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ADVANCES IN ROTORCRAFT SYSTEM IDENTIFICATION

Peter G. Hamel and Jürgen Kaletka

Deutsche Forschungsanstalt für Luft- und Raumfahrt e.V. (DLR), Institut für Flugmechanik, 38108 Braunschweig, Germany

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Abstract—System identification can best be described as the extraction of system characteristics from measured flight test data. Therefore it provides an excellent tool for determining and improving mathematical models for a wide range of applications. The increasing need for accurate models for the design of high bandwidth control systems for rotorcraft has initiated a high interest in and a more intensive use of system identification. This development was supported by the AGARD FVP Working Group 18 on 'Rotorcraft System Identification', which brought together specialists from research organisations and industry, tasked with exploring the potential of this tool. In the Group, the full range of identification approaches was applied to dedicated helicopter flight-test-data including data quality checking and the determination and verification of flight mechanical models. It was mainly concentrated on the identification of six degrees of freedom rigid body models, which provide a realistic description of the rotorcraft dynamics for the lower and medium frequency range. The accomplishment of the Working Group has increased the demand for applying these techniques more routinely and, in addition, for extending the model order by including explicit rotor degrees of freedom. Such models also accurately characterize the higher frequency range needed for high bandwidth control system designs. In the specific case of the DLR In-Flight Simulator BO 105 ATTHeS, the application of the identified higher order models for the model-following control system was a major prerequisite for the obtained high simulation quality.

CONTENTS

1. INTRODUCTION AND OVERVIEW

The rotorcraft dynamic response behaviour (output) due to pilot control or gust disturbance effects (input) is described by the interaction of inertial and aerodynamic forces as well as elastomechanic and control forces acting on the rotor and airframe. The relative interactions and interferences between these forces and their effects on the rotorcraft dynamic response vary with flight states and configurations. The prediction of aeromechanical forces and loads of the rotor system and its wake interferences with the empennage and tail rotor require wind-tunnel and flight test validation experiments.[13] Because of aerodynamic scale effects, wind-tunnel model deficiencies, and constrained 'free flight' capabilities, certain limitations in the quality and applicability of rotorcraft wind-tunnel model data must always be envisaged. Therefore, flight tests are necessary to isolate limitations and assess uncertainties in prediction techniques of rotorcraft aeromechanics. Here, a key tool for rotorcraft flight/ground test correlation is provided by system identification, which in its most general form can be defined as the deduction of system characteristics from measured flight data.

For flight mechanical applications, the aircraft dynamics are modeled by a set of differential equations describing the forces and moments in terms of accelerations, state and control variables. 'Parameter identification' is the term used to describe the process of comparing and minimizing the differences between the calculated model response due to measured control inputs and the corresponding measured aircraft response by iteratively adjusting the model coefficients. In this sense, aircraft system identification implies the determination of physically defined aerodynamic and flight mechanics parameters from flight data. Usually, this is an off-line procedure as some skill and iteration are needed to select appropriate data, develop a suitable model structure, identify the coefficients and, finally, verify the results. This explains why rotorcraft system identification is still seen as an area of research rather than a routine set of tools. For fixed wing aircraft, however, it is more and more routinely applied. Here, the majority of experience has involved the estimation of stability and control derivatives for obtaining linearized rigid body equations of motion for 3 or 6 degrees of freedom (DOF) models.[17,20,25–27] A recently developed application of such in-flight estimated and validated mathematical models includes the provision of extremely accurate models required for advanced (Level D) flight simulators,[3,21,29] certified by FAA.

Unlike the flight dynamics of most fixed wing aircraft, the dynamics of rotary wing aircraft are characteristically those of high order systems. The large number of degrees of freedom associated with the coupled rotor-body dynamics leads to a high number of unknowns to be estimated, making it rather difficult to successfully apply system identification. A conventional six DOF model may be adequate for some handling qualities evaluations, whereas higher order model structures are required for simulation validation or flight control system design. Due to the highly complex dynamic behaviour of rotorcraft, fundamental interdisciplinary scientific knowledge combined with practical research expertise is needed to use identification and mathematical modelling tools in a most efficient way. This is the main reason why, in the past, these techniques were mainly concentrated in research organisations.[30] To establish an improved contact between research institutions and industry, it was deemed appropriate for the Flight Vehicle Integration Panel (FVP) of the Advisory Group for Aerospace Research and Development (AGARD), which for the last twenty years has supported activities in the field of flight vehicle system identification,[1,12,27] to sponsor the Working Group WG 18 and, based on the findings of the working group the lecture series LS 178, both on *Rotorcraft System Identification.* The Group was established in 1988 and brought together a wide range of research specialists and industry representatives, tasked with exploring and reporting on the topic of rotorcraft system identification. For a three year time period the group exercised the full range of available individual system identification approaches, using flight-test-data from three different helicopters. A detailed documentation of the activities is provided in the final report[14] and the lecture series publication.[15]

The present paper first introduces the principle and some basic requirements of system identification. Based on some selected results[16] obtained from the AGARD Working Group for the DLR BO 105 helicopter (Fig. 1) it then concentrates on the determination of linear state space derivative models with six DOF for the rigid body motion. Such models are widely used for various applications. However, for more demanding applications, such as high bandwidth control system design, higher order models with a more appropriate representation of the main rotor dynamics are required.[36] Therefore, the paper will then address the identification of extended models with additional degrees of freedom, which is

Fig. 1. DLR BO 105 Helicopter used for system identification flight tests.

still a subject of research activities and reflects the actual state of the art. As an application example, the use of extended models for the model-following control system design of the DLR helicopter In-Flight Simulator BO 105 ATTHeS is described and simulation results from flight tests are presented.

2. PRINCIPLE OF SYSTEM IDENTIFICATION

The general approach used in aircraft system identification is depicted in Fig. 2. In flight tests, specifically designed control input signals are used to excite the aircraft modes of interest. Both control inputs and aircraft response are measured and recorded. The data quality is analyzed by applying data compatibility and state reconstruction approaches. The identification techniques can be divided into methods which work either in the time or in the frequency domain. Consequently, the measured data are evaluated in the format of time histories or they are transferred into the frequency domain by Fast Fourier (FFT) or Chirp-Z transformations. For the identification step, the aircraft mathematical model is formulated by a set of differential equations or transfer functions. Based on the differences between the model response due to the measured control inputs and the real aircraft response, the unknown model coefficients are adjusted to obtain a better agreement. Usually the estimation is an iterative process. From the principal approach it is obvious that the identification framework can be divided into three major parts:

- *Instrumentation and filters*, which cover the entire flight data acquisition process including adequate instrumentation and data recording.
- *Flight test techniques*, which are related to selected rotorcraft maneuvering procedures in order to optimize control inputs.
- *Analysis of flight test data*, which includes the definition of the mathematical model structure of the rotorcraft, data quality assessment and the identification of the unknowns.

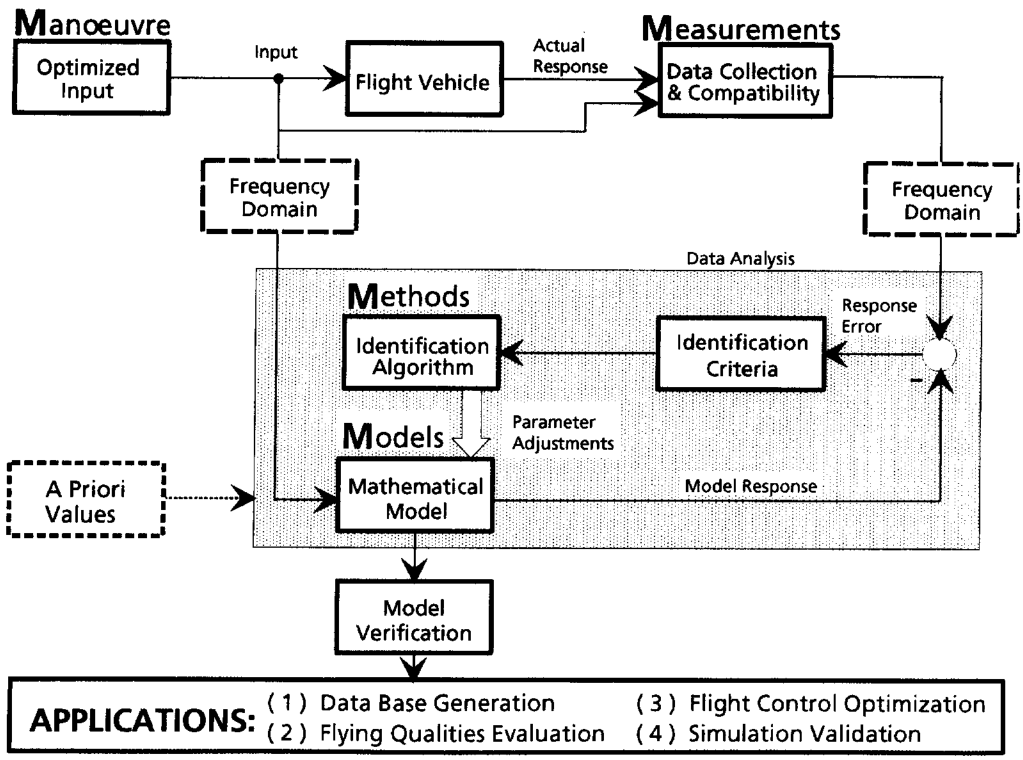

Fig. 2. The 'Quad M' basics of rotorcraft system identification.

Corresponding to these strongly interdependent topics, main emphasis has to be placed on four important aspects of the art and science of system identification:

(1) The **M**aneuver of the helicopter due to control inputs must provide as much information as possible about the dynamic characteristics of the aircraft. It implies the development of appropriate input signals, optimized in their spectral composition, but still flyable by the pilot.
(2) Highly reliable **M**easurements are indispensable, as system identification is always based on the input/output relationship of the vehicle motion. Measurement inaccuracies, that cannot be corrected or compensated, lead to biased estimation results.
(3) The definition of the **M**odel structure is a key element. Depending on the intended purpose, often high order rotorcraft models are needed and a practical compromise between high bandwidth model accuracy, efforts for flight testing, measurement, and data processing, and system identifiability must be found.
(4) Suitable identification **M**ethods are required for both data quality analysis and parameter identification. Here, various techniques of different complexity and performance, working in the time or frequency domain are available.

These '**Quad-M**'-requirements must carefully be investigated from a physical standpoint in order to define and execute a successful experiment for system identification.

In the following, the paper will concentrate on a detailed discussion of each of the four **M**s, Maneuvers, Measurements, Models and Methods.

3. MANEUVERS FOR SYSTEM IDENTIFICATION

Independently from the actually applied technique, system identification approaches rely on the information content about the system under test, provided by the amplitude and phase relationship between the measured control inputs and the resulting measured system response. Therefore, the test input is one of the major factors influencing the accuracy with which the model parameters can be determined. Generally, it should at least meet the following requirements: all helicopter modes within the frequency range of interest must properly be excited; the flight test duration must be long enough to provide sufficient low frequency information; when linear models are estimated, the aircraft response should stay within small perturbations assumptions from trim; no or only minor additional pilot inputs are allowed, e.g. for keeping the response small. Furthermore, it is often desired that the input can be flown by the pilot.

There are a few standard input signals, which are widely applied for aircraft system identification, namely doublet, DLR 3211, and frequency sweep.[23] As an example, Fig. 3 presents the input shapes for the longitudinal control and the resulting pitch and roll rate responses:

(1) Doublet control inputs can be considered as classical input signal for aircraft flight testing in general. In the case of highly coupled high order helicopter models it is generally regarded as having limited value due to its small bandwidth.

(2) The DLR 3211 multi-step input signal has become standard in system identification. The numbers used in the designation refer to the relative time intervals between control reversals. As it excites a wide frequency band within a short time period, it is also suited for moderately unstable systems. Typically, the input signal lasts for about 7 s, then the controls are kept constant until the pilot retrims the aircraft. For rotorcraft identification the signal was slightly modified by intermediate steps to increase the higher frequency content.

(3) Frequency sweeps have been increasingly used in recent years. Starting with a sinusoidal signal of the lowest frequency of interest, the frequency is progressively increased up to its maximum and then the aircraft is returned to trim. Typically, this broad-band input signal lasts for about 50 s.

If possible, flight test data obtained from at least two different input signals should be provided: (1) For the identification, a more complex signal, like DLR 3211 or frequency sweep, is preferable; (2) for the verification of the identified model, flight test data with an input signal dissimilar from the one used for the identification should be available. Here, the classical doublet input is often applied.

Practical experience has shown that for stable or only slightly unstable systems the control inputs and flight tests could be conducted without major problems. However, some general tendencies can be noted:

(1) Pilot-flown control inputs are preferable rather than electronically generated inputs for two main reasons: (a) too large deviations from trim can easily be avoided; (b) in

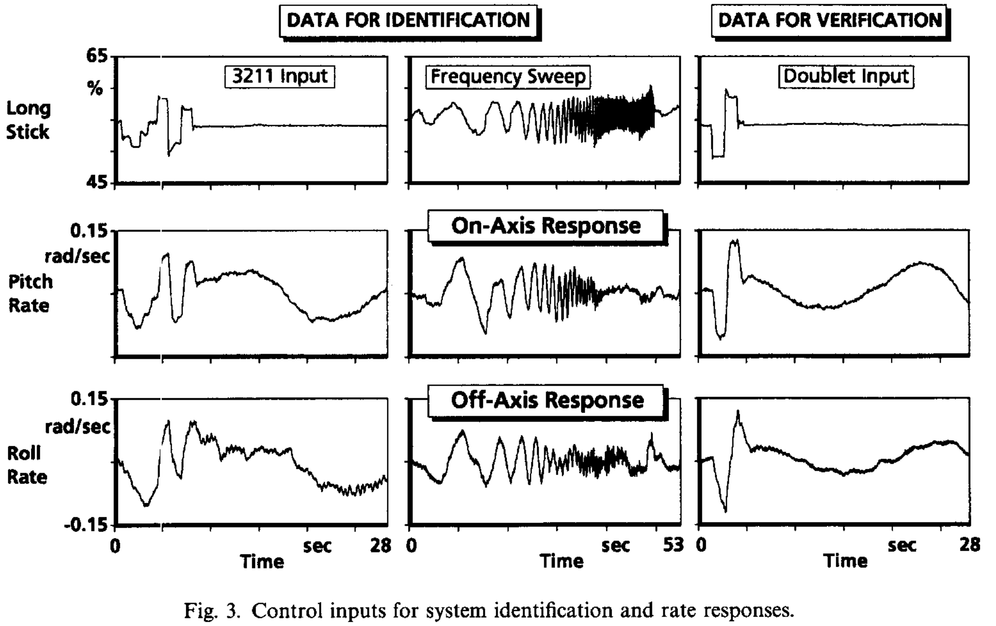

Fig. 3. Control inputs for system identification and rate responses.

contrast to Stability Augmentation Systems, any correcting or stabilizing pilot inputs in the form of small steps or pulses do not cause output/input correlations, which can render the identification impossible. The 'ideal' approach for pilot-flown tests is: (a) to establish trim for the selected flight condition; (b) to generate a prescribed input in one single control and avoid coupling into other controls; (c) at the end of the input signal to let the aircraft respond without further control activity and retrim the aircraft for the next test.

(2) Inputs should be given separately to each control variable. A test run should last at least 25–30 s and the response amplitudes should stay within the small perturbation assumptions of linear derivative models. A test should be repeated to provide redundant data for the evaluation.

(3) It is often difficult to use existing data bases for identification. Dedicated flight testing is necessary, where emphasis is placed on the specific needs associated with these techniques (e.g. same trim flight conditions, proper excitations, small perturbations, sufficient data record lengths, calm air, etc.). Therefore, it is preferable to conduct a flight test program specifically defined for system identification purposes.

As already mentioned above, it is usually relatively easy to conduct the identification tests with prescribed input signals at flight conditions, where the helicopter is stable. For the BO 105 helicopter, this is the case for the medium speed range of about 60–80 knots. Here, the test results summarized in Fig. 3 show that the pilot was able to generate the desired inputs without any further control activity. There was also no significant coupling between the control variables.[23] In hover and low speed flight conditions, however, the BO 105 helicopter has stronger coupled DOF and is highly unstable, with a time to double amplitude of the phugoid mode of about 3.4 s. Consequently, it was not possible to conduct the flight tests in the same manner as those at 80 knots.

As illustrated in Fig. 4 it was already difficult to establish trim at the beginning of the test.[7] Then, a few seconds after the input signal was started a diverging response in the lateral directional motion (roll and yaw) was building up. Although the pilot tried to reduce the roll response by a short pulse in the lateral control, he had to retrim the helicopter after about 14 s as the roll attitude had reached about 45° and the yaw rate was more than 20 deg/s, which exceeded the selected measurement range of the rate gyro. At the end of the test, after about 20 s, the BO 105 had almost made a full turn. The resulting flight test data are not suited for system identification, the pilot was asked to start the input signals as

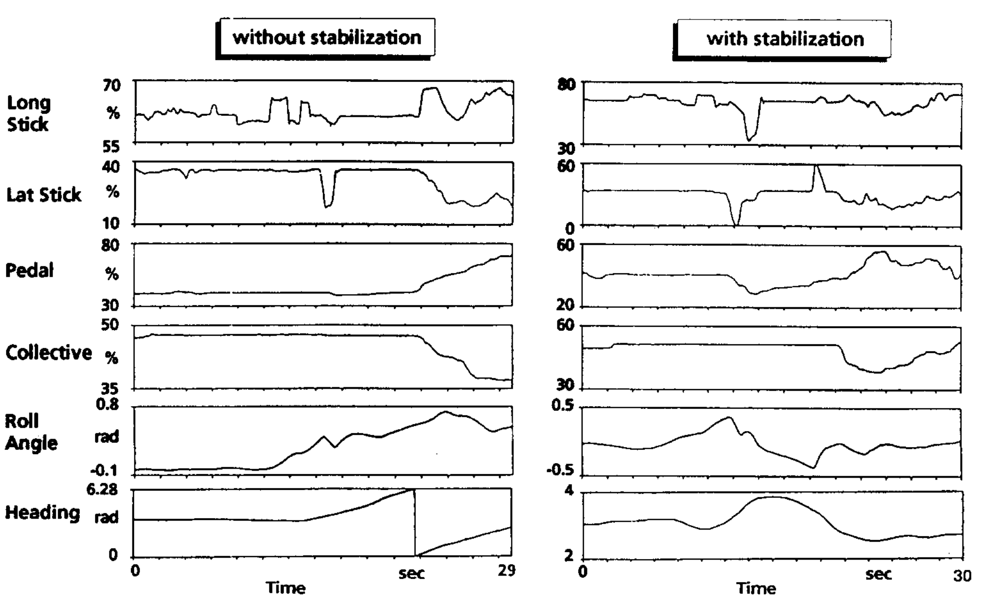

Fig. 4. System identification control inputs for hover flight condition and roll and heading response. Flight tests without and with stabilization by the pilot.

usual, but then to apply additional inputs on his own to keep the response within acceptable amplitudes of about 25° deviation from trim. When comparing the resulting control activity to the desired input signal, it appears impracticable to design optimized control inputs for the BO 105 in hover. It may be better to ask an experienced pilot to excite the aircraft modes as far as possible taking also into account the constraints on the aircraft response given by system identification requirements.

All hover flight tests were flown with additional stabilizing pilot inputs. From a system identification point of view it is certainly not the desired method of flight testing, but it was considered as the best practical solution. However, the pilot inputs were now influenced by the aircraft response and it had to be checked very carefully to what extent the pilot was acting as a feedback control system. To avoid any output/input correlation problems and difficulties related to the so called closed loop identification, the pilot was asked to use step and pulse-type control inputs instead of continuous control motions, which can be more correlated to output variables. This approach proved to be very effective since a subsequent evaluation of the flight test data did not reveal any data correlation due to feedback influences.

For several applications, such as high bandwidth control system design, it is required that the identified models accurately describe the high frequency characteristics. Consequently, the control inputs must also contain higher frequency excitation to provide sufficient information in the flight test data. However, the excitation of the lower frequency rotor modes and structural coupling modes must be avoided, as it can severely damage the helicopter. In particular, when automatically generated inputs are applied or when pilots are not experienced in flight tests for system identification, the flight test engineer must be aware of this danger. In recent years such problems have indeed occurred, particularly with frequency sweeps flown with large amplitude responses in the higher frequency range.

4. MEASUREMENTS AND DATA ANALYSIS

Measurements needed for the identification are defined by the measurement and control vectors of the selected model. Some techniques (e.g. equation error methods) also require measurements of the state vector variables. In addition, measurements characterising the flight condition are desirable. For the identification of six DOF rigid body models, a standard set of variables to be measured is:

- controls;
- translational information for all three axes (horizontal, lateral, vertical)—velocities, linear accelerations;
- rotational information for all three axes—rates, roll and pitch attitudes, heading;
- optional—rotational accelerations to emphasize higher frequency information (often derived by numerical differentiation of the rates).

Except for the velocities, these variables can be measured with modern instrumentation systems using standard sensors like accelerometers, gyros, potentiometers or synchros. In general, they also provide additional data, such as rotor speed (RPM), engine torque, or a second set of accelerations at the pilot position. There is no specific preference for using individual sensors, strap down or inertial system packages. However, a detailed knowledge about sensor characteristics and internal data processing is indispensable; this is often difficult to obtain for commercially available instrumentation systems. From this point of view, some advantages can be seen in the use of individual sensors, which also provide more redundancies in the measurements.

A high measurement accuracy is the dominant requirement for system identification. The presently available transducers generally provide this accuracy and are appropriate. However, emphasis should be placed on four aspects: (1) the measuring range should be adapted to the test to obtain a high resolution; (2) cross axis sensitivity and sensor misalignment must be minimized; (3) signal (analogue) filtering must be known exactly since phase shifts contaminate identification results; (4) a careful calibration of all components is absolutely necessary.

Air data measurement is still the major problem area. Data obtained from noseboom-mounted pressure sensors and vanes may be acceptable at higher speed, but cannot be used for low speed and hover. Only air data systems especially designed for helicopters, like the Helicopter Air Data System (HADS) or the Low-Range Airspeed System (LORAS), cover the full flight regime, but even these systems do not provide all three axis velocities for all speeds. The measurement accuracy may be acceptable for operational use but for identification purposes these systems are generally unsatisfactory. Therefore, it is often necessary to reconstruct the velocities from linear acceleration and attitude measurements, making use of their kinematic relationship given by the nonlinear differential equations:

$$\dot{u} = a_x - g\sin\theta + rv - qw$$

$$\dot{v} = a_y + g\cos\theta\sin\phi + pw - ru$$

$$\dot{w} = a_z + g\cos\theta\cos\phi + qu - pv\,.$$

Here, the linear accelerations and rates are taken from the measurements and treated as known 'control' inputs. For the attitude angles either measurements or calculated data obtained from the kinematic relationship between rates and attitudes are used. The integration of the differential equation system then yields calculated velocities in the body-fixed axis system as state variables, known as *'reconstructed data'*. They are compared to the measured data to correct for drift effects and initial condition errors in the integration and to determine scaling errors.[23] Then, the analyst can decide to use either the (corrected) measured or the reconstructed data for the further evaluation. However, it has to be remembered, that velocities, measured with respect to air are compared to those with respect to the ground. They can only agree for flights in calm air with no wind or gusts. The DLR BO 105 is equipped with a HADS. In hover it is working within the rotor downwash and provides useful longitudinal and lateral, but not vertical velocities. Therefore, the vertical velocity has to be calculated. To support this reconstruction and to avoid drift effects, Kaletka and Gimonet[24] included the aircraft height, obtained from static pressure measurements, and its relation to the velocities as an additional equation:

$$\dot{h} = u\sin\theta - v\cos\theta\sin\phi - w\cos\theta\cos\phi$$

Although a hover test was conducted, the measurement for the steady state forward velocity showed up to 9 m/s for some tests. Although there was some wind when the tests were flown, the measured values were felt to be too large. During the data consistency analysis, the velocity initial conditions were modified, and it was seen that they have a large influence on the time histories of the reconstructed data. For a flight test with a longitudinal control input, Fig. 5 compares the measurements to the reconstructed velocities for three different initial velocities: (1) assumed to be zero as for ideal hover; (2) fixed at the measured value; (3) identified; together with an associated offset in the measurement equations. This figure illustrates that, except for the offsets, the obtained time histories for the forward velocity are similar and in good agreement with the measured variable. However, larger differences are seen for the lateral speed component and particularly for the vertical velocity. For the selection of the 'right' data, the comparison with the measured height proves that the initial conditions and associated time histories obtained from the identification agree best with the flight data. It also demonstrates the need for the height information for the velocity reconstruction when no vertical velocity measurements are available. It was then decided to use the reconstructed data instead of the measured ones, because of the high quality of linear acceleration data and angular measurements. However, it implies that inertia data are used and not air data as required. More recent approaches, based on differential GPS, provide very reliable position and ground speed measurements. But again, these are inertia data and the problem of accurate airflow measurements is still unsolved.

For the identification of extended models with explicit rotor degrees of freedom, measurements of rotor blade motions are required in the measurement vector to match the calculated rotor response of the model with the flight data. Therefore, the BO 105 rotor blades were instrumented with strain gauges at the location of the equivalent hinge offset, to measure the blade flapping motion. The sensors were calibrated on the rotating blades under realistic conditions of airloads and centrifugal forces. The data were transferred by a multiblade coordinate transformation from the rotating to the fixed body axis system to obtain the tip plane motion in terms of longitudinal and lateral flapping and coning. As an example for the data accuracy obtained, Fig. 6 shows the high correlation between the vertical acceleration and the coning angle (body fixed axis system) for collective lever control inputs. The low noise level in the tip path plane variables also verifies the reliability of the blade flapping calibration.

As system identification is usually an off-line process and conducted after completion of flight testing, it is of utmost importance to check and ensure the data quality. Here,

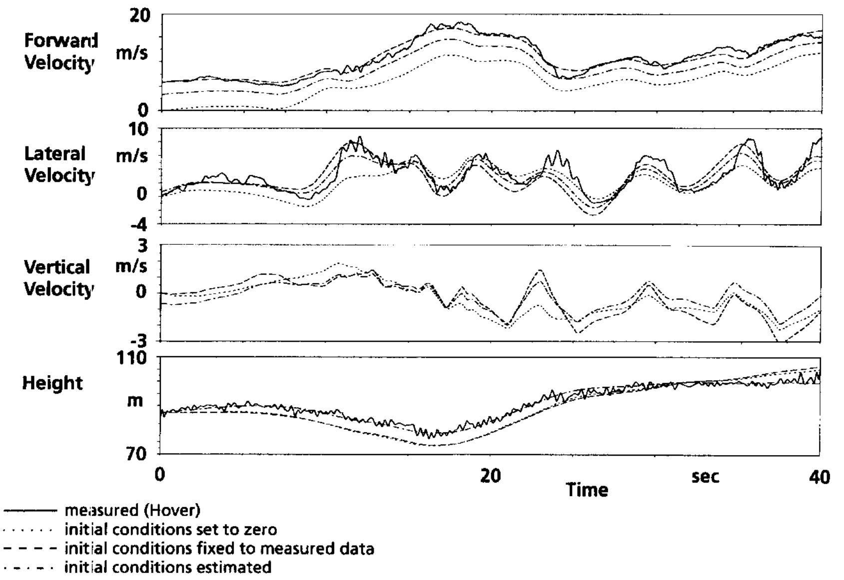

Fig. 5. Comparison of measured velocities to reconstructed velocities and their sensitivity to initial conditions.

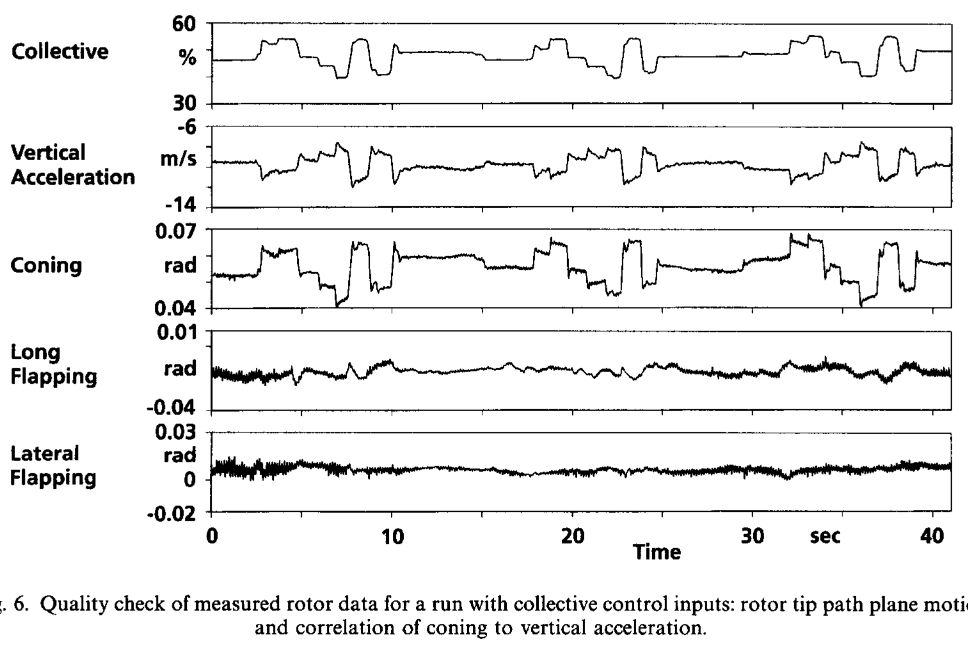

Fig. 6. Quality check of measured rotor data for a run with collective control inputs: rotor tip path plane motion and correlation of coning to vertical acceleration.

dedicated flight tests conducted right before and during the actual flight tests should be evaluated immediately to detect any errors as soon as possible. Such checks range from visual data inspection to complex data kinematic compatibility analyses, like Kalman filter state reconstruction or nonlinear maximum likelihood identification methods.

5. MATHEMATICAL MODELS

The selection of the model structure is a critical step in system identification, which will greatly affect both the degree of difficulty in extracting the unknown parameters and the utility of the identified model in its intended application. Presently, most of the system identification work is still devoted to the determination of parametric fully coupled 6 DOF linear derivative models. These relatively simple models are considered appropriate for the description of the rigid body dynamics for the low and mid frequency range and applicable e.g. for flying qualities evaluations. They are formulated by the state equations characterizing the system dynamics:

$$\dot{\mathbf{x}} = A\mathbf{x} + B\mathbf{u}$$

and the measurement equations, which give the relationship between measured data and model responses to be matched by the identification technique:

$$\mathbf{y} = C\mathbf{x} + D\mathbf{u}$$

with

$$\text{state vector} \quad \mathbf{x} = [u, v, w, p, q, r, \phi, \theta]^{\mathrm{T}}$$

$$\text{control vector} \quad \mathbf{u} = [\text{long stick, lateral stick, collective, pedal}]^{\mathrm{T}}$$

and a representative measurement vector

$$\mathbf{y} = [u, v, w, p, q, r, \phi, \theta, a_x, a_y, a_z]^{\mathrm{T}}$$

(can be modified).

Such models have their justification as, usually, the eigenvalues associated with blade dynamics represent modes that are sufficiently higher in frequency than the rigid body modes. But it has to be remembered that use of 6 DOF models is a major restriction and that for a more realistic and broader bandwidth representation at least main rotor state variables associated with blade flapping are needed, which adds a significant amount of complexity to measurement requirements, model formulation and identification. Here, only first approaches to determine higher order models are made. They will also be addressed later in the paper in the chapter on extended models.

Mostly, linear models based on small perturbation assumptions are used for the identification. They are only valid for the particular flight condition considered. Although some identification techniques can be applied to fully nonlinear model formulations, the preference for linear models has two main reasons: (1) most tools for further system characterization, such as stability analysis, and for intended applications, such as control system design, require linearized differential equations; (2) because of the high complexity of the helicopter model it is difficult to specify those nonlinear terms that have a significant effect on the system response and can accurately be identified. An often used compromise is a model formulation with linear aerodynamics and nonlinear kinematic and gravity terms. It significantly helps to reduce prediction errors, when the identified models are applied to calculate higher amplitude maneuvers.

The present standard for rotorcraft system identification is the formulation of system dynamics in the format of flight mechanical stability and control derivative models, which represent a global description of the helicopter behaviour. There are two additional different approaches to rotorcraft modelling: generic analytical models and information obtained from wind-tunnel testing.

5.1. GENERIC ANALYTICAL MODELS

Based on theory or experimental data generic models determine the forces and moments acting on individual helicopter components, like main and tail rotor blades, fuselage, engine, etc. The obtained results are evaluated depending on their intended purpose. As an example, for helicopter motion simulation the forces and moments are introduced into the equations of motion to obtain time history responses due to control inputs. Various implementations of simulation computer codes are in use. They differ in application (real time/non real time) and sophistication, mainly depending on the complexity of rotor inflow calculation and rotor blade motion resolution. Comparisons of simulation results with flight measurements usually show good agreement for the on-axis responses. A major problem area is the accurate prediction of cross coupling between pitch and roll motions, which can even have the wrong sign.[11]

5.2. ROTORCRAFT WIND-TUNNEL TESTING

For fixed-wing aircraft, wind-tunnel testing has become standard and is extensively used to determine and optimize aerodynamic characteristics. However, except for fuselage tests, rotorcraft wind-tunnel tests are still rare, mainly for three reasons: (1) Rotor tests have to be conducted with rotating blades, which requires a test rig with a rotor drive system, precise control of the rotor states, measurement of high frequency blade variables in the rotating system, data transfer to the fixed system and real time high sampling data recording, evaluation, and monitoring. (2) Because of limited wind-tunnel size, usually model rotors are tested. Then, rotor scaling must provide both aerodynamic and dynamic similarity. This is only approximately possible and only, when relatively large model rotors are used. Consequently, large wind-tunnels are needed. (3) Because of scaling effects, model blades must be manufactured with extremely high accuracy.

From these requirements, it is obvious that in comparison to fixed-wing aircraft, wind-tunnel testing of rotorcraft is significantly more complex and expensive.[13] In addition, wind-tunnel tests usually only provide information about steady-state characteristics. Simulated 'free' flights, as sometimes conducted for airplane models to obtain information about the dynamics, cannot easily and realistically be accomplished with helicopter models. Therefore, often helicopters are still designed without wind-tunnel measurements and then tested and modified during the flight test phase.

5.3. CROSS-FERTILIZATION OF MODELING APPROACHES

The three approaches, analytical modeling, wind-tunnel model measurements, and flight test evaluation by system identification are well established and are widely used. Each of these techniques has its strengths and weaknesses. An efficient way to combine the individual strengths for a general model improvement has not yet been developed. Analytical simulation uses wind-tunnel results, but confidence levels are not given and the transferability from model to real size helicopter characteristics is not exactly known. System identification certainly provides the most accurate model, but the formulation in derivative format is too global for the extraction of more detailed information, as needed for analytical model improvement. Comparisons of all three approaches applied to the same helicopter type are very rare. Here, more problem oriented research is needed to define practical approaches for rotorcraft model improvement, which combine the benefits of the individual approaches. In the AGARD Working Group Report it was tried to stimulate this topic by a chapter on simulation validation.[14,15]

6. SYSTEM IDENTIFICATION METHODS

System identification techniques can be divided into methods working in the time domain or in the frequency domain. Both approaches were extensively used in the AGARD

Working Group and are described in more detail in the Working Group's final report.[14] All techniques have in common, that a 'best' set of unknown parameters is determined by minimizing a given cost function. Depending on the assumptions for the cost function formulation, the minimum search ranges from easily applicable analytically derived closed solutions up to complex iterative solution algorithms. The main difference is the ability of the techniques to handle noisy data, where two sources of noise are considered: process (or state) and measurement noise. Estimation of all unknowns and both noise characteristics leads to very sophisticated methods and is computationally tedious. Assuming that the flight tests were flown in calm air with negligible process noise, the general form of the cost function is given by:

$$J(\xi) = \Sigma\, e^{\mathrm{T}} \mathbf{R}^{-1} e\,.$$

It is minimized by adjusting the unknown parameters ξ. Differences between the individual techniques are principally characterised by the definition of the error e and the weighting matrix $\mathbf{R}$.

6.1. TIME-DOMAIN APPROACHES

The majority of aircraft identification work has been based on the maximum likelihood technique accounting for measurement, but neglecting process noise. Strictly, this is a so called 'output error method' with:

$$e(t) = \mathbf{y}(t)_{\mathrm{meas}} - \mathbf{y}(t)_{\mathrm{model}}$$

$$\mathbf{R} = \frac{1}{N} \sum \left[\mathbf{y}(t)_{\mathrm{meas}} - \mathbf{y}(t)_{\mathrm{model}}\right] \left[\mathbf{y}(t)_{\mathrm{meas}} - \mathbf{y}(t)_{\mathrm{model}}\right]^{\mathrm{T}}$$

Here, $\mathbf{R}$ is the measurement error noise matrix. To obtain the minimum of the cost function $J(\xi)$ with respect to the unknowns ξ, all first derivatives $\delta J/\delta \xi$ have to be zero. This leads to a set of nonlinear equations that can only be solved iteratively with the main steps: (1) calculation of the cost function J; (2) determination of the matrix $\mathbf{R}$; (3) update of the unknowns ξ by minimizing the cost function J with respect to the unknowns ξ; (4) calculation of a new measurement vector $\mathbf{y}$ and either return to step one or stop. In addition to the parameter estimates, the technique also provides uncertainty levels (standard deviations) for the estimates. The parameters to be determined are primarily the system coefficients (derivatives), but, in addition, so called bias terms have to be estimated to compensate for effects due to variable zero offsets and drifts. These biases can drastically increase the total number of parameters to be determined and complicate the identification.

For aircraft identification the maximum likelihood approach is commonly used as it has several desirable statistical properties. In addition, it allows the identification of non-linear models, where the nonlinearities can be formulated using actual state variables.

As an alternative to the assumptions of zero process noise, algorithms can be based on an assumption of perfect measurements of state and control variables. Then, the state variables are formally treated as (pseudo) controls and the individual equations of motion can be identified separately by minimizing the equation error. With:

$$\dot{\mathbf{x}} = A\mathbf{x}_{\mathrm{meas}} + B\mathbf{u}$$

$$\mathbf{y} = \dot{\mathbf{x}} + e$$

$$e = \mathbf{y} - A\mathbf{x}_{\mathrm{meas}} - B\mathbf{u} = \mathbf{y} + \xi\mathbf{X}$$

$$J(\xi) = \Sigma\, e^2$$

it follows

$$\xi = (\mathbf{X}^{\mathrm{T}}\mathbf{X})^{-1}\mathbf{y}$$

which is the closed, non-iterative solution for the so-called equation error technique. It is a fast and easily applicable method. However, it requires extremely accurate data and, therefore, high emphasis has to be placed on data pre-processing.

6.2. FREQUENCY-DOMAIN APPROACHES

Frequency-domain approaches have some unique characteristics which make them attractive for rotorcraft identification. In particular, they allow concentration on selected frequency ranges, are insensitive to system instabilities, and do not require estimation of biases. Two different approaches are increasingly applied assuming linear model structures: A maximum likelihood technique and a method based on transfer function evaluations.

Based on the work of Klein,[25] Marchand and Fu[28] developed the frequency-domain maximum likelihood approach. The measurements are represented in the format of Fourier Transforms. The model, cost function, and covariance matrix are formulated and determined in the frequency domain:

$$j\omega\mathbf{x}(\omega) = A\mathbf{x}(\omega) + B\mathbf{u}(\omega)$$

$$\mathbf{y}(\omega) = C\mathbf{x}(\omega) + D\mathbf{u}(\omega)$$

$$e(\omega) = \mathbf{y}(\omega)_{\mathrm{meas}} - \mathbf{y}(\omega)_{\mathrm{model}}$$

$$J(\xi, \omega) = \Sigma\, e(\omega)^{\mathrm{T}} R(\omega)^{-1} e(\omega)$$

$$R(\omega) = \frac{1}{N}\sum[\mathbf{y}(\omega)_{\mathrm{meas}} - \mathbf{y}(\omega)_{\mathrm{model}}]\,[y(\omega)_{\mathrm{meas}} - y(\omega)_{\mathrm{model}}]^{\mathrm{T}}$$

Comparisons between flight test data and model response are made in frequency-domain formats (e.g. Bode plots) or, after re-transformation, in the form of time histories.

A different approach, registered as CIPHER, was developed and applied by Tischler.[36–39] Here, a specified group of conditioned transfer functions $T(\omega)_{\mathrm{measured}}$ is first derived from the flight test data. Major emphasis is placed on high accuracy over a wide frequency range. In a second step, the model transfer functions are formulated as:

$$T(\omega)_{\mathrm{model}} = (C(I - A)^{-1} B + D)e^{\tau\omega}$$

where I is the identity matrix and τ is the time lag.

The error and cost function are then

$$e(\omega) = T(\omega)_{\mathrm{meas}} - T(\omega)_{\mathrm{model}}$$

$$J(\xi, \omega) = \Sigma\, e(\omega)^{\mathrm{T}}\, \mathbf{W}(\omega)\, e(\omega)$$

where the weighting matrix $\mathbf{W}$ is based on the coherence of each frequency point to emphasize the most accurate data. For the comparison to flight test data, usually frequency responses are used. Time histories are generated by driving the identified model with the measured control inputs.

6.3. VERIFICATION OF RESULTS

The verification of the identified model is the final key step in the identification process, which assesses the predictive quality of the extracted model. Flight test data not used during the identification process are selected to ensure that the model is not tuned to specific data records or input shapes. It is desirable to conduct the verification with data from the same flight conditions, but with dissimilar control inputs.

As an example, the BO 105 identification results were generated from flights with DLR 3211 multistep or frequency sweep inputs. Then, for verification, data with doublet control inputs for each control were used: The previously identified model was driven with the measured control variables and the obtained response was compared to the measured helicopter response. When a satisfying agreement is obtained the identification is completed and the identified model is made available for the intended application. Usually it is provided in form of model coefficients and their standard deviations.

7. IDENTIFICATION RESULTS FOR 6 DOF MODELS

The AGARD Working Group has evaluated flight test data from three different helicopters, AH 64, BO 105, and PUMA, where the data bases were provided by MDHC*, DLR and DRA*. The obtained results are presented and discussed in the final report[14] and the AGARD lecture series.[15] For this chapter, some results obtained from DLR BO 105 flight test data were selected to give a representative insight into the model accuracy that can be expected from system identification.

7.1. IDENTIFICATION

After data processing and reliability checking, three suitable data sets were selected. Each of them consisted of four different runs with single control pilot inputs. (1) Data runs with 3211 input signals were used for identifications in the time-domain as well as the frequency-domain identification based on Fourier transforms. (2) Frequency sweep data were used for the frequency-domain identification based on transfer functions. (3) Data obtained from doublet control inputs were applied for the verification of the identified models.

The Working Group members applied their individual identification techniques:

AFDD—the frequency domain method first evaluates frequency sweeps to generate transfer functions, which are then used to identified linear state space models.

CERT, DLR, NAE—time domain maximum likelihood methods were applied. The individual approaches are slightly different from each other. Main differences refer to the treatment of the nonlinear gravity terms, like $g^* \sin\theta$: CERT identified fully linearized models, NAE used so called 'pseudo control' or 'forcing function' approaches where the nonlinear terms are calculated from measurements and treated as known additional control inputs, DLR identified nonlinear models.

Glasgow University—linear state space models were identified from Fourier transforms in the frequency domain.

NLR—A time-domain equation error technique based on least squares was applied to determine linear models.

Only 6 DOF rigid body models were identified. Most members developed their model structures by first isolating variables, which have little or no influence on the aircraft response. They were assumed to be zero. Consequently, from a total of 60 possible derivatives the number of actually identified parameters varied from 58 (NAE) to only 30 (University of Glasgow). A further important difference was the inclusion of equivalent time delays to approximate rotor dynamic effects. Here, either the time delays were neglected, set to values suggested by DLR, or individually identified.

As a representative identification result, Fig. 7 demonstrates the good agreement between the measured data and the model response for four different data runs (jointly evaluated as concatenated runs). It confirms that the basic model structure with 6 DOF can correctly represent the helicopter dynamics. A full list of the associated identified parameters and their variances is provided in the Working Group's final report. Some of the major identified derivatives are given in Table 1. Comparisons showed significant differences, even for some of the more important derivatives. The discrepancies were investigated in more detail and could mostly be related to the above mentioned differences in the approaches. As an example, it was experienced that the identification of pitch and roll damping for the BO 105 is very sensitive to both time delays and correlation with control derivatives. In general, it was seen that results obtained from the frequency-domain approaches and the maximum likelihood time-domain techniques are relatively close, when time delays are included. The

* AFDD, Aeroflightdynamics Directorate, U.S. Army, U.S.; CERT, Centre d'Etudes et de Recherches de Toulouse, France; DRA, Defense Research Agency, U.K.; MDHC, Mc Donnell Douglas Helicopter Company, U.S.; NAE, National Aeronautical Establishment, Canada; NLR, National Lucht-en Ruimtevaartlaboratorium, Netherlands.

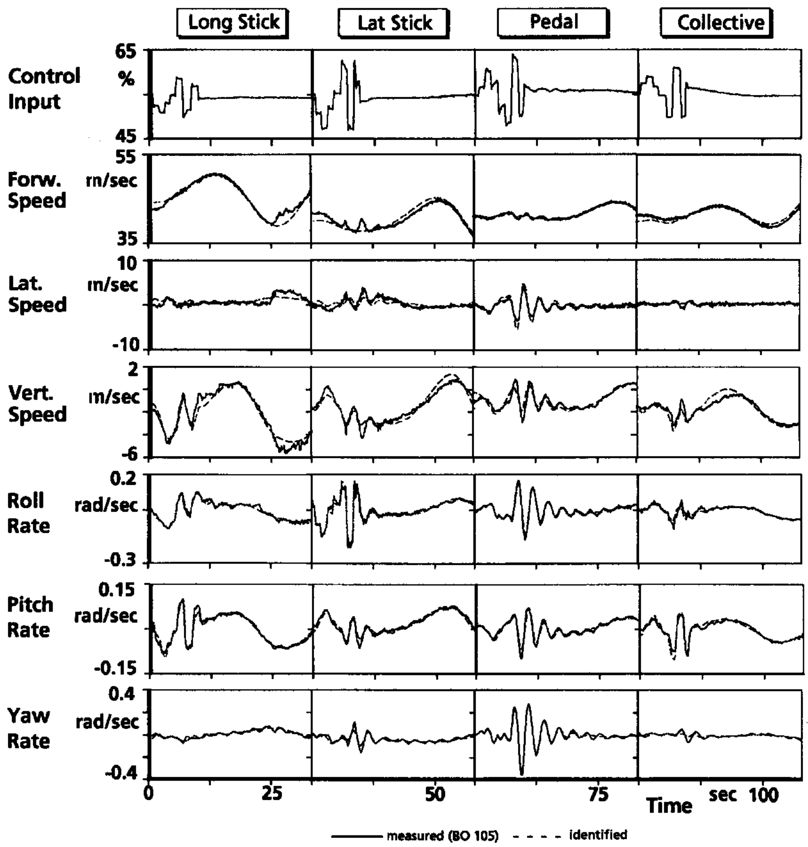

Fig. 7. Representative identification result obtained in the AGARD WG 18 for six DOF models: comparison of BO 105 flight test data to the response of the identified model.

Table 1. Comparison of selected BO 105 derivatives identified by the members of the AGARD FVP Working Group 18

	AFDD frequency domain		CERT ML (linear)		DLR ML (nonlinear)		Glasgow University ML frequency domain		NAE ML (linear)		NLR LS equation error	
Derivative	Value	(S.D.)	Value	(S.D.)	Value	(S.D.)	Value	(S.D.)	Value	(S.D.)	Value	(S.D.)
X_u	−0.0385	(0.004)	−0.058	(0.002)	−0.059	(0.001)	−0.032	(0.004)	−0.050	(0.010)	−0.050	(0.001)
Y_v	−0.221	(0.011)	−0.177	(0.003)	−0.170	(0.003)	−0.131	(0.055)	−0.279	(0.005)	−0.064	(0.003)
Z_w	−1.187	(0.056)	−1.171	(0.012)	−0.998	(0.007)	−0.791	(0.018)	−1.106	(0.013)	−0.875	(0.006)
L_p	−8.779	(0.641)	−2.693	(0.056)	−8.501	(0.110)	−4.470	(0.392)	−7.048	(0.201)	−1.995	(0.114)
L_q	3.182	(0.624)	1.558	(0.090)	3.037	(0.125)	0.0*		4.454	(0.222)	0.558	(0.171)
M_p	−0.998	(0.066)	−0.491	(0.015)	−0.419	(0.038)	−1.367	(0.032)	−1.414	(0.066)	−0.878	(0.062)
M_q	−4.493	(0.235)	−2.26	(0.025)	−3.496	(0.047)	−2.217	(0.009)	−2.992	(0.074)	−1.441	(0.093)
N_r	−1.070	(0.061)	−1.116	(0.013)	−0.858	(0.007)	−0.756	(0.064)	−1.017	(0.023)	−0.699	(0.035)
$Z_{\delta\,\mathrm{col}}$	−0.388	(0.020)	−0.273	(0.003)	−0.349	(0.002)	−0.259	(0.021)	−0.337	(0.005)	−0.242	(0.003)
$L_{\delta\,\mathrm{lon}}$	0.073	(0.006)	−0.0028	(0.002)	0.024	(0.003)	0.0*		−0.005	(0.004)	0.0301	(0.005)
$L_{\delta\,\mathrm{lat}}$	0.179	(0.014)	0.0551	(0.001)	0.185	(0.002)	0.0764	(0.010)	0.1361	(0.004)	0.0534	(0.003)
$M_{\delta\,\mathrm{lon}}$	0.098	(0.004)	0.050	(0.002)	0.093	(0.001)	0.0565	(0.002)	0.0787	(0.002)	0.054	(0.003)
$M_{\delta\,\mathrm{lat}}$	0.00*		−0.0003	(0.0003)	−0.009	(0.0008)	0.00*		0.0106	(0.0013)	0.005	(0.0013)
$N_{\delta\,\mathrm{ped}}$	0.057	(0.003)	0.043	(0.001)	0.049	(0.001)	0.443	(0.002)	0.0484	(0.001)	0.044	(0.001)
$\tau_{\delta\,\mathrm{col}}$	0.168	(0.0056)	0.0		0.040		0.102		0.040		0.040	
$\tau_{\delta\,\mathrm{lon}}$	0.113	(0.006)	0.0		0.100		0.044		0.100		0.080	
$\tau_{\delta\,\mathrm{lat}}$	0.062	(0.006)	0.0		0.060		0.074		0.060		0.060	
$\tau_{\delta\,\mathrm{ped}}$	0.044	(0.006)	0.0		0.040		0.0		0.040		0.040	

*Eliminated from model structure.

associated eigenvalues are provided in Table 2. It is seen that the oscillatory modes (phugoid and dutch roll) and the lower frequency aperiodic modes (pitch and spiral) agree satisfactorily. Larger differences occurred for the roll and higher frequency pitch modes, which are directly related to the damping derivative differences.

7.2. VERIFICATION

The verification of the identified models assesses the predictive quality of the extracted model. It is based on flight data different from those used in the identification. The identified model is kept constant and its response due to the measured control inputs is calculated and compared to the measured response variables. Figure 8 gives the pitch and roll rate results

Table 2. Comparison of BO 105 eigenvalues identified by the members of the AGARD FVP Working Group 18

Motion	AFDD frequency domain	CERT ML (linear)	DLR ML (nonlinear)	Glasgow University ML frequency domain	NAE ML (linear)	NLR LS equation error
Phugoid	[−0.36, 0.30]	[−0.17, 0.32]	[−0.15, 0.33]	[−0.010, 0.35]	[−0.14, 0.33]	[−0.07, 0.33]
Dutch roll	[+0.22, 2.60]	[+0.13, 2.51]	[+0.14, 2.50]	[+0.16, 2.27]	[+0.13, 2.58]	[+0.17, 2.17]
Roll	(8.32)	[+0.99, 2.89]	(8.49)	(5.12)	(8.47)	(2.38)
Pitch	(6.04)	—	(4.36)	(1.98)	(4.38)	(1.37)
Pitch	(0.49)	(0.66)	(0.60)	(0.64)	(0.63)	(0.71)
Spiral	(0.03)	(−0.05)	(0.02)	(−0.007)	(0.03)	(0.04)

Shorthand notation: $[\zeta, \omega_0]$ implies $s^2 + 2\zeta\omega_0 s + \omega_0^2$, ζ = damping, ω_0 = undamped natural frequency (rad s^{-1}) $(1/T)$ implies $(s + 1/T)$, (rad s^{-1}).

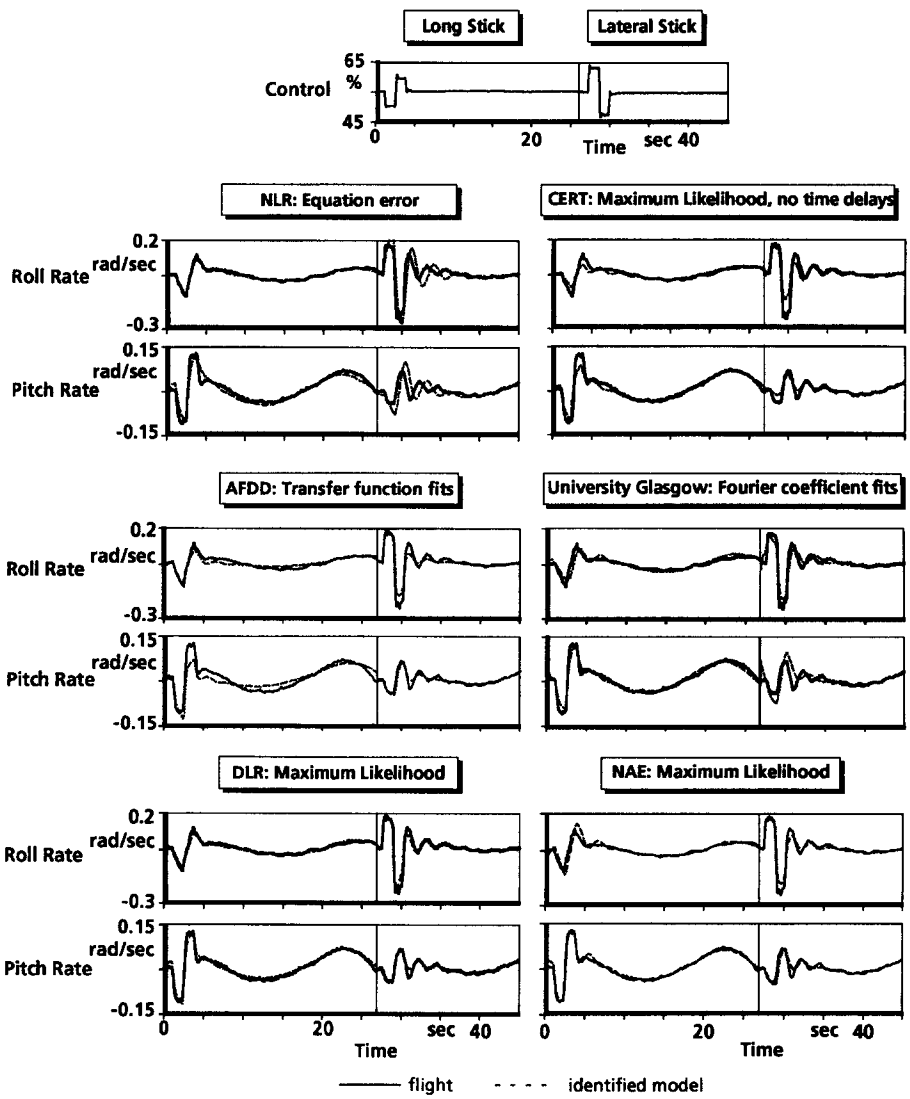

Fig. 8. Verification results obtained in the AGARD WG 18 for six DOF models: comparison of BO 105 flight test data to the predicted response of the identified model.

obtained from flight tests with doublet control inputs. Generally, it is seen that all identified models predict the correct tendency of the aircraft response in both on- and off-axis variables. A closer comparison shows a good agreement for the long-term (low frequency) response, whereas some larger differences are seen for the time history segments, where the control input was given. Here, some models are more damped or the coupling is less accurate. The fact, however, that none of the models can fully reach the maximum peak amplitude of the rates, demonstrates that 6 DOF models cannot describe the higher frequency range completely. In this sense, the models generated by the Working Group can be considered as the best possible solution for rigid body model formulations. Further improvements require an extension of the model order by additional degrees of freedom that provide a more realistic characterisation of the rotor dynamics than the equivalent time delay approach. The comparisons of the individual results revealed larger differences. However, in comparison to other modeling approaches, such as analytical simulation, all identified models show a high level of accuracy. To demonstrate the system identification potential, Fig. 9 compares measured time histories for the responses of both identification and analytical simulation results. The simulation results shown first (in the middle of the figure) are based on the classical blade element technique, which is present state-of-the-art and widely applied. It clearly reveals a fundamental simulation problem: the prediction of cross coupling, which often has the wrong sign. Prouty[34] talks about the 'Cross-Coupling Mystery' and makes it clear that the improvement of simulation codes has become a major challenge. Tischler applied system identification to extract the physical parameters of a 14 DOF analytical rotor formulation from dynamic wind-tunnel tests of a bearingless main rotor. He obtained good agreement for all parameters except for the control-phase angle, which had to be 17° larger than its given geometric value to improve the coupling response. This discrepancy confirmed the fundamental problem in the aerodynamic modeling of the rotor. Rosen[35] investigated the distortion of tip vortices due to a pitch motion of the helicopter. For a nose down motion, the reduced spacing of vortices over the nose increases the local induced velocity and consequently the blade local angle of attack, leading to improved coupling prediction. A very promising approach was also published by von Grünhagen,[10] who extended the effects of the dynamic inflow by including the influence of the downwash rotation of the airmass as virtual inertia effects.

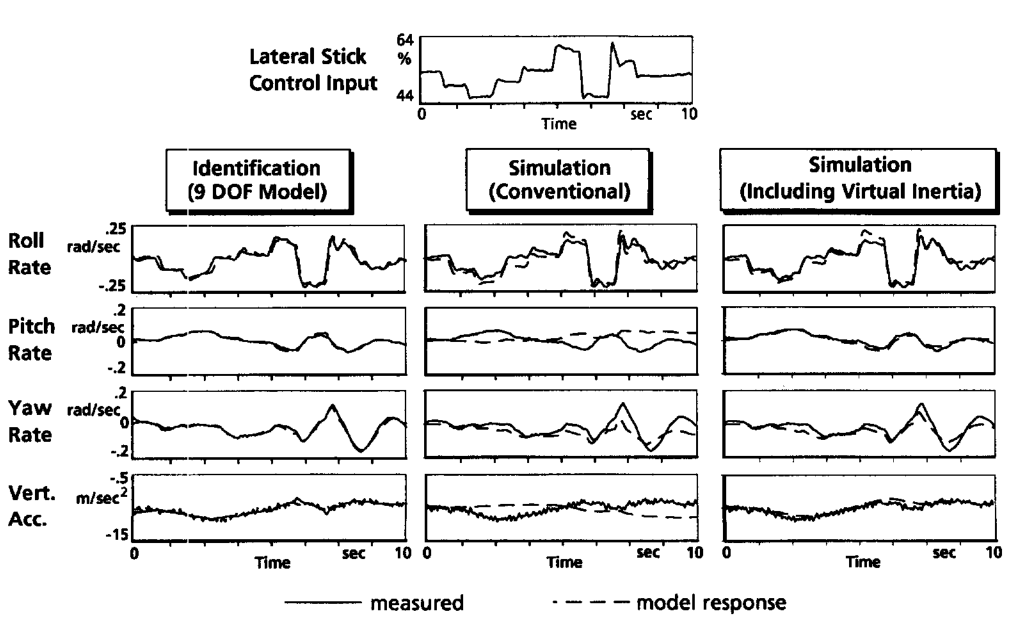

Fig. 9. Comparison of the BO 105 flight test data to the response obtained from the identified model and analytical simulation.

This physically meaningful representation is computationally efficient and therefore also ideally suited for on-line application. The result obtained from this approach is given in the right hand part of Figure 9. The obtained improvement of the coupling response is clearly seen.

8. IDENTIFICATION OF EXTENDED MODELS

The AGARD Working Group mainly concentrated on the identification of 6 DOF models. They are useful for various applications, where less accuracy in the high frequency range (from about 2 Hz for the BO 105) can be tolerated. However, for applications such as high bandwidth control-system design, higher order models with rotor DOF may be required. Here, some first approaches were made during the time period of the Working Group activity. Meanwhile, successful identifications of extended models were conducted by Working Group members. This was particularly emphasised by AFDD and DLR, who need more suitable models for the control system design of their In-Flight Simulators.[4–8, 22–24, 32, 33, 39] Both are presently applying a model structure where, in addition to the 6 d.f. of the rigid body, the rotor tip path plane is represented by first or second order differential equations for coning, longitudinal and lateral flapping (a_0, a_1, b_1). The model is still a system of linear first order differential equations as already shown above for the 6 DOF model. But the state vector and a representative measurement vectors are now:

$$\mathbf{x} = [u, v, w, p, q\ r, \Phi, \Theta, \dot{a}_0, a_0, \dot{a}_1, a_1, \dot{b}_1, b_1]^T$$

and

$$\mathbf{y} = [u, v, w, p, q, r, \Phi, \Theta, a_x, a_y, a_z, a_0, a_1, b_1]^T.$$

As seen from the measurement vector, the identification of higher order models requires the main rotor tip path plane motion variables. They can only be obtained accurately from reliable measurements of the blade motions and the rotor azimuth followed by extensive data processing and/or state estimation.

8.1. REPRESENTATIVE RESULTS

The basic approach for extending the 6 DOF model is illustrated in Fig. 10. The state vector of the rigid body motion is extended by rotor motion variables. Such a model considers 9 DOF and is of 14th order. In comparison to a 6 DOF model, it is more appropriate for

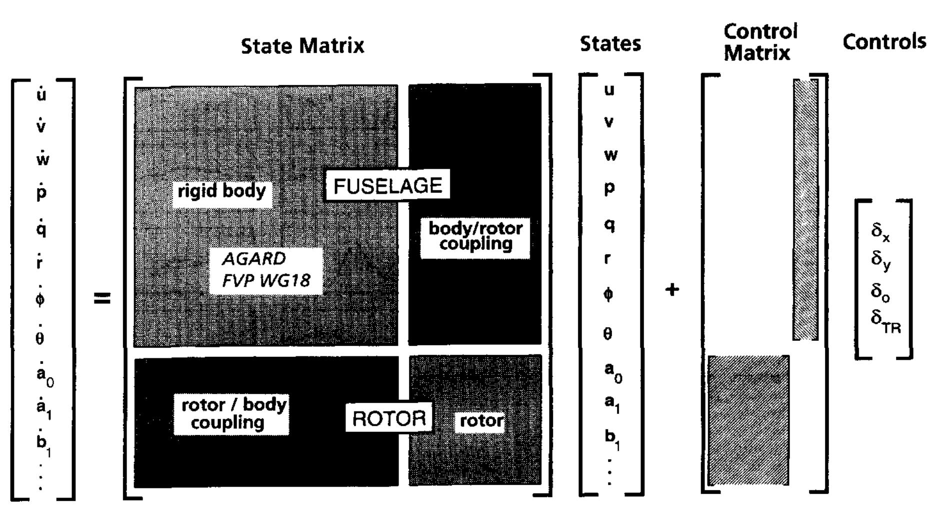

Fig. 10. Principle of extended model structure.

helicopters as it includes a physically realistic representation of the dynamics of the most important and dominant helicopter component: the main rotor. In consequence, the rather crude approximation of rotor dynamics by equivalent time delays is no longer needed. Based on BO 105 identification results from the medium speed range of about 80 knots, Fig. 11 compares the eigenvalues of both identified models, the conventional 6 DOF and the extended 9 DOF model. It illustrates that low frequency rotorcraft eigenvalues (phugoid, spiral, Dutch roll, and pitch) are practically the same for both models, whereas the higher frequency behavior is characterized by the more refined description of the 9 DOF model.

The obtained improvement in the model quality is illustrated in Fig. 12, which compares the roll axis time history responses for both models to the measured data. It is seen that only the 9 DOF model can match the amplitude peaks of the roll acceleration data. These data segments, where the control input is reversed, reveal the model higher frequency behaviour. The model differences become even more obvious when the results are compared in the

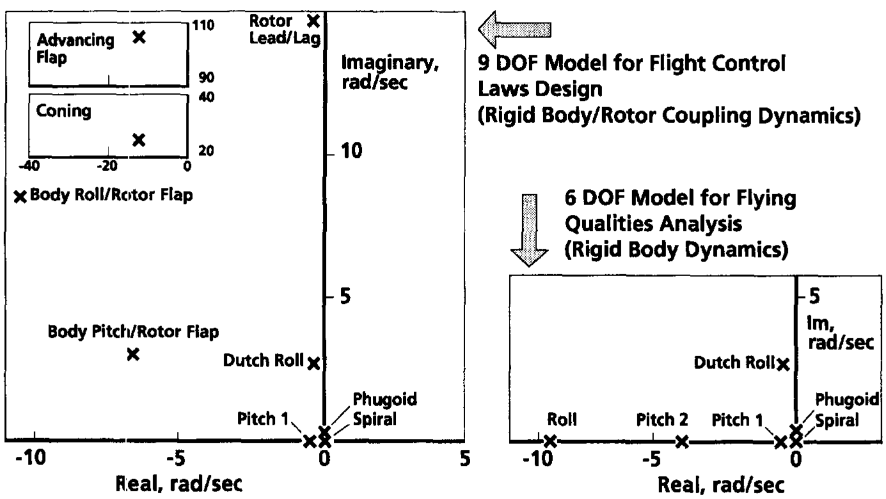

Fig. 11. Comparison of poles from two different identified models: a conventional 6 DOF rigid body model and a nine DOF extended model with second order longitudinal and lateral rotor flapping and coning DOF.

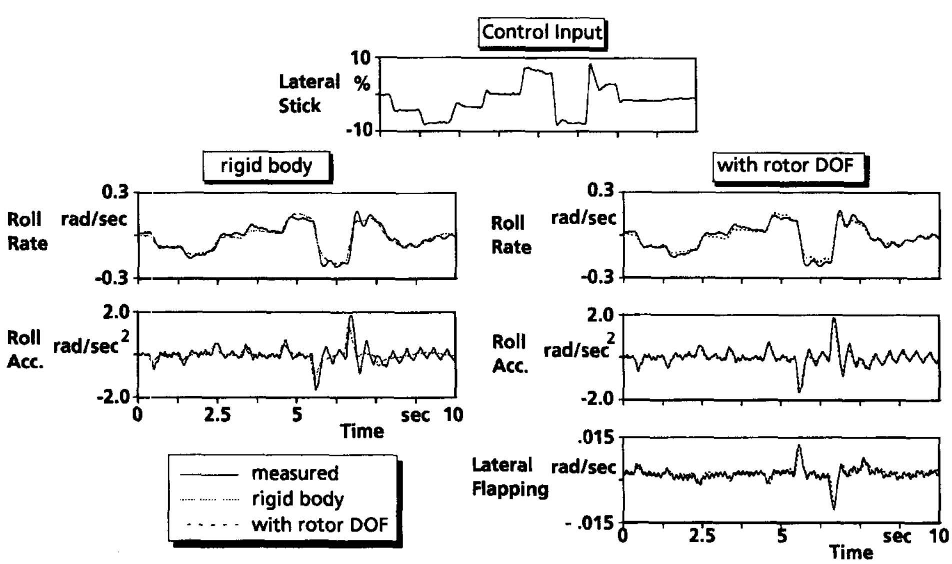

Fig. 12. Comparison of flight test data to six DOF rigid body and nine DOF extended model time history responses.

frequency domain format as is demonstrated in Fig. 13 for the roll rate response due to lateral stick inputs. The results presented above were obtained from BO 105 flight data. However, their general validity is supported by results generated by Fletcher[(5)] for a UH-60 helicopter. Fig. 14 gives frequency responses for the pitch rate (on-axis) and roll rate (off-axis) responses due to longitudinal control inputs. It compares measured UH 60 flight data to both an identified extended model with rotor DOF and a 6 DOF model. Again, it is seen that the conventional 6 DOF model only agrees for the lower frequency range and that a higher order model is needed to accurately represent the rotorcraft high bandwidth dynamics.

To demonstrate that the model formulation is not only valid for the forward flight regime, Fig. 15 presents a short data segment from BO 105 hover flight test data. The excellent agreement of the measured tip path plane variables for the longitudinal and lateral flapping to the response of the identified extended model is clearly seen.

8.2. APPLICATION OF EXTENDED MODELS

At DLR and AFDD main emphasis is placed on the identification of higher order models, which are needed for the control system design of In-Flight Simulators. In Germany, DLR

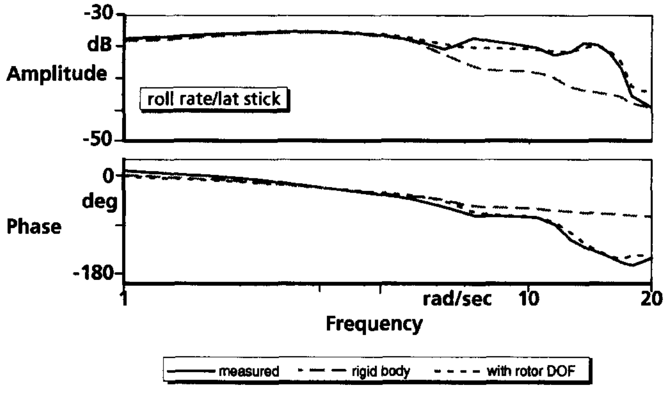

Fig. 13. Comparison of BO 105 frequency responses from flight test data, a six DOF rigid body model and a nine DOF extended model.

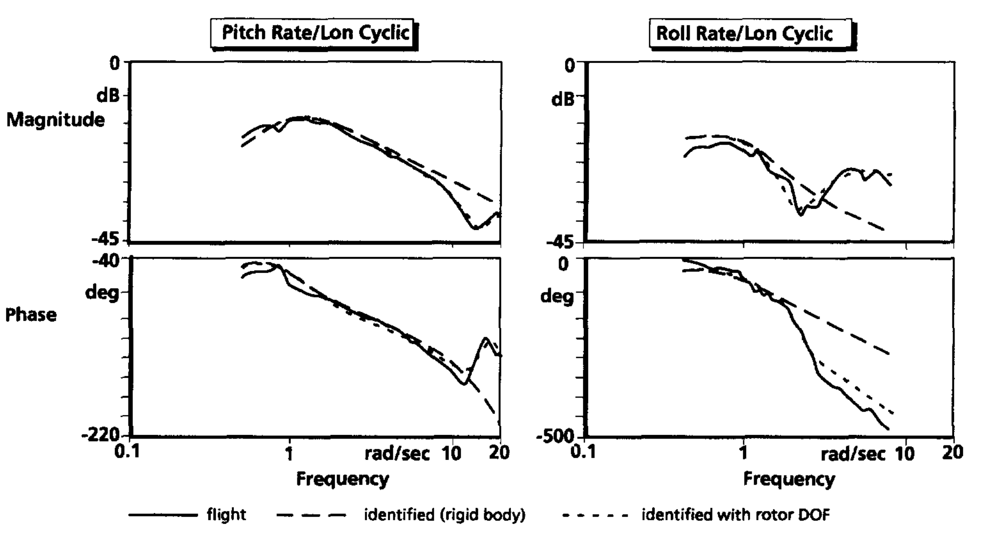

Fig. 14. Comparison of UH-60 frequency responses from flight test data, a six DOF rigid body model and a nine DOF extended model.

operated the BO 105 ATTHeS[31,33] until 1995 and is now developing together with Eurocopter Deutschland and the Wehrtechnische Dienststelle WTD-61 the In-Flight Simulator ACT/FHS on the basis of a EC 135, as described by Gollnick *et al.*[9] In the U.S., Fletcher[5,6] and Tischler[39] applied system identification for the development of the UH 60 RASCAL at AFDD. In the past, ATTHeS was mainly used to provide contributions to the new Handling Qualities Specifications ADS 33. Here, flight conditions at medium speed (about 80 knots) and hover were considered using the in-flight capability for modifying the response characteristics of the original helicopter.

The general concept for an explicit model following control system and the role of system identification is illustrated in Fig. 16. The objective is to make the basic (host) helicopter follow an explicitly defined command model. The commanded model response due to pilot inputs is calculated and fed to the controllers. Here, the feedforward controller is defined as the inverse model of the host aircraft, compensating the host aircraft dynamics. In consequence, the aircraft behaves like the prescribed command model, e.g. like a different helicopter to be simulated. The feedback system mainly corrects for external disturbances. The performance of the control system highly depends on the accuracy of the host aircraft model used for the feedforward controller. First attempts to use 6 DOF models for the BO 105 ATTHeS had only limited success mainly for two reasons: (1) 6 DOF models *without* equivalent time delays to account for rotor dynamics are too inaccurate; (2) models *with* time delays cannot be applied because the inverted model requires 'time lead', which is

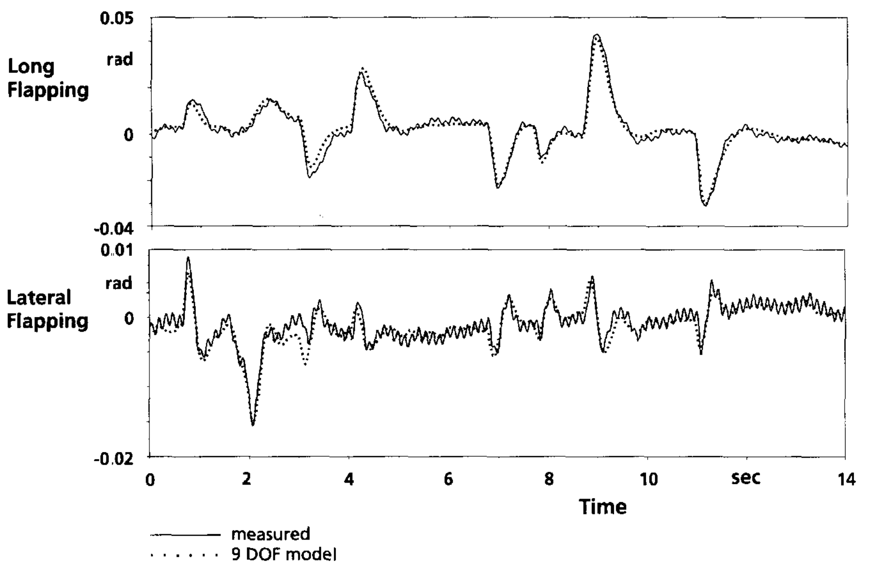

Fig. 15. Comparison of the measured rotor tip path plane data to the response of the extended nine degrees of freedom model for a representative data segment.

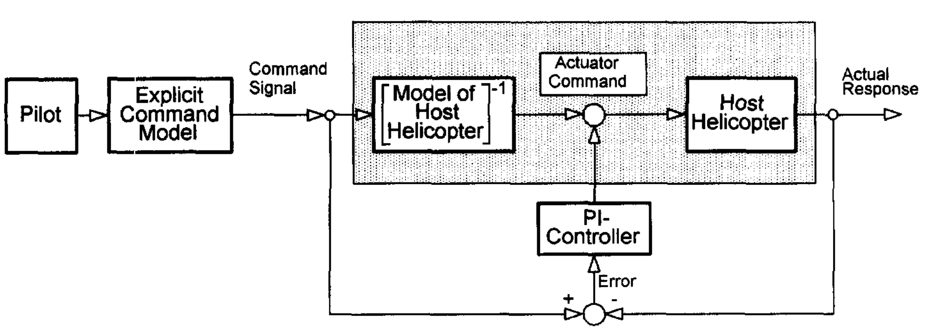

Fig. 16. Principle of explicit model-following control system for in-flight simulation.

unrealistic for real time processes. To eliminate the time delay problems and also to generate rotorcraft models that are accurate within a wider frequency range and in particular at higher frequencies, extended models with rotor DOF were applied for the feedforward controller. Since then, numerous flight tests with various command models and different objectives have confirmed the validity of the control system design for the BO 105 ATTHeS In-Flight Simulator and the high accuracy of its feedforward controller.[10]

To illustrate the significant improvement in model following fidelity, Fig. 17 compares the frequency responses for measured roll rate and model roll rate for two different cases: (1) the design of the model-following control system was based on an identified conventional 6 DOF rigid body model; (2) the control system was designed using an identified extended 9 DOF model with rotor DOF. For a perfect simulation, the frequency response should have a magnitude relationship of one and a zero phase angle difference for a broad frequency range. It is obvious that only the control system based on the model with rotor dynamics is close to this ideal situation, although there are increasing deviations for higher frequencies. The major question for a quality assessment is, whether the flight control systems performance meets the objectives for its intended application. Therefore, boundaries of so-called '*unnoticeable dynamics*' are also shown in Fig. 17. They were originally suggested by Hoh *et al.*[19] and later also applied by Anderson[2] for low-order equivalent modeling of high-order dynamic effects for fixed-wing aircraft. They characterize the limits, where pilots will not notice significant differences in flying qualities. Hamel and Jategaonkar[17] showed that these boundaries could successfully be applied for the validation of high quality flight simulators designed for pilot training (according to the highest requirements defined by level D of FAA). Applying these criteria to flight data from the helicopter In-Flight Simulator ATTHeS, Fig. 17 demonstrates that the frequency response, based on the extended model, lies within the boundaries. Good pilot ratings obtained for the simulation confirm the definition of the boundaries and indicate that they can also be used as a more general quality criterion for high bandwidth flight control systems as used by Hanke and Rosenau[18] for the DLR fixed wing aircraft In-Flight Simulator ATTAS.

8.3. EXAMPLE FOR IN-FLIGHT SIMULATION QUALITY

To demonstrate the quality and the potential of the system identification methods, the specific application to the ATTHeS control system design is discussed below in some detail.

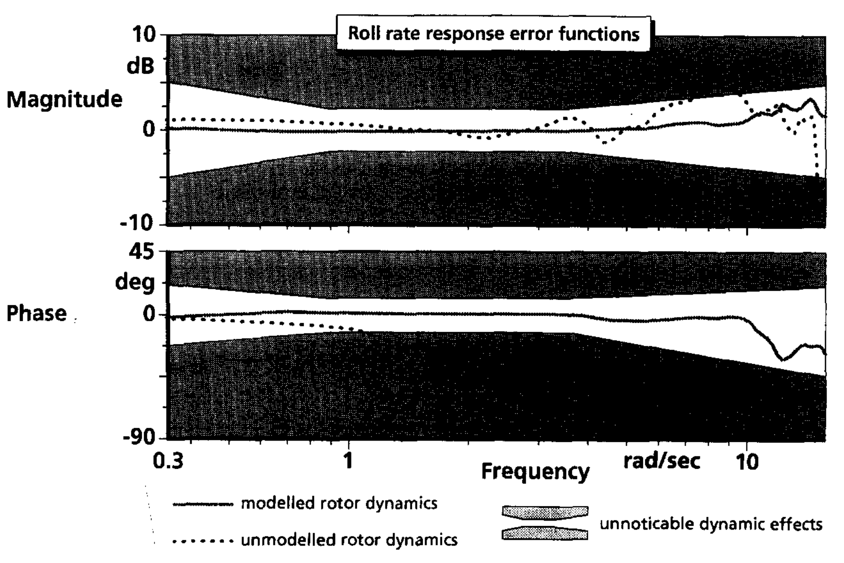

Fig. 17. Effect of unmodelled rotor dynamics on the in-flight simulation fidelity of the fly-by-wire BO 105 ATTHeS.

Three very different application examples are given: (1) the decoupling of the originally highly coupled pitch/roll motion of the BO 105; (2) the simulation of a different helicopter type; (3) an automatic position hold task.

(1) Figure 18 illustrates the achieved decoupling ratio of ATTHeS in comparison to the basic BO 105. It shows the measured rate responses due to longitudinal control inputs obtained from two different flight tests. For the basic BO 105 the left side of the figure shows that the roll rate is as high as the pitch rate. It clearly demonstrates the high coupling of the longitudinal and lateral motions. Then, the model following control system was engaged, where the command model was formulated to decouple the longitudinal and lateral responses. The obtained flight test result is given on the right side of the figure. As intended, the pitch rate follows the longitudinal stick input, but the coupling into the roll rate is significantly suppressed, reducing also the yaw rate response.

(2) To explore the potential of simulating the dynamics of different helicopters, a mathematical model of the Westland Lynx helicopter, provided by DRA, was used as a command model for the ATTHeS In-Flight Simulator. Various flight tests with Lynx experienced pilots were conducted. They rated the simulation as to be representative for the Lynx for both tested configuration, with the Stability and Control System (SCAS) engaged and disengaged. As example, Fig. 19 gives the time histories for a simulated turn maneuver for the Lynx helicopter configuration at 80 knots with SCAS engaged. As control variables the 'Lynx pilot' stick inputs are given together with the BO 105 actuator motions that are needed to make the BO 105 respond like a Lynx helicopter. In particular it should be noted, how the lateral control illustrates the typical different characteristics of the two helicopters: Lynx is an attitude command, BO 105 a rate command system. The simulation fidelity is demonstrated by the good agreement of the model and flight attitude angles.

(3) For the hover and low speed flight regime an accurate position hold above a ground fixed or moving object under wind and gust conditions is of special interest, e.g. for rescue missions. To demonstrate the capability of the model following flight control system, ATTHeS was equipped with an innovative measurement system for the hover position above a target. A video camera in combination with a computer for processing the optical information was used as an integrated sensor system for the measurement of the relative position of the aircraft to a target. As shown in Fig. 20 an additional feedback loop for the control system was defined, where deviations from the target were evaluated and converted to command signals. For the flight tests, a black square mounted on a car roof was used as target. The helicopter approached the car at a prescribed altitude with the Model Following Control System (MFCS) engaged in Position Hold Off mode. When the camera was focused

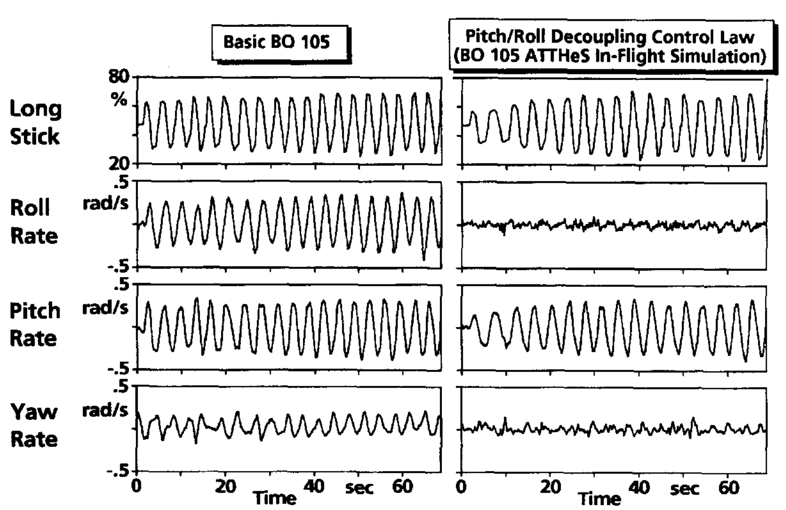

Fig. 18. Decoupling of the roll/pitch degrees of freedom.

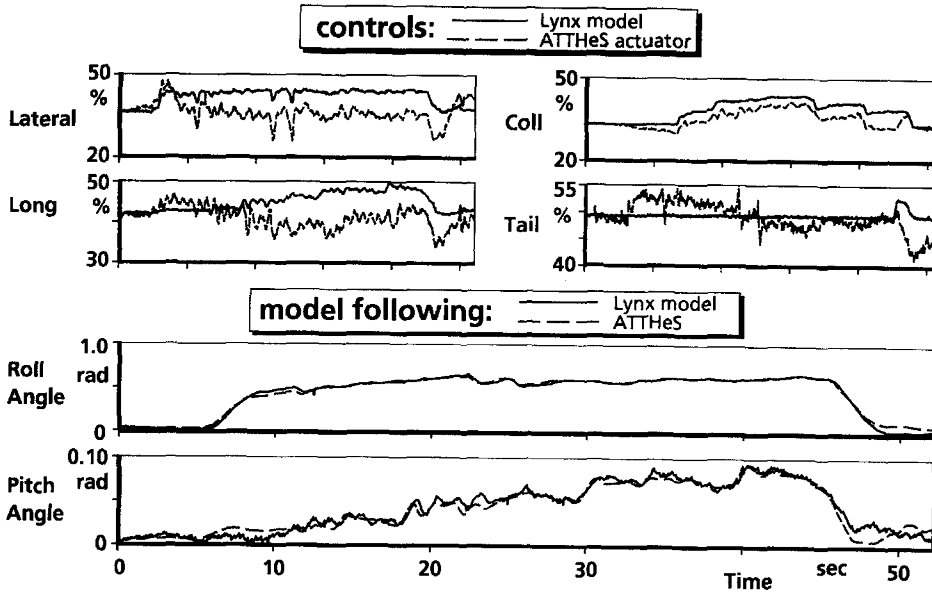

Fig. 19. In-flight simulation of Lynx helicopter with BO 105 ATTHeS.

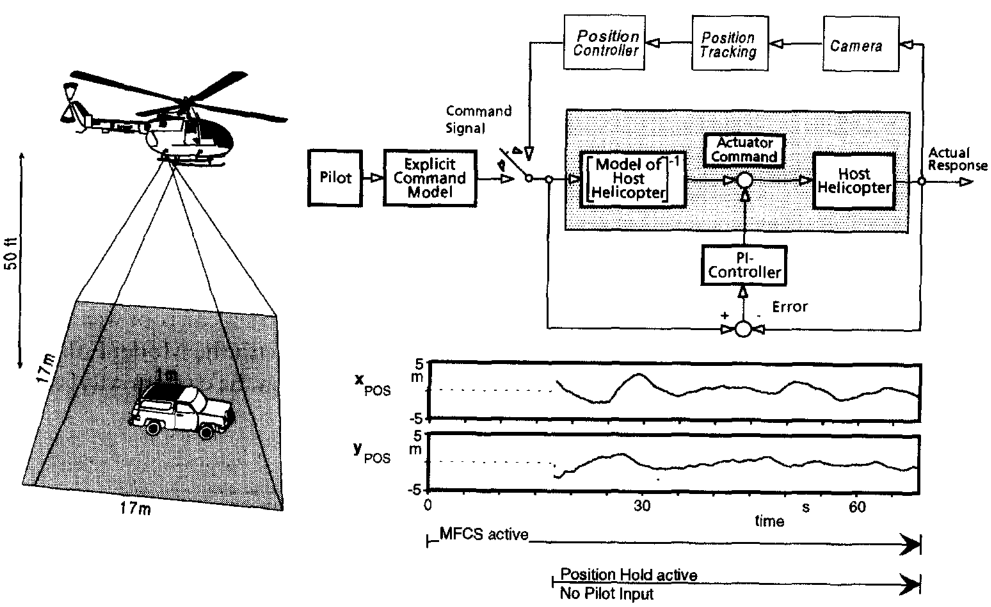

Fig. 20. In-flight simulation with additional feedback for an automatic position hold tracking task. Flight test set-up and results for a position hold over a car, moving in a full circle.

on the target, the Position Hold mode was engaged. Then, the vision based control system had to keep the helicopter above the target (left part of Fig. 20) with constant altitude and heading. In this mode, the pilot flew hands off. Actually, he could not even see the car below the helicopter. This stationary flight condition became a tracking task, when the car began to move. It was demonstrated that the helicopter still stayed above the target and followed the moving car successfully. Bouwer *et al.*[4] evaluated flight tests with 8 minutes duration and showed that the maximum relative position error was about 3 m for the moving car and about 1.5 m for the stopped car. As a representative example, Fig. 20 also gives the position error from a test run, when the car was driven in a full circle.

9. CONCLUSIONS AND RECOMMENDATIONS

From the present state-of-the-art in rotorcraft system identification the following conclusions and recommendations can be given:

(1) Flight tests must meet specific identification requirements, mainly with respect to control inputs and response information content, accurate trim, small perturbation responses, and repeated tests for data redundancy. Therefore, a dedicated flight test program for the identification purpose should be conducted.

(2) Data accuracy is the main prerequisite for a successful identification. Therefore, strong emphasis has to be placed on instrumentation, data quality checking, detection and correction of errors, and careful data processing. Correctly calibrated sensors easily provide the required accuracy, except for velocity measurements, which are still problematic, especially in the low speed regime.

(3) Identification techniques have reached a level of maturity that makes them powerful tools to support industry projects. From the techniques that have become standard, the least-squares method is appropriate for high quality data but usually it is applied to generate starting values. The more complex techniques in the time- or frequency-domain have shown to be suitable for rotorcraft identification. Both provide results of similar quality and it is more the skill and the experience of the analyst than the choice of the method that ensures the final success.

(4) For the most part, the identification of classical 6 DOF rigid body models can be treated as a routine task. The determination of higher order models with rotor degrees of freedom is also done successfully, but it requires more effort in aircraft instrumentation, data processing, model structure definition and identification.

(5) Higher order models can accurately characterize the higher frequency range. Therefore they are required for more demanding applications, such as high bandwidth control system design and high fidelity simulation. The need for reliable higher order mathematical models was clearly seen in the development of high quality In-Flight Simulators. The success of the simulation, demonstrated by the DLR ATTHeS, has shown that identification approaches can provide the required models needed for the control system design.

(6) As system identification extracts the rotorcraft characteristics from flight data of the existing aircraft, it is an ideal tool to generate a 'true' mathematical description of the vehicle dynamics. However, for specific applications, the presently used derivative models may be a too global representation of the aircraft dynamics. Approaches that combine the advantages of system identification and the advantages of other modeling approaches, such as generic simulation or wind-tunnel measurements, still have to be found. It is a major research challenge to develop ideas and guidelines for combining the individual techniques and to provide a powerful tool for accurately modeling the characteristics of a helicopter from its design phase through the flight testing phase until the final production.

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