynamics of the control system. The flight recordings mainly provide the necessary information for the determination of the hinge moments.

Besides tests performed in the air also tests were executed on the ground in order to analyse the undercarriage dynamics. Apart from the use of the ground rolls of take-off and landing for this, also special taxi trials have been done incorporating turns at different speeds and turn rates as well as left/right braking exercises. Finally shock absorber deflections were measured during static tests under various mass and fuel distributions.

For the ATG flight test programme, reference 5 was used as a guide from which the flights were grouped and the test cards drafted. Reference 5 specifies a complete set of manoeuvres over the flight envelope of interest, including how the manoeuvre has to be performed, the conditions and configurations for which the test is required and finally the parameters that have to be recorded.

With the aid of this information the various tests were grouped as optimal as possible within a number of ATG flights. In principle the instrumentation system allowed the recording of parameters from engine start-up until engine shutdown. By this it was possible to perform take-offs, in-flight tests, landings and taxi trials in one test flight lasting no more than approximately two hours.

4. Data Processing

The procedure of the data processing is schematically indicated in figure 6.

After the test flight the tape and additional documentation is transported to the data processing facility of NLR (DVSV). If necessary quick-look plots can be generated within less than twelve hours after the flight, so that it is possible to check on a short term basis if all instrumentation systems have functioned correctly. At DVSV the data is converted to physical equivalents of the measured signals, including the calibrations required. Furthermore additional parameters are computed based on physical relationships between measured variables.

After this all parameters as a function of time are sent to a database in the central computer (Control Data Cyber 170-855) of NLR. This database forms the backbone of the NLR Post-processing Process (NPP), which is a part of the NLR Measurement, Recording and Processing System (MRVS).

In the dataprocessing it is necessary to know the instantaneous gross weight, centre of mass position and the inertias during a particular test flight. In order to compute these parameters as a function of time the recorded fuel quantities and recording times are required. Therefore these data are taken from the database and handed to the flight test department. Here the required parameters are calculated as a function of time, based on the mass properties model, that was developed. Then these corrected auxiliary data are included in the so-called Central Data Base (CDB), which also contains the data with respect to instrumentation including PEC. In the CDB also physical constants and geometric aircraft data are stored.

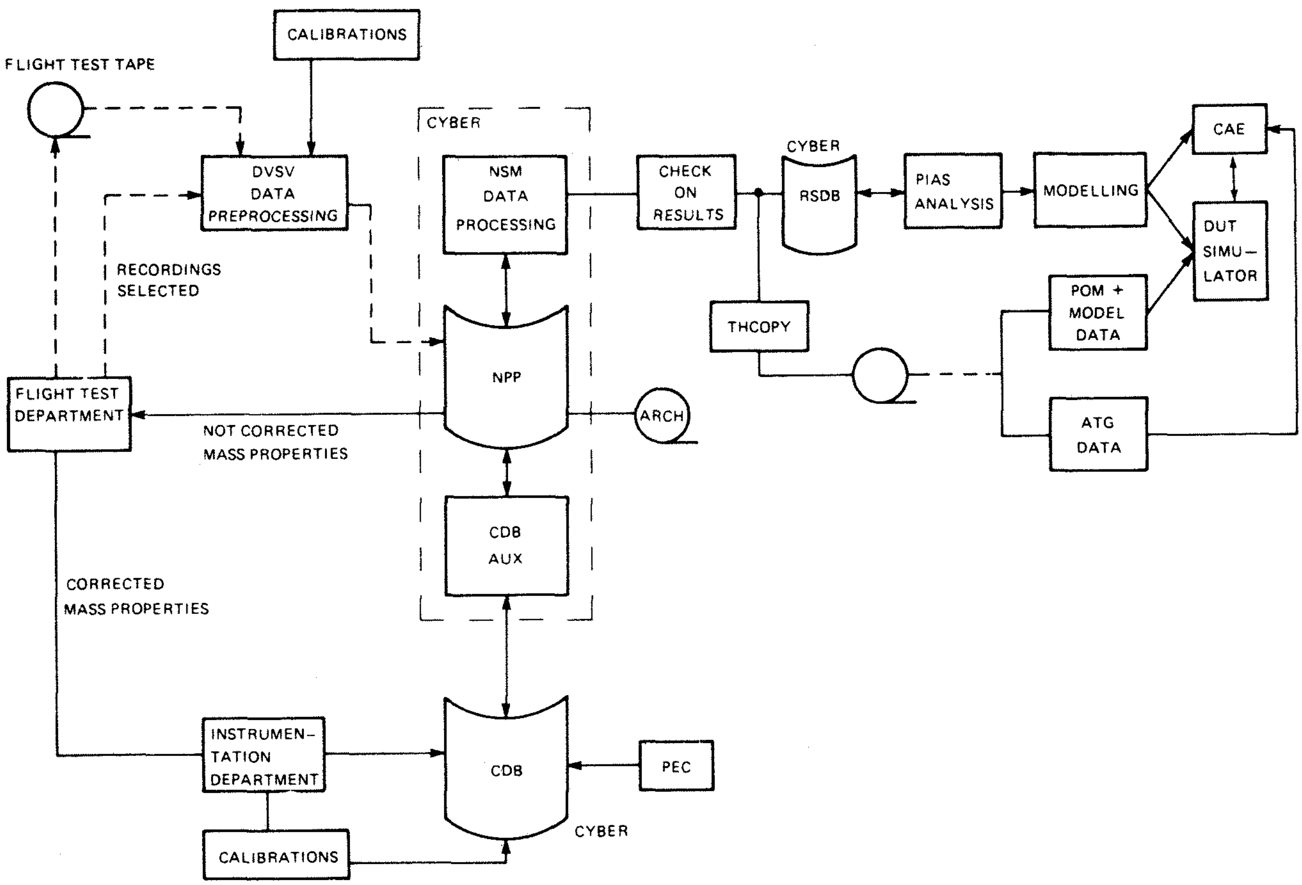

Fig. 6 Data processing of Citation flight test programme

The information in the CDB can be transmitted to the NPP database through a procedure called CDBAUX. Because in the CDB data can be present applicable to more than one aircraft and instrumentation systems this procedure is attached uniquely to a particular aircraft, flight and recording.

At this point all required information is available to start up the so-called NSM data processing sequence. In the NSM data processing four important submodels can be distinguished:

1. COR
2. FPR
3. CAP
4. PPM.

In module CORrection the following actions are performed:

1. All data are accurately synchronized to the same time grid.
2. Because motion about the centre of mass is considered, transformations are performed to the actual centre of mass for measurements obtained from vanes, pressure transducers and accelerometers, which are situated at various places in the aircraft.
3. The pressure measurements are corrected for time lags in the tubes.
4. The position error correction (PEC) is applied.
5. Sensor calibrations are applied to the vanes.

Per recording in module FPR (Flight Path Reconstruction), the trajectory of the mass centre through the air during steady, quasi-steady and unsteady motion is reconstructed. This process is based upon the use of the flight test measurements, both inertial and with respect to the local atmosphere (air data system), and the kinematic equations of motion governing the rigid body modes.

For steady conditions the determination of the flight path is rather simple and straightforward, because a number of time dependent variables are eliminated in the equations of motion. However, in quasi-steady and unsteady conditions the flight-path reconstruction is more complicated.

The variable which is crucial in this respect is the angle of attack, which can be measured accurately in steady conditions by means of a vane. Due to upwash of the flow, rotations of the aircraft, elastic deformations and the dynamics of the vane itself, this method is not suitable for the other conditions. Consequently an alternative manner had to be found.

The flight path of an aircraft can be described mathematically by the force equations of motion with respect to an inertial frame. Only three equations describing the translations of the mass centre are required.

Expressed in body axes the force equation can be represented in matrix form as:

$$[\vec{a}]^B_{mc} = \frac{[\vec{f}]^B}{m} = [\vec{A}]^B_{mc} - [\vec{g}]^B \qquad (4.1)$$

in which:

$[\vec{a}]^B_{mc}$: specific forces as measured by accelerometers attached to the body axes frame.

$[\vec{f}]^B_B$: aerodynamic and propulsive forces expressed in body axes.

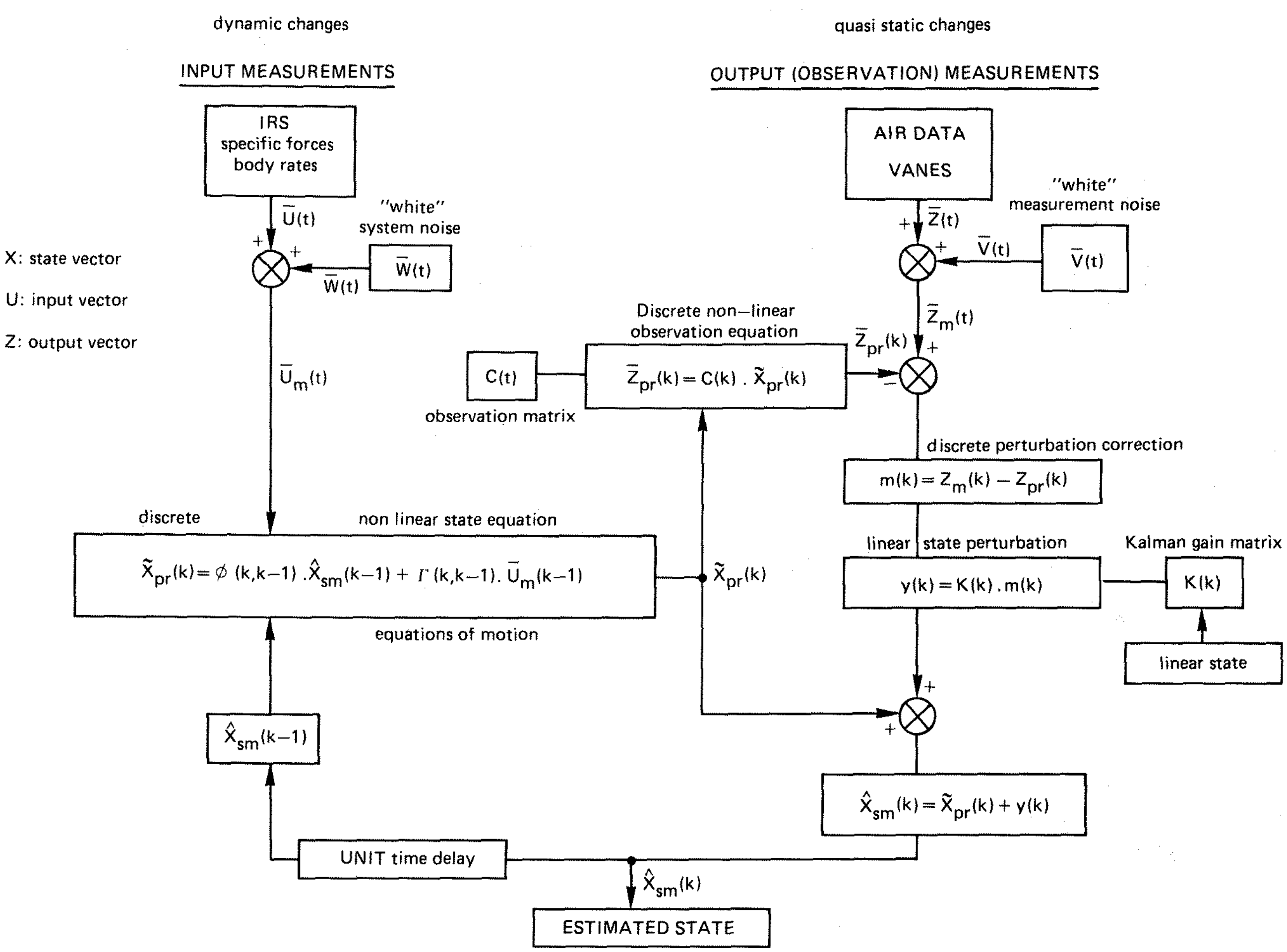

Fig. 7 Kalman filtering in NLR fligh test data processing

$[\vec{A}]^B_{mc}$: kinematical acceleration vector of the mass centre.

$[\vec{g}]^B$: acceleration vector due to gravity along the body axes.

$[\vec{A}]^B_{mc}$ consists of the kinematical accelerations along the body axes.
Expression (4.1) represents a non-linear set of differential equations with respect to time, which in principle can be integrated when either the specific forces and angular velocities or the aerodynamic and propulsive forces are known.
The flight path reconstruction in the data processing in fact is based on the integration of (4.1) (state) employing the measured time histories of the specific forces, angular rates, airspeed and the height. A schematic view is presented in figure 7. The specific forces and angular rates directly result from the inertial measurement unit (input measurements), whereas the air velocity and height increment or decrement can be derived from the air data measurement system (output measurements).

However, the measurements are corrupted with errors and the exact initial conditions necessary for the start of the integration are not known. Therefore, statistical procedures are required to attenuate the effects of these errors on the time histories of the state variables. In the data processing of the flight tests the well-known Kalman filtering and smoothing technique is employed (Ref. 6).

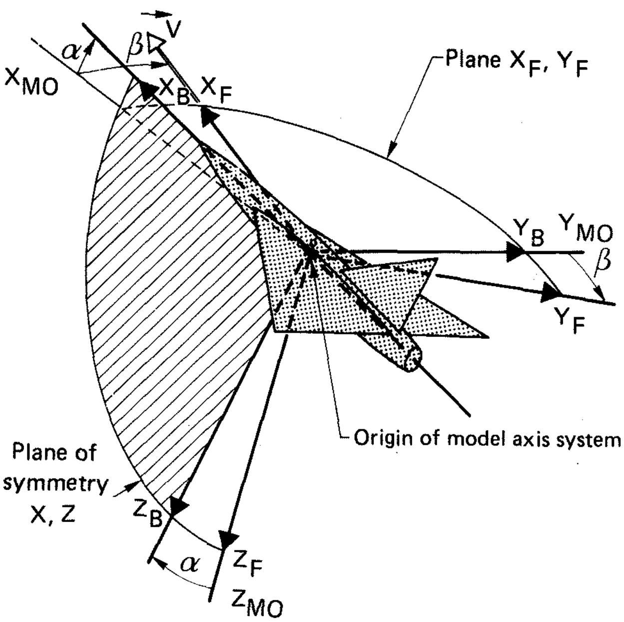

Fig. 8 Model axes reference frame

In the flight path reconstruction the most important quantities to be reconstructed are the air velocities u, v and w along the body axes and the Euler angles Θ, φ and ψ. Actual distances are not important in this respect. From the reconstructed velocities the angle of attack and side slip angle can be derived.

As a result of module FPR smooth time histories are obtained of the state variables, which are included in the NPP database.

In module CAP (Calculation of Aerodynamic Parameters) results of COR and FPR are used to compute additional parameters, which are necessary for the identification of the mathematical models. Amongst these are the dynamic pressure, Mach number and time derivatives of angle of attack and slip angle. Furthermore dimensionless specific forces and moment coefficients are computed. Finally the dimensionless specific forces are transformed to the so-called model axes frame (Fig. 8) and the moment coefficients are reduced to a reference centre of mass position. The results of CAP are added to the NPP database.

In module PPM (Power Plant Model) the static engine thrust is determined from measured engine parameters, such as fan speed and Mach and atmospheric measurements such as static air temperature and static pressure. A performance deck valid for the JT 15D-1 turbo fan was available developed by the engine manufacturer (Pratt & Whitney) based on test data. Both gross thrust and ram drag are computed in this programme as a function of compressor speed an Mach.

Finally the gross thrust and ram drag are also transformed to the model axes frame, so that lift, drag and other dynamic coefficients can be derived.

After this two possibilities for further processing were present. If data had to be processed for POM purposes and/or, for the identification of the FCS and gear models, they were converted to another tape format (THCOPY) suitable for implementation on the computer of DUT. After this the data was written to magnetic tape and transported to DUT. However, if data was required for the evaluation of the aerodynamic and thrust models the data was written to the so-called Result Storage Data Base (RSDB) for further analysis with the module PIAS (Processing of dynamic manoeuvre measurements with an Interactive Adaptive System).

The ATG data, consisting of time histories of selected parameters, were processed in the same way as the modelling data and transported to the simulator manufacturer according to the same procedure as the model/POM data transport to DUT.

When the NSM data processing for a particular recording is finished and the results stored in the RSDB all time histories for the aerodynamic modelling are present, viz.:

1. The aerodynamic coefficients
2. The state variables
3. The control and trim tab angles.

With respect to the FCS and undercarriage modelling, also time histories are available of the control forces and displacements, shock absorber deflections and nosewheel steering angle.

At the PIAS analysis stage the coefficients appearing in a postulated submodel structure, representative for the description of the aerodynamics in the various test points, are estimated by means of an equation error technique.

The regression algorithm is based upon the theory of the solution of linear least squares (Ref. 7).

In this procedure (so-called two step method) during the first step, which is mathematically the most complex part, the state of the aircraft is reconstructed as a function of time expressed in characteristic flight mechanical variables. Hereby the aerodynamics are not used to compute these variables. The flight path reconstruction has to be performed once for a particular recording.

The parameters of the aerodynamic models are estimated in the second step of the procedure, which is mathematically rather straightforward. Thus the separation of the trajectory reconstruction and the parameter identification process makes it possible to select and evaluate models in a flexible way. Hereby use is made to a great extent of both computer plotting and "batch" processing.

Finally in the modelling phase the submodel coefficients as a result of the PIAS analysis of the flown test points are integrated into a complete model. This final model then is converted to tables, suitable for implementation in the simulation programme at DUT. The complete model and corresponding data are also sent to the simulator manufacturer for implementation on the actual Citation 500 flight simulator.

5. Data analysis and modelling

To gain insight in a preliminary stage it was decided to build an a priori model of the Citation 500, based on the available wind tunnel and flight test information, completed with data of comparable aircraft and engineering judgement. At that stage it was not clear, which terms within the models were relevant for the Citation 500. Therefore rather comprehensive models were developed, including all kinds of non-linearities and dependencies on the aircraft state. Also from a software managing point of view it was thought a better philosophy to develop this comprehensive model in advance where time was less restrictive than during the rather short period available for analysis and modelling.

If particular parts of the model appeared to be insignificant or could not be identified from the test data, it would be much more simple to set the corresponding coefficients to zero instead of expanding the model by means of software changes.

The a priori models were implemented on the Gould/Sel computer of the moving base flight simulator operated by the Stability and Control Group of the Department of Aerospace Engineering of DUT. As a result a complete a priori model for the Citation 500 was available preceding the flight test programme.

After integration of the model within the simulation software and off-line testing, it was possible to fly an "a priori" Citation on-line with the pilot in the loop.

Although, in principle, no on-line simulation is required for the models and data, it was considered as a very valuable option both for the a priori, and final models. In case of the a priori model confidence could already be built up with respect to the flyability and/or functioning of the various models. Furthermore pilots can be familiarized more easily with the manoeuvres that are planned in the test flights.

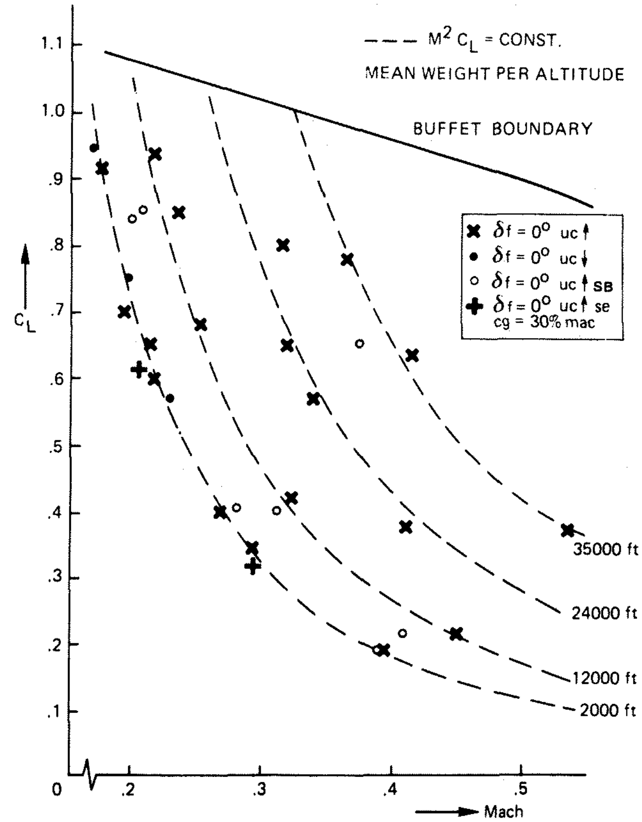

Fig. 9 Testpoints for δ_f=0 as a function of altitude

At the start of the analysis it was decided to concentrate at first on the quasi-stationary horizontal symmetrical reference conditions of the test points and the corresponding dynamic manoeuvres, both for the evaluation of the performance model and the additional stability and control parts. In figure 9 for the zero flap configuration, the test points are indicated in a lift coefficient versus Mach plot. As a reference also plotted herein are $M^2C_L \div W/\delta$ curves at four test altitudes. The weight appearing in this formula is chosen to represent the mean value of the test weights at that altitude.

Applicable to the reference mass centre (30% mac) in figure 9 the test points are indicated for the clean configuration (19 pts), configuration with flaps up and gear down (3 pts) and the configuration with flaps up, gear up and speedbrakes extended (7 pts).

Because in flight the inertial sensors always measure the simultaneous effects of the aerodynamics and thrust, there is always the issue, which effects must be contributed to thrust and which to drag. As already has been mentioned in section 4 for this project it was assumed, that the thrust could be computed with sufficient accuracy by means of a performance deck valid for the Pratt and Whitney turbofan. The input for this curve reading programme are measured parameters of which the fan speed is the most direct variable.

As a result, in the data processing the aerodynamic parts transformed to the model axes system (Fig. 8) can be isolated from the measurements. The aerodynamic model is described in the model axes frame, because this corresponds to the way windtunnel measurements are performed. Partly as a result of this, the aerodynamic models used in simulation programmes are expressed also in this frame.

The static performance thrust model used in the flight test data processing also is implemented in the simulation software. The rationale behind this is, that the simulation process more or less can be considered as the inverse of the NSM data processing (Ref. 8). The integration of the equations of motion incorporating the sum of the outputs of the aerodynamic and thrust model, theoretically must lead to the same flight path as in actual flight. By this construction it was attempted to enhance the correlation between simulation and flight.

For the test points involved a submodel was postulated for the three force and three moment coefficients. These submodels are valid for and around the reference condition and can be described as follows:

1. Longitudinal submodel structure

$$C_{D_{MO}} = C_{D_0} + C_{D_\alpha}\cdot\alpha + C_{D_{\alpha^2}}\cdot\alpha^2 + C_{D_{q_{MO}}}\cdot\frac{q_{MO}\cdot\bar{c}}{V} + C_{D_{\delta_e}}\cdot\delta_e + C_{D_{\beta^2}}\cdot\beta^2$$

$$C_{L_{MO}} = C_{L_0} + C_{L_\alpha}\cdot\alpha + C_{L_{\alpha^2}}\cdot\alpha^2 + C_{L_{q_{MO}}}\cdot\frac{q_{MO}\cdot\bar{c}}{V} + C_{L_{\delta_e}}\cdot\delta_e \qquad (5.1)$$

$$C_{m_{MO}} = C_{m_0} + C_{m_\alpha}\cdot\alpha + C_{m_{\alpha^2}}\cdot\alpha^2 + C_{m_{q_{MO}}}\cdot\frac{q_{MO}\cdot\bar{c}}{V} + C_{m_{\delta_e}}\cdot\delta_e + C_{m_{\beta^2}}\cdot\beta^2 + C_{m_{\delta_r^2}}\cdot\delta_r^2 + C_{m_{\delta_{e_t}}}\cdot\delta_{e_t}$$

2. Lateral submodel structure

$$C_{y_{MO}} = C_{y_o} + C_{y_\beta}\cdot\beta + C_{y_{p_{MO}}}\cdot\frac{p_{MO}\cdot b}{2V} + C_{y_{r_{MO}}}\cdot\frac{r_{MO}\cdot b}{2V} + C_{y_{\delta_r}}\cdot\delta_r$$

$$C_{\ell_{MO}} = C_{\ell_o} + C_{\ell_\beta}\cdot\beta + C_{\ell_{p_{MO}}}\cdot\frac{p_{MO}\cdot b}{2V} + C_{\ell_{r_{MO}}}\cdot\frac{r_{MO}\cdot b}{2V} + C_{\ell_{\delta_r}}\cdot\delta_r + C_{\ell_{\delta_a}}\cdot\delta_a + C_{\ell_{\delta_{a_t}}}\cdot\delta_{a_t} \qquad (5.2)$$

$$C_{n_{MO}} = C_{n_o} + C_{n_\beta}\cdot\beta + C_{n_{p_{MO}}}\cdot\frac{p_{MO}\cdot b}{2V} + C_{n_{r_{MO}}}\cdot\frac{r_{MO}\cdot b}{2V} + C_{n_{\delta_r}}\cdot\delta_r + C_{n_{\delta_r}}\cdot\delta_r + C_{n_{\delta_{r_t}}}\cdot\delta_{r_t}$$

These submodels are valid within the normal flight envelope. Stall phenomena, buffeting, ground effects etc. are not represented by this formulation.

The mathematical model as postulated by (5.1) is based on the type of aircraft, that is investigated and the experience, which parameters can be identified accurately within a chosen model structure.

The Cessna Citation 500, a small executive jet transport, has a straight, tapered wing optimized for the low speed flight regime. The maximum speed in horizontal flight is approximately .62 Mach. Because of the size of the aircraft, the speed regime of interest and the objective of the mathematical model, it was assumed, that the aircraft could be considered as rigid.

The aircraft has a fixed tailplane setting with an elevator used both for trimming and manoeuvring. Control forces existing at a selected steady condition can be eliminated by means of a small trim tab in the right elevator surface. In a steady rectilinear symmetric flight, which usually is present, the lateral coefficients are zero. Therefore the aerodynamic coefficients representing the longitudinal performance model can be described by:

$$C_{D_{MO}} = C_{D_0} + C_{D_\alpha}\cdot\alpha_{tr} + C_{D_{\alpha^2}}\,\alpha_{tr}^2 + C_{D_{\delta_e}}\cdot\delta_{e_{tr}}$$

$$C_{L_{MO}} = C_{L_0} + C_{L_\alpha}\cdot\alpha_{tr} + C_{L_{\alpha^2}}\,\alpha_{tr}^2 + C_{L_{\delta_e}}\cdot\delta_{e_{tr}}$$

$$C_{m_{MO}} = C_{m_0} + C_{m_\alpha}\cdot\alpha_{tr} + C_{m_{\alpha^2}}\,\alpha_{tr}^2 + C_{m_{\delta_e}}\cdot\delta_{e_{tr}} + C_{m_{\delta_{e_t}}}\cdot\delta_{tr} \qquad (5.3)$$

Herein the subscript tr indicates the angle of attack, elevator angle and trim tab angle at the reference condition of a particular test point. It is assumed, that the effects of the trim tab on lift and drag are small enough to be neglected. The above described equation models are written as a linear function of α, α^2, δ_e and δ_{tr}. It is well-known, that the drag coefficient usually can be represented by a quadratic function of angle of attack. Because of the wing shape and flaps employed on the Citation 500 it is expected that the lift and pitching moment coefficient exhibit a linear relation with respect to angle of attack for a large part of the angle of attack range. However, at high angles and higher Mach numbers quadratic terms may be required also.

Frequently windtunnel measurements are executed as a function of angle of attack, whereby the trim and control deflections are equal to zero. If the same is done in (5.3), omitting the $\delta_{e_{tr}}$ and δ_{tr} contributions, this will lead to the same representation.

In this context it must be mentioned, that only the "geometric" contributions of thrust are accounted for in C_D and C_L. Aerodynamic effects due to thrust are assumed to be negligible on C_L and C_D. The rationale behind this is the fact, that the powerplant is a jet engine. Furthermore the position of the engines on the aircraft is such, that no large impact on C_L and C_D can be expected. For the contribution of the thrust on the pitching moment a somewhat different philosophy is followed. Hereby it is assumed, that the geometric and aerodynamic effects are such, that they more or less are balanced. This conclusion was derived from analyses of flight test data in which throttle transients were present. If no aerodynamic effects due to thrust would be present (thus only thrust x arm) manipulations of the throttles would lead to a step response particular noticable in the angle of attack. However, this trend could not be found consistently in the timehistories of the flight test data. The thrust arm affecting the pitching moment is such, that an increase in power results in a nose down moment, However, augmenting thrust leads to a higher nozzle velocity and consequently to the phenomenon of jet entrainment. Hereby the surrounding air is affected by the jet in such a way, that the downwash is enforced at the horizontal tailplane leading to a decrease of tailplane angle of attack and this in a reduction of tailplane lift. This results in a pitch up moment opposite to the nose down geometric moment. Therefore in the mathematical model it is assumed, that the effective thrust arm is equal to zero.

As has been indicated in section 3, 5 à 6 testpoints at an approximately constant altitude were chosen.

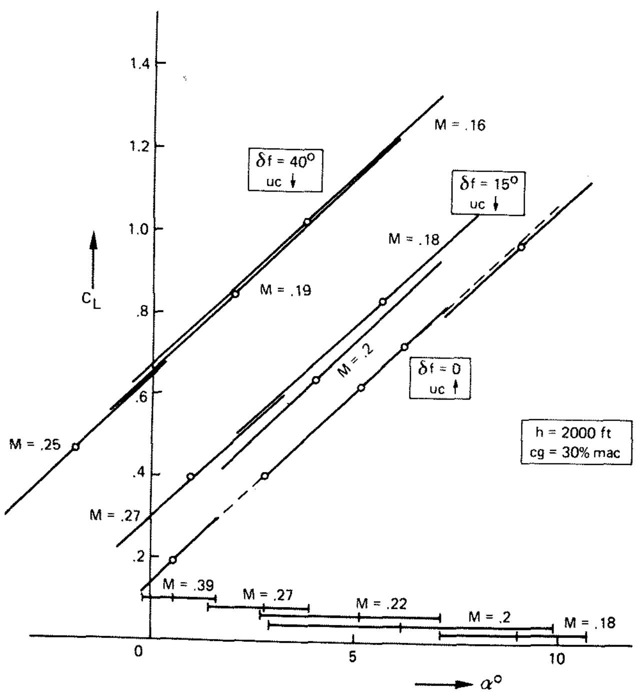

Fig. 10 C_L-α curves following from submodels for various flap settings

As an example in figure 10 it is indicated for C_L how these points are located as a function of angle of attack. Because the testpoints are flown at different airspeeds possible (Mach) speed effects are embedded. In the figure apart from the reference conditions also the α-sweeps are shown as result of the elevator doublets. In accordance with windtunnel presentations the control and trim tab deflections are set equal to zero, just as the dynamic effects due to pitch rate, so that only the angle of attack terms remain, now valid for the angle of attack range, covered by the manoeuvre.

Thus the α-sweeps as shown in figure 10 are described by the following equations and are valid around α_{tr}:

$$C_{D_{MO}} = C_{D_0} + C_{D_\alpha}\cdot\alpha + C_{D_{\alpha^2}}\cdot\alpha^2$$

$$C_{L_{MO}} = C_{L_0} + C_{L_\alpha}\cdot\alpha + C_{L_{\alpha^2}}\cdot\alpha^2 \qquad (5.4)$$

$$C_{m_{MO}} = C_{m_0} + C_{m_\alpha}\cdot\alpha + C_{m_{\alpha^2}}\cdot\alpha^2$$

The integration of the various submodels will be described hereafter. As an example here the lift coefficient is discussed. However, the same philosophy applies for the drag and pitching moment coefficient.

From the equilibrium condition in horizontal flight it follows:

$$W = C_L\cdot q_c\cdot S \approx C_{L_\alpha}(\alpha-\alpha_0)\cdot q_c\cdot S \qquad (5.5)$$

showing, that for constant values of C_{L_α}, α_0 and airplane weight the dynamic pressure is the same at constant angle of attack. In other words for constant EAS the angle of attack is the same at each altitude. Obviously the weight during the testflights varied. As a result of this and the fact, that C_{L_α} and α_0 variations may occur, lines of constant α do not coincide with all testpoints flown at that particular EAS. Now the assumption is made, that the submodel also is able to represent steady reference conditions deviating from the actual testpoint. These deviations, however, must be relatively small, otherwise a reduction to a particular steady state must take place. Then from a plot as given in figure 11 (in the analysis all available submodels have been used) for a number of constant angle of attack lines the lift coefficients can be plotted as a function of Mach. Using least-squares techniques this results in a family of curves of constant α as a function of Mach.

From this set of curves again C_L-α plots can be constructed for a set of constant Mach numbers. As can be noticed from figure 11 because of the low subsonic range the variation due to Mach is relatively small.

In this way plots are obtained, which are comparable to windtunnel results. Because here, in contrast to flight, Mach and angle of attack are not coupled kinematically to each other, measurements can be performed in which the Mach number remains constant during α-variations directly.

In the same way as described here the submodels for the drag and pitching moment coefficients are integrated.

For all configurations of interest in the same way as described the coefficients in the submodels are determined by means of regression analysis and integrated to $C_L(M)$-α, $C_D(M)$-α and $C_m(M)$-α plots. In total six sets are constructed for the flight envelope of interest, viz.:

1. clean uc↑
2. clean uc↑ speedbrakes extended
3. clean uc↓
4. 15° flaps uc↑
5. 15° flaps uc↓
6. 40° flaps uc↓

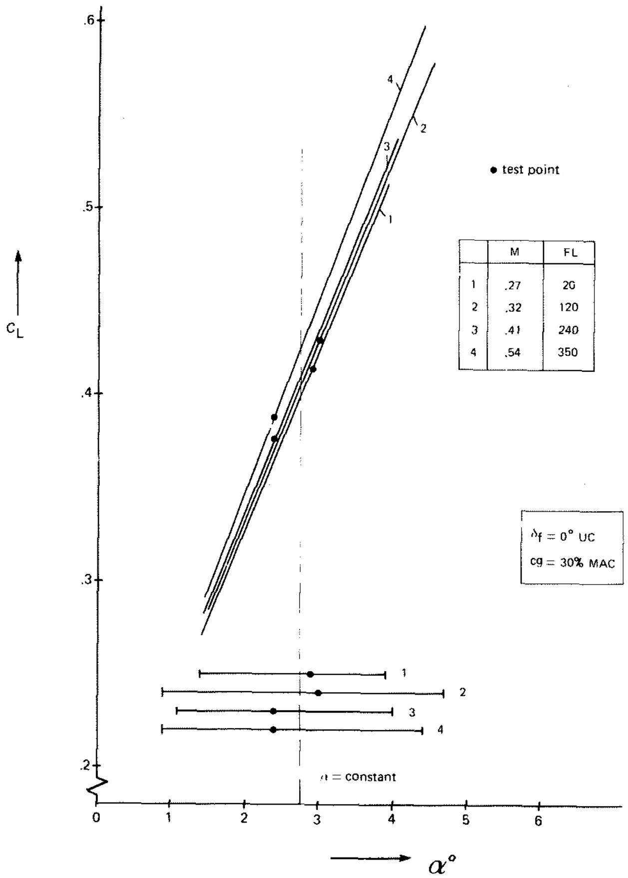

Fig. 11 C_L-α submodels as function of altitude

Set 1 (clean, uc↑) was considered as a base model, which consistently was subtracted from the other sets. By this, effects due to flap manipulations, gear extension/retraction and speedbrake operation could be modelled as incremental contributions, which are superimposed on the base models of C_L, C_D and C_m.

The base model and increment families of curves are modelled as a function of angle of attack. This would allow a rather easy integration of the stall and ground effect submodels.

Finally the models are digitized and converted to tables. It appeared, that the structure of the a priori model was such, that these tables directly could be implemented in the software, employing table look-up techniques.

The coefficients in the C_L, C_D and C_m submodels, that effect stability and control primarily are the terms with respect to angle of attack and elevator. However, also the pitch rate coefficients are very important. Frequently also coefficients are modelled, that represent the non-stationary behaviour of the flow over the wings, which in stability and control considerations usually is represented by the quasi-steady $C_{L_{\dot\alpha}}$, $C_{D_{\dot\alpha}}$ and $C_{m_{\dot\alpha}}$ coefficients of which the latter is the most relevant.
Because in the identification process use is made of regression techniques and the fact, that a strong correlation exists between pitch rate and $\dot\alpha$ coefficients, an accurate identification of both parameters directly is very difficult. For this project it was decided to identify the effect of $\dot\alpha$ and q_B simultaneously in one pitch rate coefficient, so that the $\dot\alpha$ effects, although in principle quite different from nature than the pitch rate effects, are embedded. It is well known, that both pitch rate and $\dot\alpha$ have a large impact in the damping (for which the sum of C_{m_q} and $C_{m_{\dot\alpha}}$ is required) and frequency of the short period mode. In particular this applies to the C_{m_q} and C_{L_q} coefficients.

Also for these parameters, as a result of the data processing, a large set of data was obtained valid for the test points. In accordance with the performance model and for simulation purposes these coefficients were formulated as a function of angle of attack. Hereby again the clean configuration was considered as a base model on which effects due to configurations changes were superimposed. Then the plots of these models were digitized to numerical tables and implemented in the existing aeromodel structure. Also for these coefficients only minor modifications were required in the structure of the a priori aerodynamic model.

In contrast to the longitudinal model no "performance" model was considered for the lateral aerodynamic model. As a result of this only flight test data were analysed yielding responses to aileron and rudder doublets.

As can be noticed from (5.2) the aerodynamic coefficients are written as a linear function of sideslip, dimensionless roll and yaw rate and the control deflections. This formulation frequently is postulated, when describing the lateral aerodynamics about a steady state reference condition.

It is assumed, that $\dot\beta$-effects are embedded in the yaw rate coefficients for the same reasons as have been mentioned for the $\dot\alpha$-pitch rate terms.

Furthermore it is assumed, that thrust has no effects on the aerodynamic coefficients and no "geometric" corrections as for the longitudinal model are necessary. This implies, that time histories from the flight path reconstruction directly can be used for identification of the coefficients appearing in the submodels. This was done in the same way as for the longitudinal model by means of regression analyses for each test point indicated in figure 9. After this the resulting data were integrated into a complete model valid for the flight envelope of interest. Hereby also for the complete model the linear structure, as postulated for the submodel, was maintained. However, the aerodynamic parameters are modelled in the same way as has been done for the longitudinal model viz. as a function of angle of attack, incorporating effects due to Mach and configuration changes.

The implementation of the complete lateral model within the a priori model structure did not give rise to any difficulties. In fact it appeared that the final model was more simple than adopted in the a priori model.

6. POM Results

As already has been indicated in section 4 the ATG and POM flight test data have been processed in the same way as the test data for the mathematical modelling. After the NSM data processing, the ATG data were not further processed

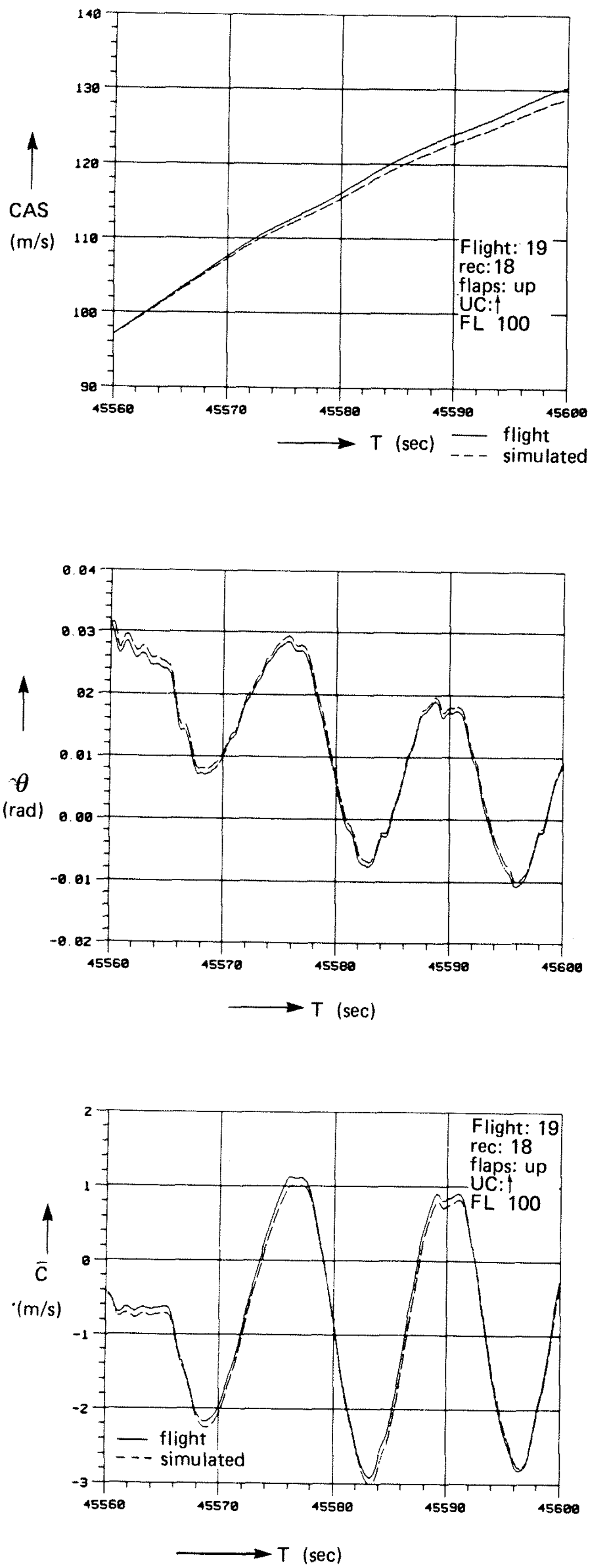

Fig. 12 Level flight acceleration

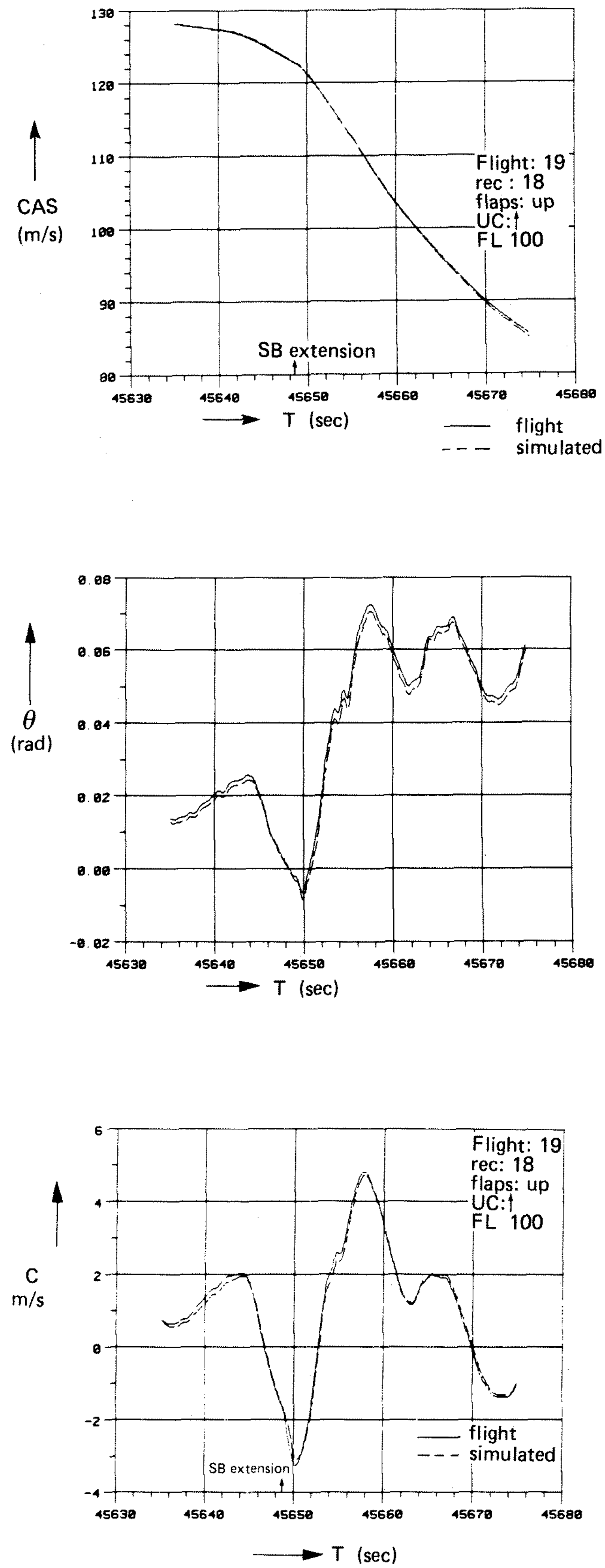

Fig. 13 Level flight deceleration

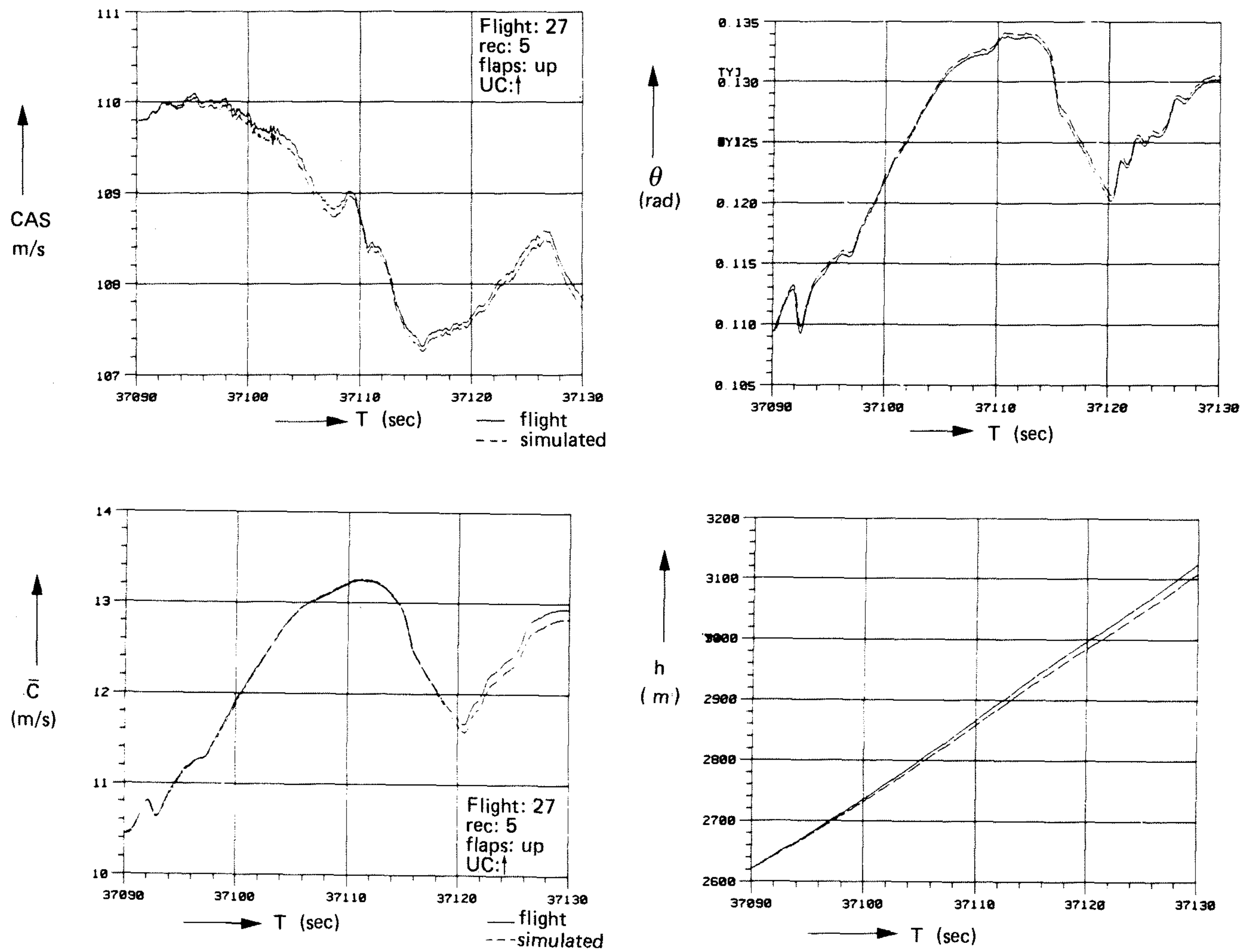

Fig. 14 All engine climb

as is indicated in figure 6. Because the reconstruction of the flight path on the ground is different in nature from the one required in free air, these cases have been dealt with separately. As a result of this also two sets of parameters were defined viz. one set of ground parameters and one set of air parameters. From these sets the variables could be selected, specified by the Simulator manuafacturer (CAE), for a particular ATG manoeuvre. The ATG recordings were visually checked and selected on a graphics terminal. After this the data were converted to TH COPY format, put on magnetic tape and sent to CAE. CAE has standard procedures available to make high resolution plots of the data in a form directly suitable for the FAA evaluation.

The POM data required by the FAA are a specific selection from the ATG test data. The flight recordings necessary for the proof of match creation were sent to DUT. Here the mass properties and initial atmospheric parameters, present at the start of a flight test manoeuvre, were adopted in the simulation programme. Because practical experience shows, that the initial conditions of the flight test manoeuvre and the computed initial condition of the simulation do not coincide exactly, small offsets in control surfaces were used to compensate for the bias. After this, the input signal of the flight test was employed to drive the simulation programme with the submodels of interest included. The resulting model responses then could directly be compared with the flight test time histories. Hereby it must be mentioned, that with respect to the angle of attack and sideslip angle the reconstructed values of the flight test data are considered in the comparison with the simulation results. This in contrast with what usually is done, viz. a comparison with the corrected measured vane angles.

Obviously criteria are required to accept or reject a certain model response. To fulfill phase II requirements in reference 1, for the requested POM tests, tolerances are defined for a number of specific parameters representative for that test. Using reference 1, it appeared that the POM data fell within these tolerances.

After the POM data were generated they were put on magnetic tape and sent to the simulator manufacturer.

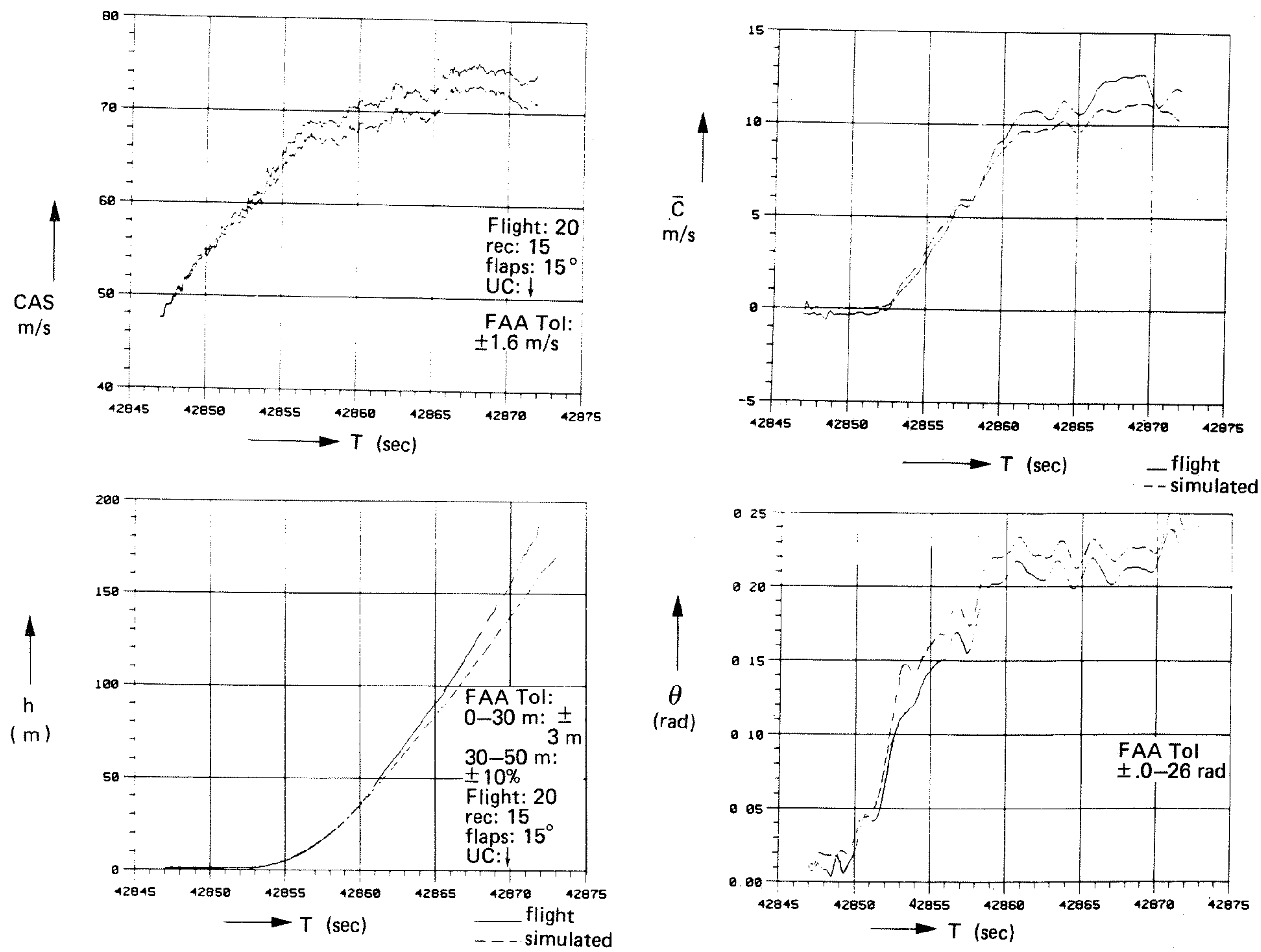

Fig. 15 Normal take-off

As an example in figures 12 to 15 time histories are presented from the POM data base. Characteristic parameters are shown of a level flight acceleration and deceleration, an all-engine partial climb and a normal take-off. Furthermore in figures 16 and 17 characteristic motions are depicted viz. the short period and dutch roll. In the figures also the simulation responses are depicted (dotted lines) as a result of the flight test measured input signals. As can be noticed a good agreement is achieved. Where appropriate the tolerances are indicated on the plots as defined by the FAA.

7. Concluding remarks

Ground based flight simulation as a training tool becomes more and more important in todays aviation. However, as a result of this the requirements with respect to the fidelity have been increased. Obviously also the proof of this fidelity is essential. In particular for small and older aircraft, however, the flight test data base is marginal with respect to the present demands of for instance a Phase II approved simulator. This situation arose when a Phase II approved simulator was required for the Cessna Citation 500 of RLS. Therefore a flight test programme was carried out with an accurate instrumentation system. Employing dynamic flight test techniques and advanced computer data processing and parameter identification techniques, it was possible to generate the mathematical models and data base of the Citation 500 in a relatively short period of time. The models were implemented in a 6 DOF engineering simulation. Using this simulation, it was possible to generate a proof of match, where a comparison was made between the flight test measured responses and the simulation responses using flight recorded input signals. From this it could be concluded, that the identified mathematical models were adequate representations of the actual aircraft characteristics (Ref. 9, 10, 11).

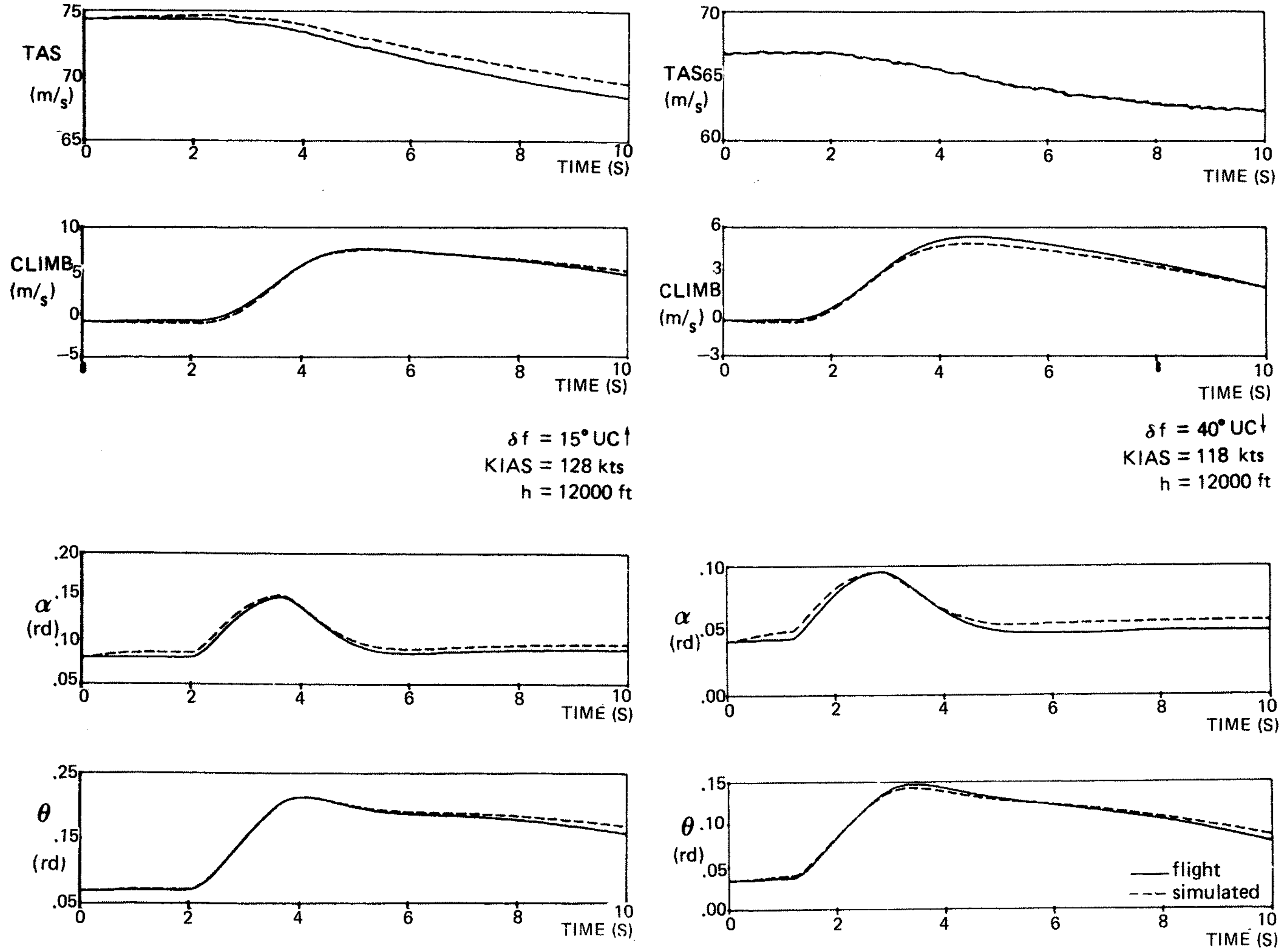

Fig. 16 Short period dynamics
FAA tol: 10% of period
.02 in damping

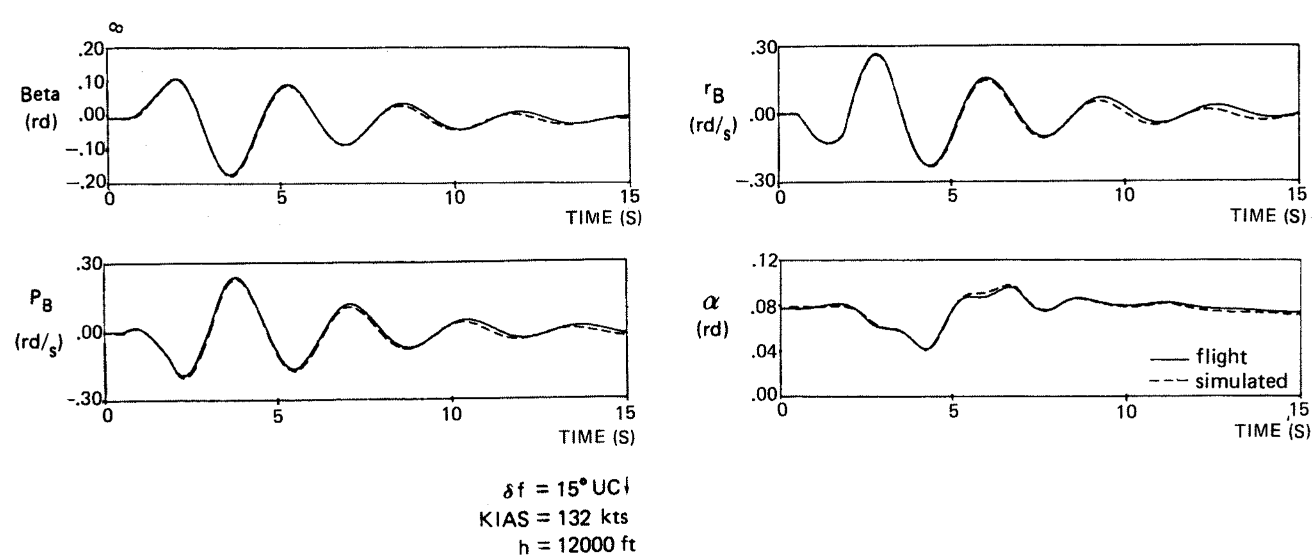

Fig. 17 Dutch roll dynamics (yaw damper off)
FAA tol: 10% of period
.02 in damping

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