SYNTHESIS REPORT

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PROJECT N°:	
TITLE:	**Improved Experimental and Theoretical Tools for Helicopter Aeromechanic and Ayeroacoustic Interactions - HELIFLOW**
PROJECT COORDINATOR:	**EUROCOPTER DEUTSCHLAND GmbH**
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SYNTHESIS REPORT CONTRIBUTIONS

Task 1 Pitch-Up

In this task experiments were conducted in the QINETIQ 24ft wind tunnel on the pitching moment changes caused by the interaction of the main rotor wake with the tailplane as a helicopter accelerates from hover to forward flight.
The tests were performed with a QINETIQ 4-bladed 3m diameter rotor mounted on the QINETIQ rotor rig (Figure 1-6). A fuselage with fin and tailplane was designed and built by AGUSTA[2] for these tests and mounted on the fuselage balance of the QINETIQ rotor rig. The fuselage had a generic streamlined shape to minimise local problems due to flow separation. The fuselage allowed the mounting of the tailplane in five positions. The tailplane options are shown in Figure 1-1. Pressures were measured on the tailplane and fuselage and the forces on tailplane, fuselage and rotor were measured along with rotor blade loads and control positions. CIRA provided flow measurements for task 1 using Particle Image Velocimetry (PIV), Hot Wire Anemometry (HWA) and hot film.

The most direct indication of pitch-up is given by the evolution of tail plane normal force with the increase of advance ratio. This is shown in Figure 1-2 for each tailplane position. At the most forward position, SYM1, the tailplane carries a considerable download even at the lowest advance ratio of 0.015. The further aft position SYM2 and then SYM3 are unaffected by the wake at the lowest speed but show entry into the wake as advance ratio increases. At position SYM4 the tailplane was slightly to the rear of SYM3 but mounted at mid-height on the fin. This is seen to further delay the build-up of load when advance ratio increases.
A sample view of velocities derived from a PIV image is shown in Figure 1-3. In this view the instantaneous position of the vortex shed by the main rotor tip is directly above the position, which had been occupied by the tailplane leading edge, implying a significant interaction.
Analytical methods used by partners fell into two categories: detailed aerodynamic prediction methods and more empirically based flight mechanics codes. The abilities of the latter codes to predict the tailplane loads were compared, sample comparisons being given in Figures 1-4 and 1-5. Empirical and modelling features of the codes were refined during the project and a reasonable indication of pitch-up is evident in the results. The flow prediction methods indicated the behaviour observed in the wind tunnel but with some detail differences, e.g. precise location of the wake, having a significant effect on the results. The pitch-up test data provides valuable validation data for use during the further development of these methods.

The experience of making these tests, the analysis of the data, and the refinement of the prediction methods will assist in quantifying the pitch-up effect on future helicopter designs.

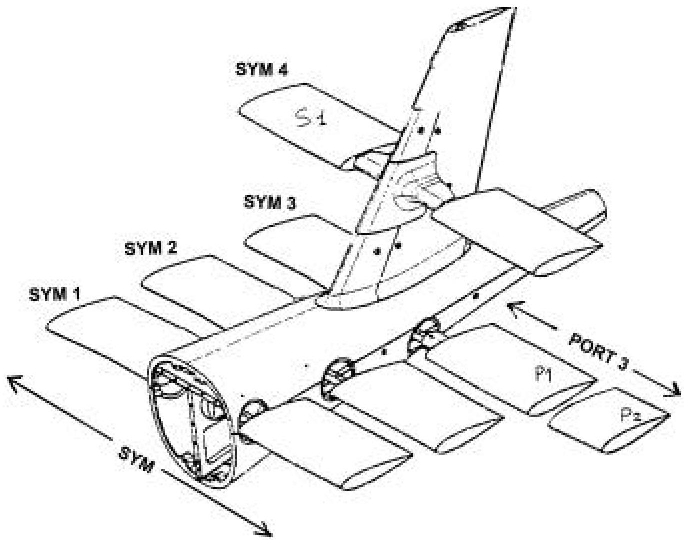

Figure 1-1 Tailplane configuration options

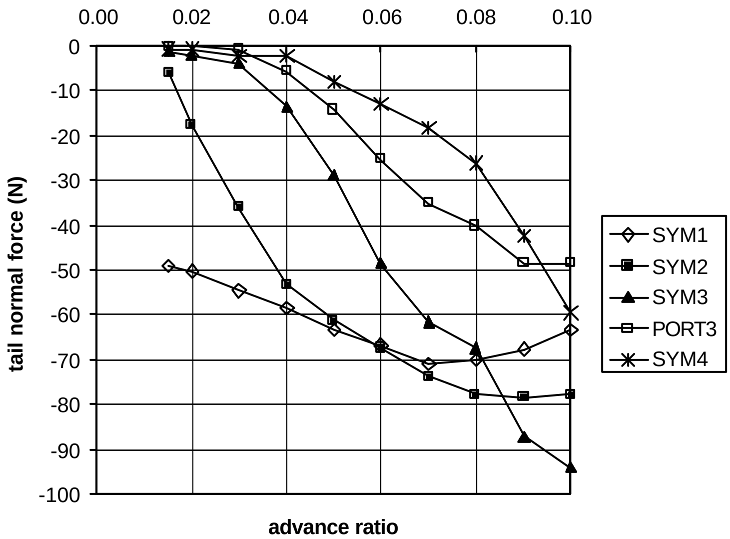

Figure 1-2 Tailplane normal force variation with advance ratio (SYM = total of port and starboard tailplanes)

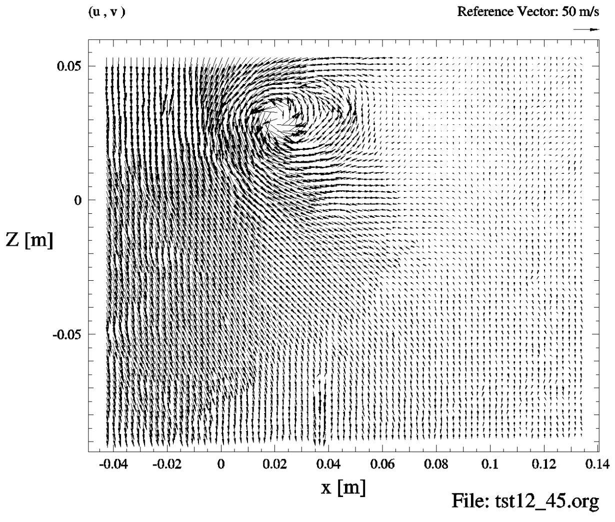

Figure 1-3 Sample velocity field from PIV

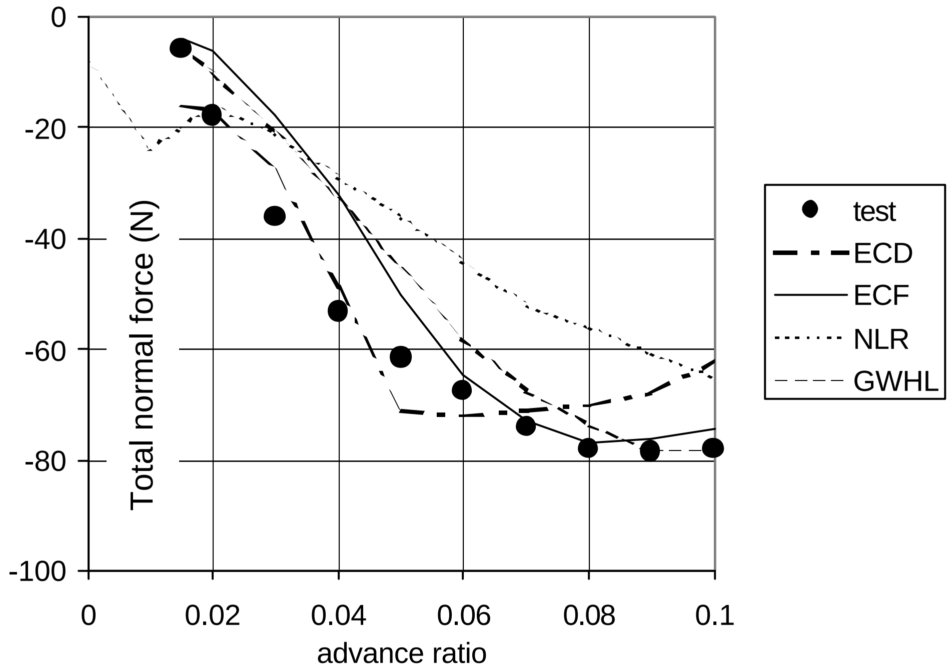

Figure 1-4 Total normal force for tail SYM2 (middle fuselage position)

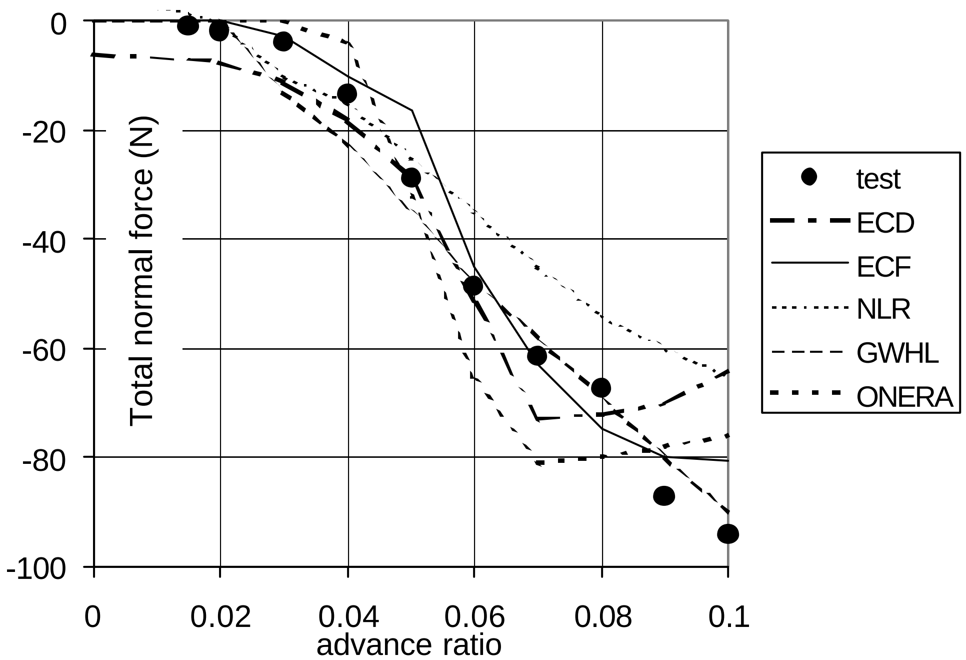

Figure 1-5 Total normal force for tail SYM3 (rear fuselage position)

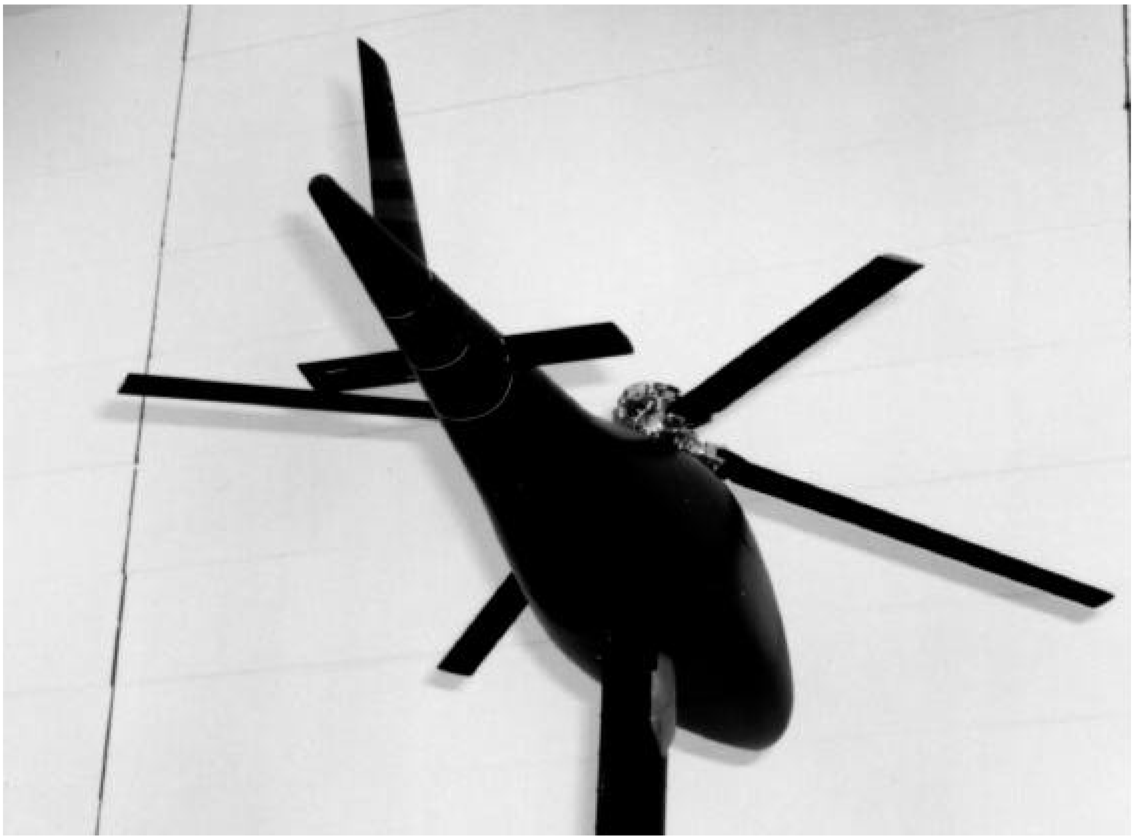

Figure 1-6 Wind tunnel model with AGUSTA fuselage on QinetiQ rig

Task 2 Quartering Flight

The experimental activity of task 2 was wind tunnel testing in the QINETIQ 24ft wind tunnel of a combination of main rotor and an independent tail rotor. The QINETIQ main rotor was the same as employed in Task 1 and the tail rotor was produced by WHL. The tail rotor was mounted on an independent stand to allow it to be moved to locations with the worst quartering flight interactions. A schematic of this low speed interaction is given in Figure 2-1.

Tests were conducted during hover and at slow forward flight speeds, up to advance ratio of 0.095. The initial survey with the tail rotor positioned 50^{o} and 60^{o} away from the regular downwind position identified clear interactions. Further tests concentrated on the 60^{o} azimuth position. Data were acquired for many combinations of main rotor thrust, advance ratio, tail rotor height, and tail rotor rotation both top-blade-forward and top-blade-aft. For most test conditions the variations of tail rotor thrust and torque with tail rotor pitch were surveyed.

Typical results for torque and thrust are shown in Figures 2-2 and 2-3 respectively. The values are plotted as the ratio between the torque or thrust in the interaction condition and the corresponding quantity for the stationary isolated tail rotor with the same pitch setting. The curves represent the 2 tail rotor height locations and the two rotation directions (TBA denoting Top-Blade-Aft and TBF for Top-Blade-Forward). The low position tail rotor with Top-Blade-Aft rotation suffers a reduction of thrust up to an advance ratio of 0.05 and then a sudden recovery when speed increases to 0.06. In this speed band the low Top-Blade-Forward rotor continues to loose thrust before recovering at a higher advance ratio. The changes are less severe for the high

position tail rotors. These results are general for almost all tail rotor pitch settings, as shown for thrust in Figure 2-4.
Eurocopter used interpolation between the experimental data to find the tail rotor pitch which would be required to maintain a constant 80N thrust. Figure 2-5 shows the result for tailplane high and low positions. At an advance ratio of 0.05, corresponding to about 21kt for a full scale helicopter, 4^{o} extra pitch is required to maintain thrust. As this represents 14% of control range this would not give acceptable handling characteristics.
Partners refined and evaluated codes similar to those employed in Task 1, with the necessary addition of a tail rotor. A comparison of thrust predictions against experiment is shown in Figure 2-6. It is apparent that some codes predict the correct initial value at low advance ratio, some are closer to the data at the highest advance ratio. However, none successfully predict the complete characteristics of the experiment either in terms of the magnitude or the advance ratio at which the sudden change occurs.
Further developments of analysis methods are required before detailed predictions can be made of the quartering flight phenomenon. However, the wind tunnel test techniques have been demonstrated successfully and these provide a route to the provision of analysis data and quantification of the quartering flight effects.

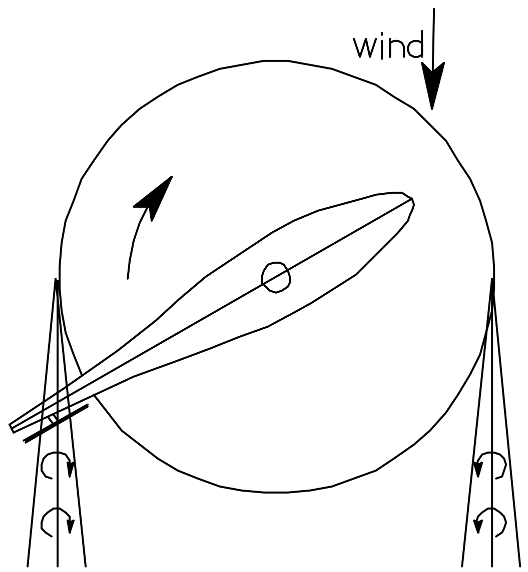

Figure 2-1 General schematic of quartering flight interaction

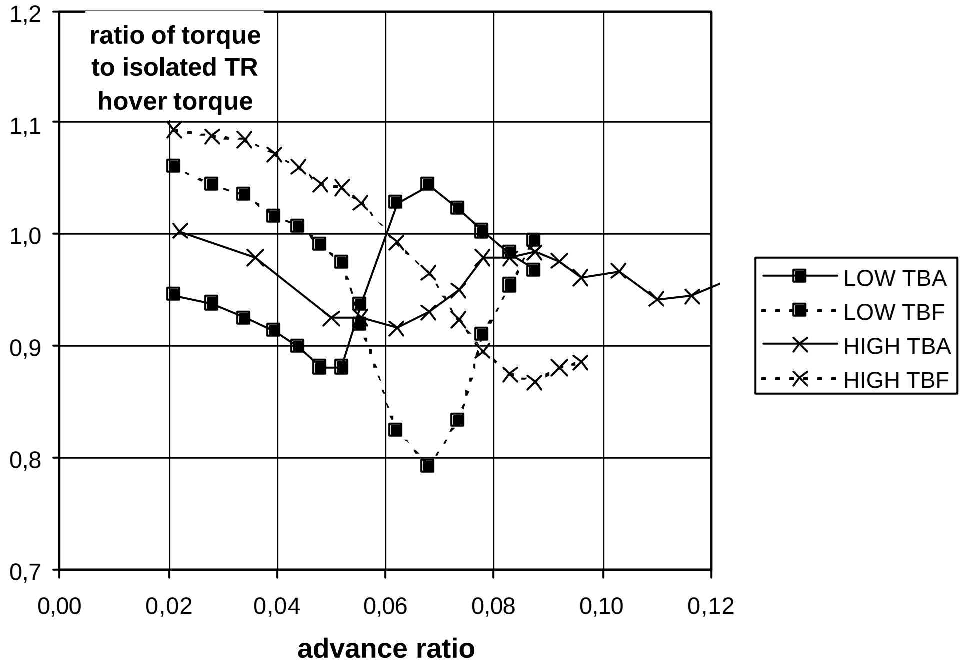

Figure 2-2 Torque of tail rotor at 60^{o} azimuth position

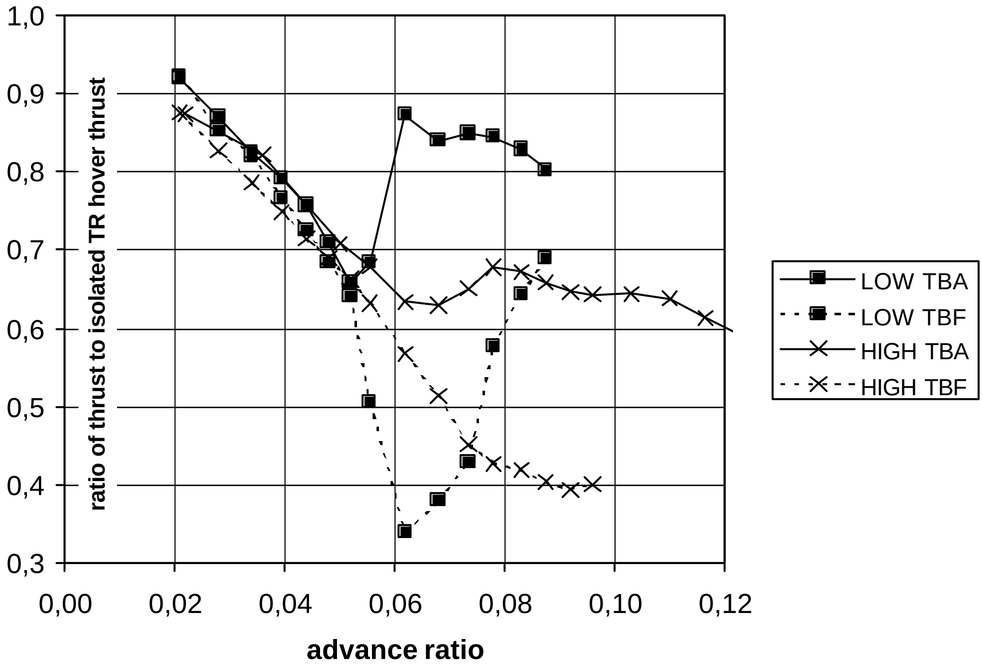

Figure 2-3 Thrust of tail rotor at 60^{o} azimuth position

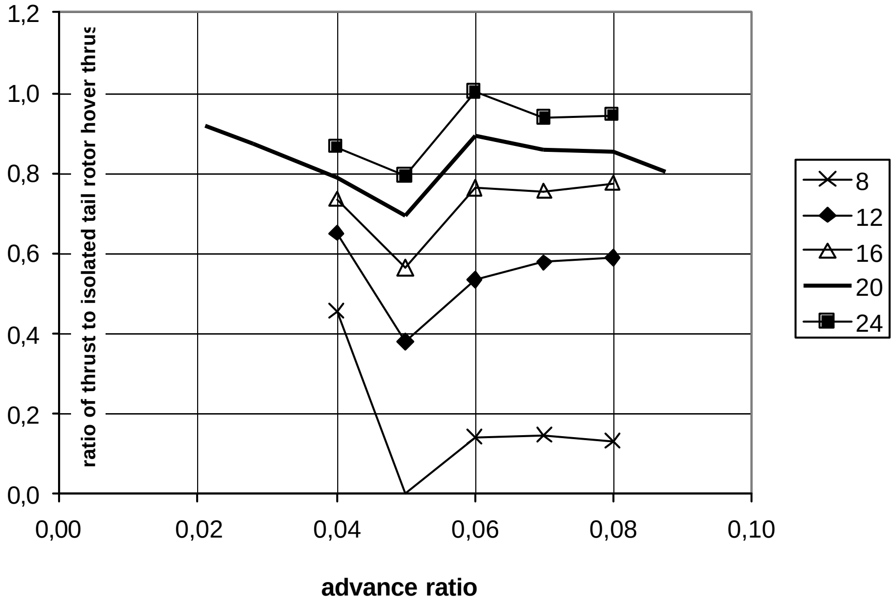

Figure 2-4 Thrust of tail rotor at different pitch settings, TBA, low, 60° azimuth position.

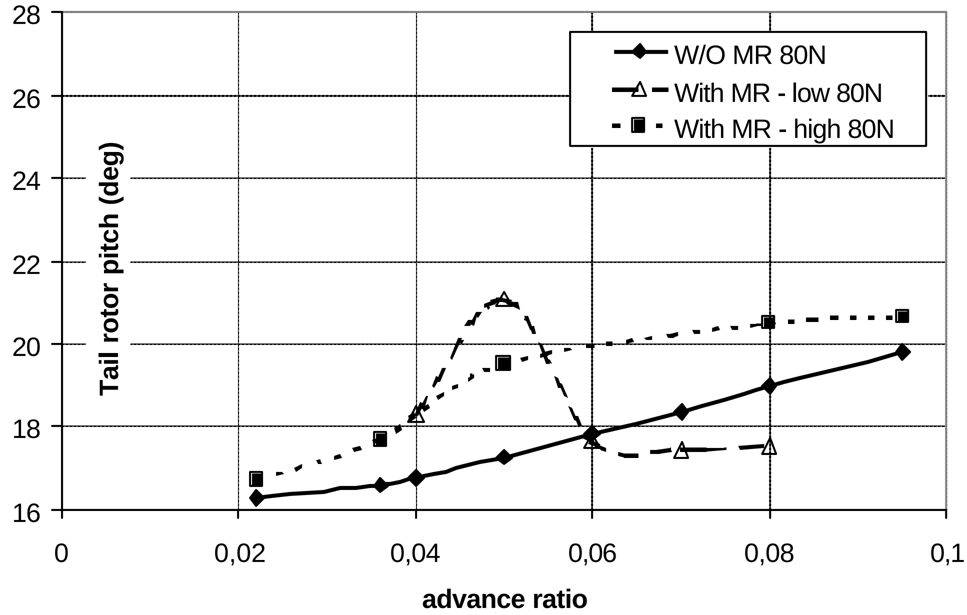

Figure 2-5 Tail rotor pitch required to maintain constant thrust (60° azimuth, TBA)

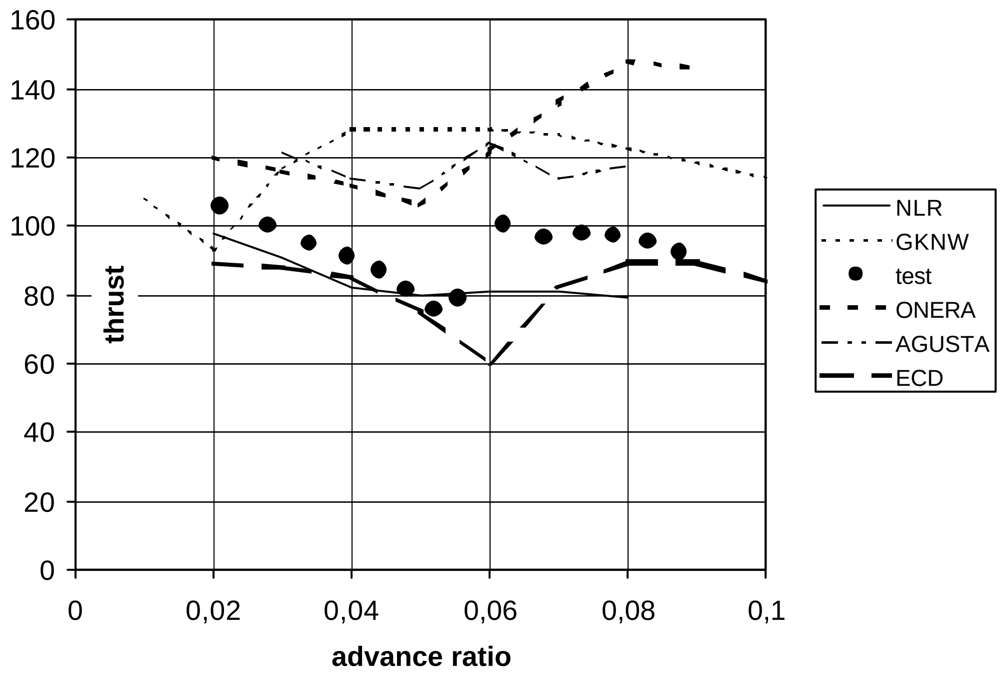

Figure 2-6 Comparison of predictions with experiment, pitch=20°, low tail rotor TBA at 60° azimut

Task 3 Sideward Flight Tests

The major objective of task 3 within the HELIFLOW programme was to design and build a scaled main rotor/tail rotor powered Bo105 model and to test and trim this model in the DNW-wind tunnel in sideward flight in and out of ground effect thereby obtaining a data basis to enhance the understanding of interactional phenomena on helicopters in quatering flight.

The new model design and construction concentrated on the tail rotor unit and its integration into the existing MWM-module with main rotor and fuselage. It was jointly undertaken by ECD and DLR.

The largest test section of the DNW-LLF in an open configuration with a ground plate and tangential blowing provided an ideal low interference test setup for in ground effect flight simulation.A trim subroutine from an aeromechanic computer code of ECD was software implemented into the model and wind tunnel measuring setup.With this procedure it was possible to achieve a balance of all the forces and moments (Fig. 3.3) on the model for each flight condition within 1 to 3 minutes.

LLS (Laser Light Sheet) and PIV (particle image velocimetry) measurements were done in one plane below the advancing side of the main rotor in order to determine the magnitude and core location of the ground vortex. A much greater sensitivity of the location of the ground vortex core with advance ratio than with rotor height above ground could be observed.(Figs.3.1 and 3.2) Using these measurements NLR showed that their computer code OUTWASH can be easily tuned and adapted to simulate the correct magnitude and location of the ground vortex.

The measured control angles and attitude angles of the trimmed Bo105 helicopter model for three different rotor heights above ground revealed a clear influence of the ground vortex on these angles. The advance ratios and heights above ground at which this ground vortex effect occurs correlate generally very well with the Princeton data of the extension and location of the ground vortex below an isolated rotor.(Figs.3.4 and 3.5)

The out of ground trim angles measured in the wind tunnel differ by 1 to 2 degrees from the flight test data and the theoretical values but show in general the same trend with advance ratio.

The present wind tunnel campaign has established a useful data base for trimmed sideward flight analysis and paves the way for future use of the wind tunnel as a powerful tool for analysing helicopter aeromechanic problems.

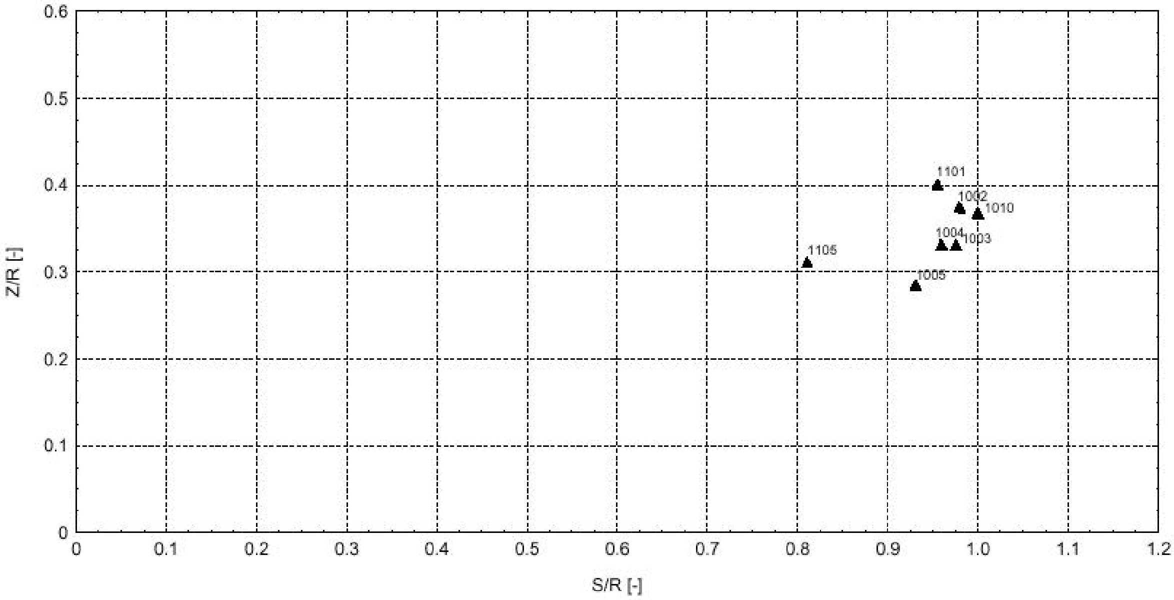

Fig.3.1: Measured ground vortex core positions in LLS-plane for varied rotor height

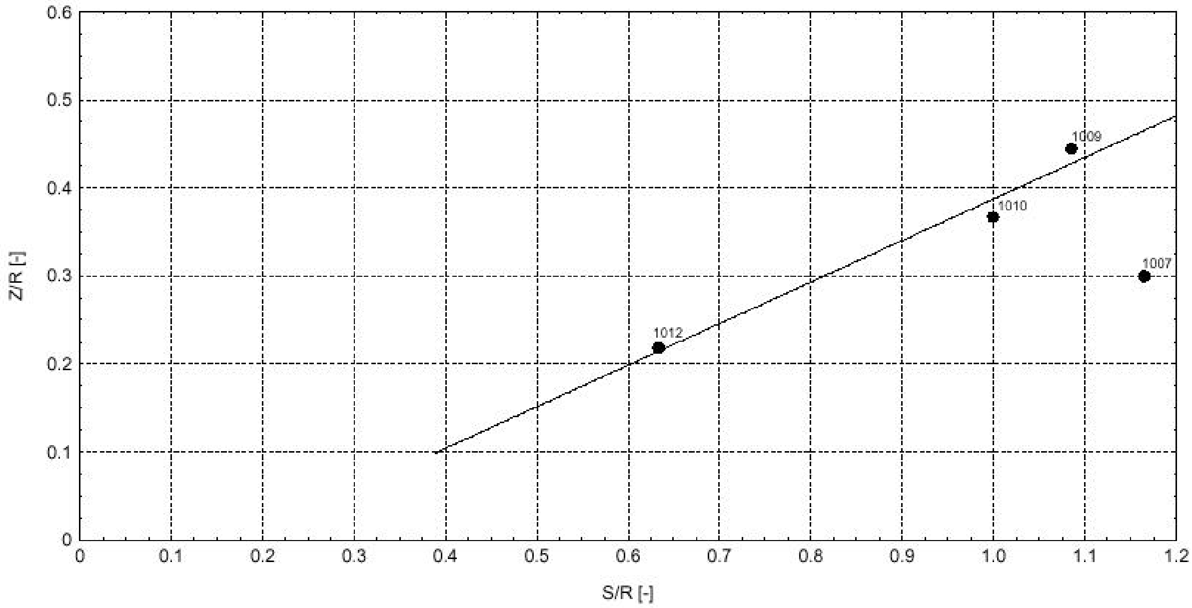

Fig.3.2: Measured ground vortex core positions in LLS-plane for varied air velocities

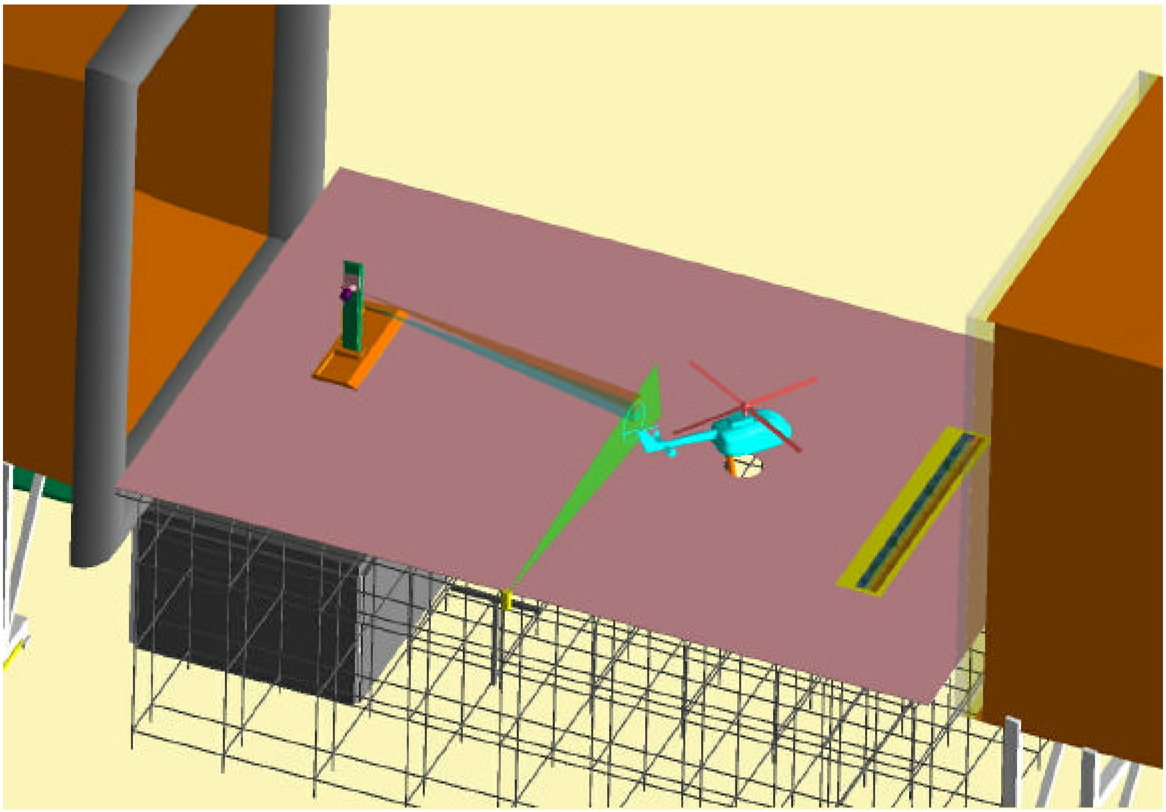

Fig. 3.3 PIV measurement configuration

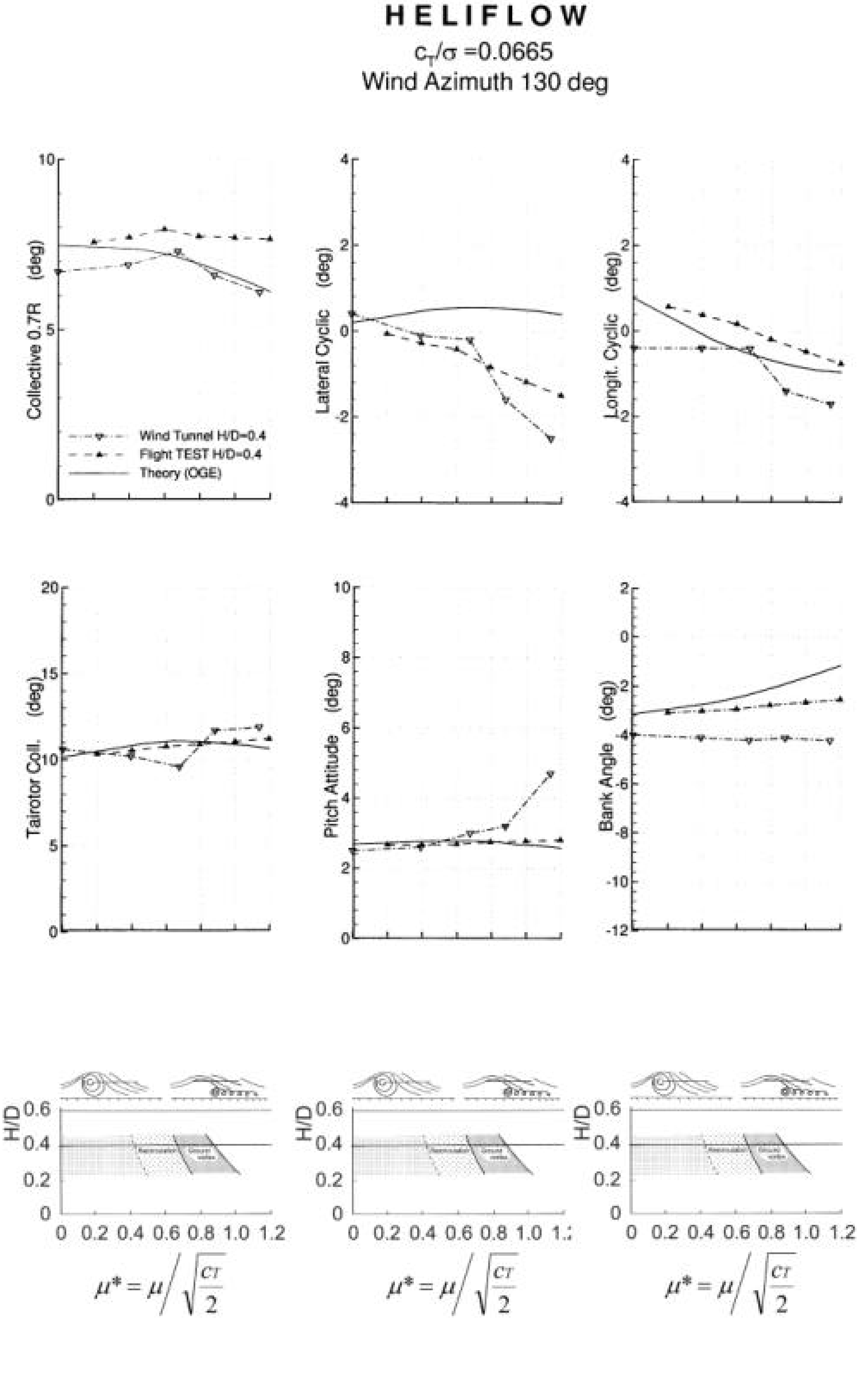

Fig.3.4 Helicopter control and attitude trim angles for wind azimuth of 130^0 at H/D = 0.4 (wind tunnel test,flight test and OGE theory)

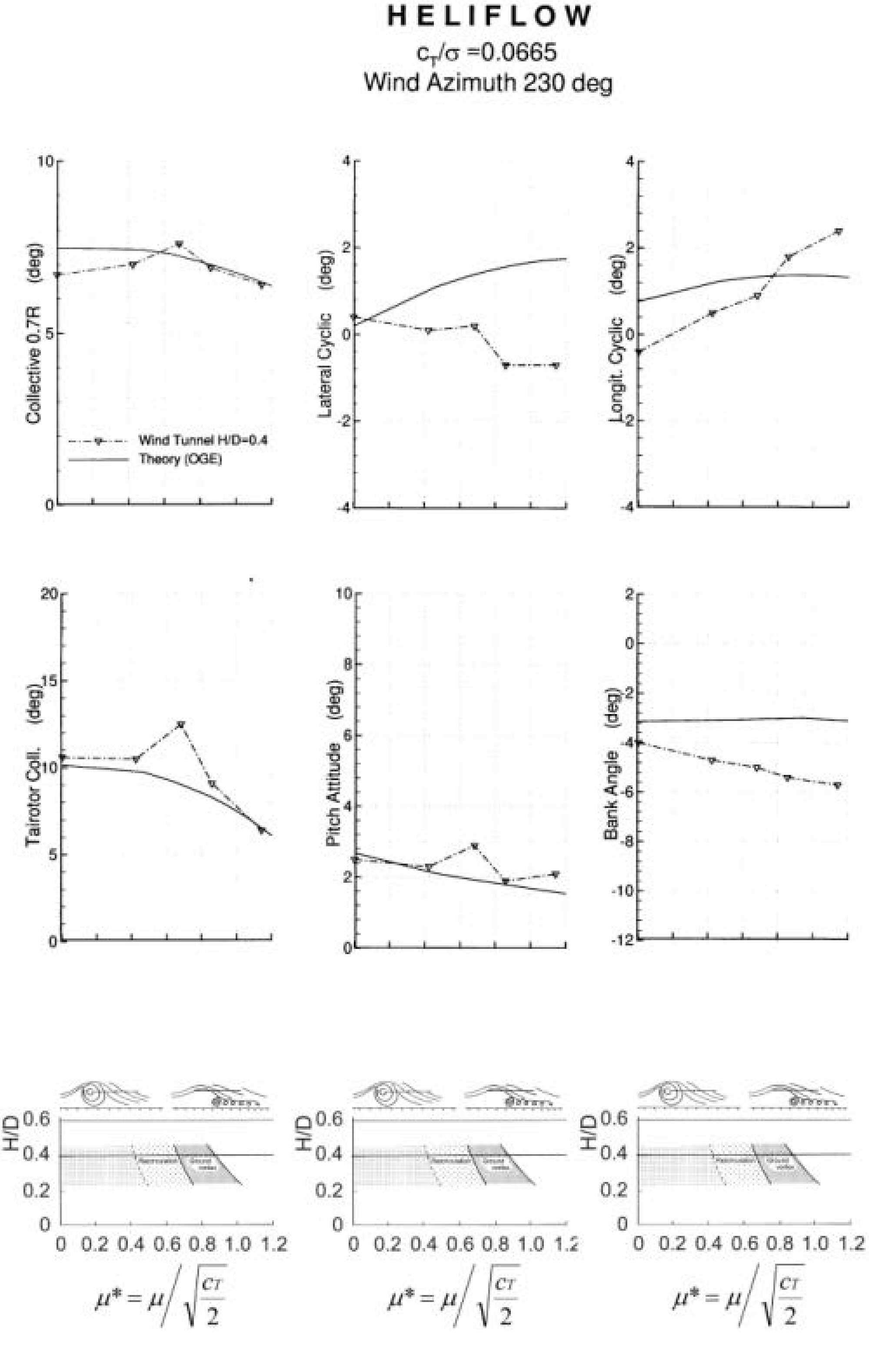

Fig. 3.5 Helicopter control and attitude trim angles for wind azimuth of 230^0 at H/D = 0.4 (wind tunnel test,flight test and OGE theory)

Task 4 Tail Shake

In some flight conditions, the helicopter exhibits unpleasant low frequency vibrations at the frequency of an eigenmode of the airframe, this is called "tail shake". This phenomenon involves both dynamics and aerodynamics. This phenomenon has been investigated in wind tunnel on a Dauphin powered model (Fig. 4.1).

Synthesis of the Dauphin Tail-shake problem encountered in flight

In order to prepare the wind tunnel campaign to be performed in S2Ch wind tunnel (ONERA facilities), a synthesis of the aircraft configurations and flight tests conditions in which Dauphin showed some tail-shake problems has shown that the most severe effects was, by far, the lateral vibrations of the tail boom at a natural frequency of 12 Hz (close to the 2/rev). This was obtained in level flight between 80 and 140 kt, and in descent flight. Among the several aerodynamic modifications tested in flight, the best solution consisted in the addition of a hub cap and a pylon fairing, to the basic configuration : the configuration so obtained is that of the Serial Dauphin.

Technical solution definition

In order to simulate the first lateral bending mode of the tail boom, the final solution retained by ONERA is based on the use of two masts:

- ☞ the first mast will support the fuselage, the motor for the driving of the rotor and the rotor itself ;
- ☞ the second mast, fixed on the same incidence and sideslip ring, will support the rear part articulated around a vertical axis which is just under the horizontal stabilizer: a torsion flexibility can be adjusted on the required frequencies (between 1Ω and 2Ω) or stiffened depending on the case test.

Test conduct

The tailshake tests have been conducted during three months (Nov. 98 to Jan. 99), in the ONERA S2 wind tunnel located in CHALAIS-MEUDON.
The wind tunnel tests have been performed in two sequences: the first one, devoted to loads and unsteady pressure measurements on the tail, and the second one, devoted to LDV measurements in the vicinity of the tail.
An extensive experimental database, corresponding to the achievement of the whole operational test matrix initially planned, has been set up.

Analysis of test results

The analysis performed by all the partners confirms the first tendencies observed during the wind tunnel test.
The following conclusions have been highlighted:

- ☞ **the erratic tailshake phenomenon has been observed** from time responses of the tail gauges, showing that puffs (varying amplitude) appear in a random way making the phenomenon time dependent ;
- ☞ the results obtained on the prototype configuration give the **same trends as observed in flight**: a maximum tailshake effect around 110 kts in level flight, a tailshake effect increasing with the rate of descent (Fig. 4.2);
- ☞ **the influence of the pylon fairing**, added to the engine cowling of the full scaled helicopter to solve tailshake, **is tremendous** with a significant reduction of the level of vibration at the tail ;
- ☞ the calculation of an integrated fin force based on the pressure transducers signal shows an excellent correlation with the tail gauges measurement ;

- ☞ the aerodynamic modification effect identified in the WTT are consistent with the flight test results.

In the case of the DAUPHIN, the origin of the excitation is aerodynamic, coming from the main rotor hub and the engine cowling. This wind tunnel tests campaign has shown that the rotation of the hub is at the origin of the phenomenon. The adjunction of the pylon fairing reduces the wake and moves it down in order to reduce the aerodynamic excitation on the fin.

Tail shake phenomenon understanding has been improved by this HELIFLOW program demonstrating that tail shake is always present, with an acceptable or unacceptable level of discomfort. The existence of a tailboom lateral mode which frequency is closed to a rotor harmonic will lead to an increase of the tail shake phenomenon. If the tail shake level is unacceptable, aerodynamic modifications can solve the problem by changing the structure and the position of the wake.

Wind tunnel test methodology

The different parameters investigation performed during this campaign has lead to the definition of a wind tunnel test methodology which can be applied to future helicopter development.

Such a methodology have different advantages:

- ☞ A low cost compared to flight test investigations ;
- ☞ The capability of analyzing several modifications with simple tools ;
- ☞ No specific model required for tail shake investigation: the aerodynamic one can be equipped with unsteady pressure transducers and accelerometers.

Nevertheless, only discrepancies between aerodynamic configurations and modifications can be investigated in wind tunnel. Such tests can not give absolute value for quantifying the magnitude of the tail shake.

In the framework of HELIFLOW program, a wind tunnel test campaign and his analysis have demonstrated the capacity of wind tunnel to detect and investigate tail shake for future helicopter development, with low cost and simple model.

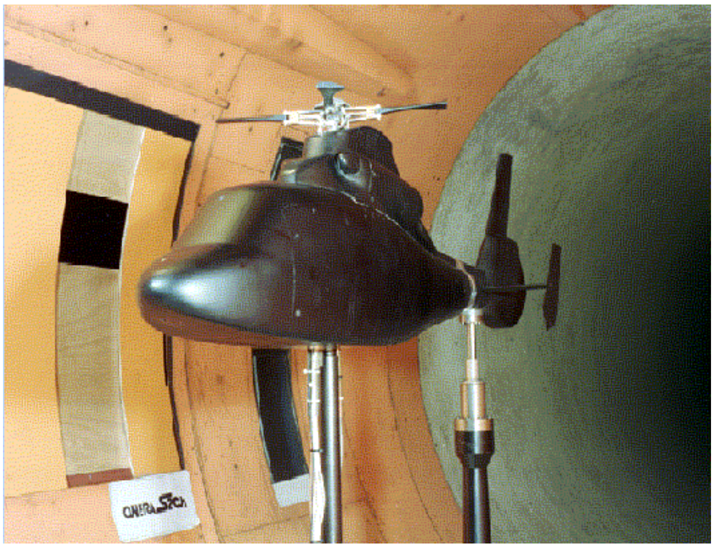

Fig. 4.1 EC/ONERA model in the Chalais-Meudon wind tunnel

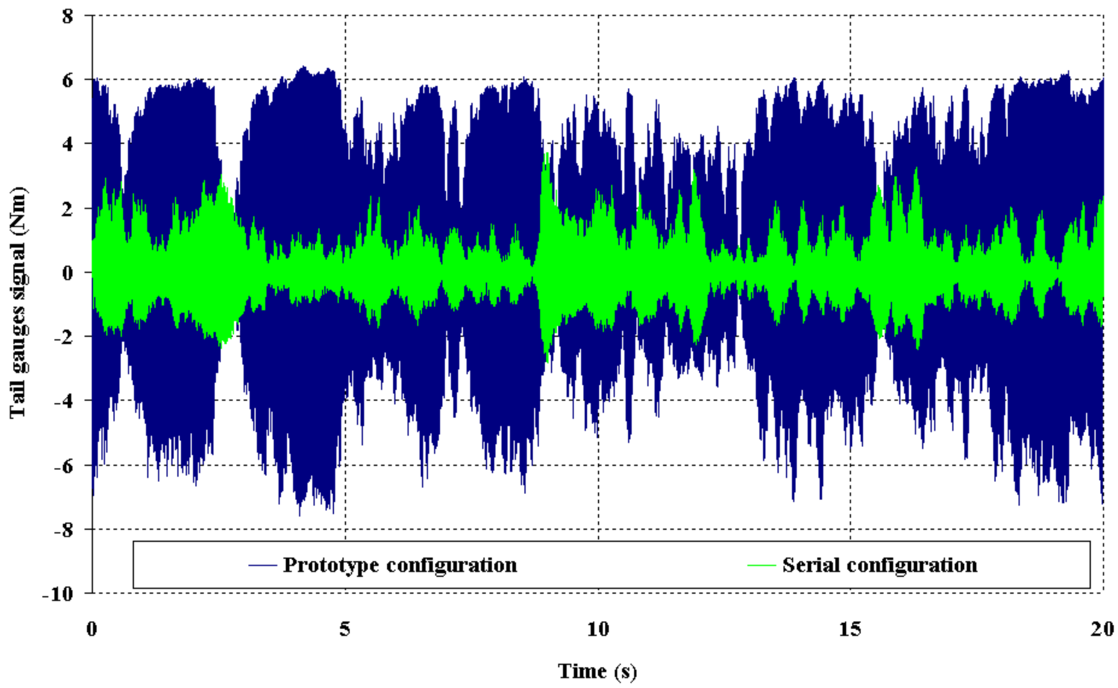

Fig. 4-2: Tail gauges time signal for level flight condition at 110 kts

Task 5 Tail Rotor Noise Investigations

Within the last two decades, the physical understanding of helicopter main rotor noise and the capability to predict its noise radiation generated by various aeroacoustic source mechanisms has significantly been improved. Research on tail rotor noise, however, which is dominating the overall noise radiation in certain flight conditions, e.g. at take-off, has been ignored for a long time. Due to the much higher rotational speed and because of the strong perturbation of the tail rotor inflow by the main rotor wake, the tail rotor radiates noise with an annoying higher frequency content, which is very difficult to predict. The objectives of this task were to improve the understanding of the aeroacoustic phenomena and the prediction capability of tail rotor noise, to identify potential noise palliatives, and to validate the theoretical results by wind tunnel experiments.

The original plan foresaw the use of the BO 105 wind tunnel model for both the Task 3 (sideward flight) and the Task 5 (tail rotor noise) tests. The wind tunnel model was a 40%-scaled BO105 helicopter model. The main rotor was a geometrically and dynamically scaled four bladed hingeless BO105 rotor operating at full-scale tip Mach number. It was mounted on the Modular Wind-tunnel Model (MWM) test rig (of partners DLR and ECD) and driven by a hydraulic motor with a maximum power of 130 kW. The tail rotor unit consisted of the 40%-scaled, see-saw type, two bladed BO 105 rotor also operating at full scale tip Mach number, the gearbox, the collective pitch control, and the slip ring system for data transmission. The tail rotor was powered by a separate hydraulic motor (placed inside the end of the tail boom) providing a maximum power output of 13 kW.The model was equipped with two six-component balances (one for the main rotor and one for the fuselage incl. the empenage). Besides the usual strain gauges for blade flapping, lagging and torsional loads, one main rotor blade was instrumented with 25 piezoresistive sensors (Kulites) and one tail rotor blade with 36 sensors of the same type for the measurement of unsteady blade surface pressures.

A final test plan was jointly developed by all partners. Measurements were planned for simultaneous acquisition of unsteady blade surface pressures on main and tail rotor, sound pressures on a large plane underneath the model, aeroelastic blade deflections and rotor operational and performance data. Furthermore, it was intended to determine the tail rotor inflow characteristics, especially the flow field of young main rotor blade tip vortices, by particle image velocimetry (PIV) provided by DNW. In a pre-test phase, tunnel background noise and reflection tests were foreseen in order to guarantee high quality aeroacoustic measurements. In order to study the aerodynamic and aeroacoustic effects of main rotor / tail rotor (MR/TR) interaction and of empennage/tail rotor interference, it was intended to repeat the measurements at nominally identical flight conditions for the following three test configurations (a) isolated main rotor, (b) isolated tail rotor, and (c) main-tail rotor operation. The force trim procedure was selected to trim the model for matching as closely as possible the forces and hub moments of main and tail rotor of the BO 105 helicopter in flight.

To demonstrate and substantiate tail rotor noise reduction potentials, a number of noise reduction means were included in the test plan, like reduction of blade tip speed, reduction of blade air load (helicopter mass), change of tail rotor sense of rotation (forward moving blade up), and tail rotor operating in tractor mode (instead of pusher mode).

The Task 5 wind tunnel tests had been scheduled for November/December 2000, immediately following the Task 3 tests. All necessary data acquisition hardware and software was made available on the test site to collect, analyse, and store the rotor operational data, the blade surface pressures, and the acoustic data. Figure 5.1 shows a photograph of the BO 105 wind tunnel model mounted in a (3/4)-open-jet test section featuring a large ground plane for ground effect investigations during Task 3 testing. These tests were successfully conducted and almost completed, when during a final checkout of the model without wind, the tail rotor test rig was unexpectedly destroyed. Investigations showed that this severe damage of nearly all tail rotor components was caused by a sudden break-down of the centrifugal force retainment element of one blade in the tail rotor hub, the separation of this blade, and the resulting severe unbalance. Consequently, the Task 5 tests had to be called off and the tail rotor noise prediction results of Subtask 5.5 could not be validated.

One of the objectives of the task was to obtain validated improved prediction codes for main rotor/tail rotor noise and to identify means of reducing the noise. Three main areas were identified so as to reach such an objective:

- Improvement of the prediction capability for the unsteady aerodynamic blade surface pressure on the tail rotor (Subtask 5.5.1);
- Improvement of the tail rotor noise prediction capability (Subtask 5.5.2);
- Improvement of the prediction of main and tail rotor noise radiation (Subtask 5.5.3).

A revised work programme was released in 1999 taking into account the delays that had taken place up to that point and the new planning for the wind tunnel tests. Despite the limited funding devoted to aeroacoustic predictions, the partners were committed to completing a very ambitious work programme. Two pre-test cases were chosen (level flight and 12°-climb) in order to compare the various predictions and to provide information to the test team prior to the wind tunnel tests. Standards were introduced for providing aerodynamic results, aerodynamic inputs for aeroacoustic codes (aeroacoustic input), and aeroacustic results. These standards were intended to simplify the communication among the various partners allowing everyone to perform aeroacoustic calculations using aerodynamic data sets provided by other partners.

Despite the extension of the programme due to the problems with the experimental part of the programme, noticeable delays persisted in the prediction activities both due to the limited funding and to the inherent complexity of the modelling activities. What follows is intended to give a general overview of the major achievements of the single partner and the co-operative efforts among the participants. Most of the partners (QINETIQ, DLR, ONERA, NTUA, ECF, ECD) intended to perform both aerodynamic and aeroacoustic preditions while AGUSTA and CIRA focused on the aeroacoustic methodologies while depending on the other partners for the aerodynamic data set. Figure 5.2 shows the free-wake simulation of the main and tail rotor in level flight and in a 12°-climb. The complex interactional environment around the tail rotor is clearly visible. Figure 5.3 shows the resulting sound field. Clearly level flight is a much noisier condition. It was also found that in the 12°-climb condition the tail rotor is the primary noise source.

The pre-prediction effort was generally completed by all participants of the project inspite of some setbacks which led to delays in completion of the computations. The obtained results form a useful basis to compare the code capabilities of the individual partners and plausibility checks. The more ambitious goal of validation and improvement of the codes could not be achieved due to failure of the test rig with the consequence of non-availability of any experimental data. Nonetheless, most of the participants have learned important lessons about their prediction capabilities and have identified areas which require further developments.

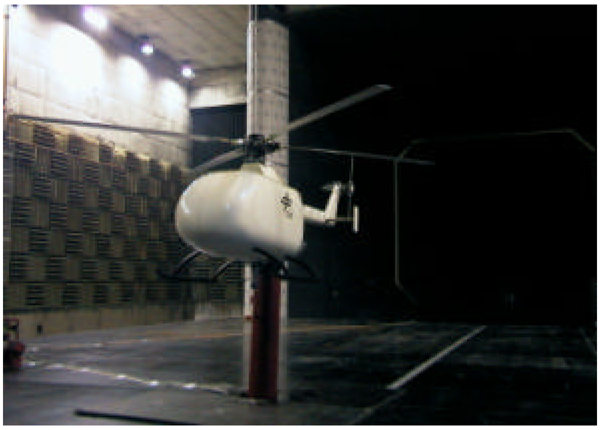

Figure 5.1: BO 105 wind tunnel model mounted in DNW open test section over a ground plane (for Task 3 tests)

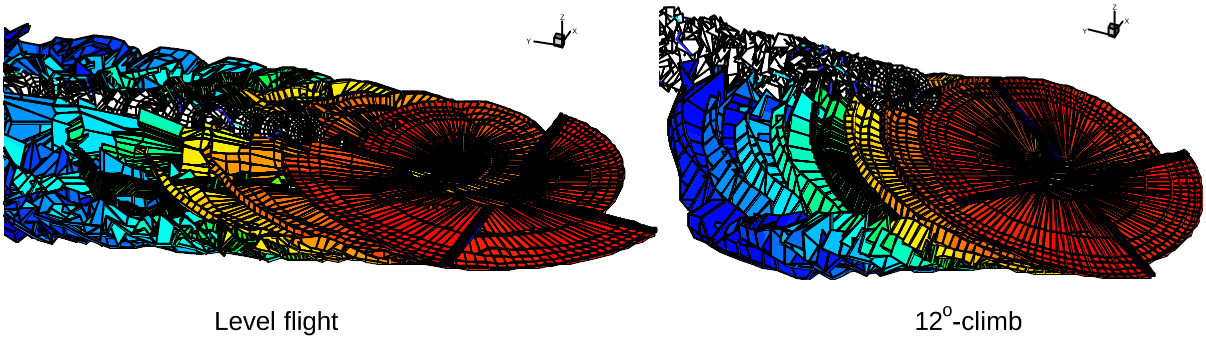

Figure 5.2: Coupled free-wake predictions of main- /tail interactions in level flight and 12°-climb.

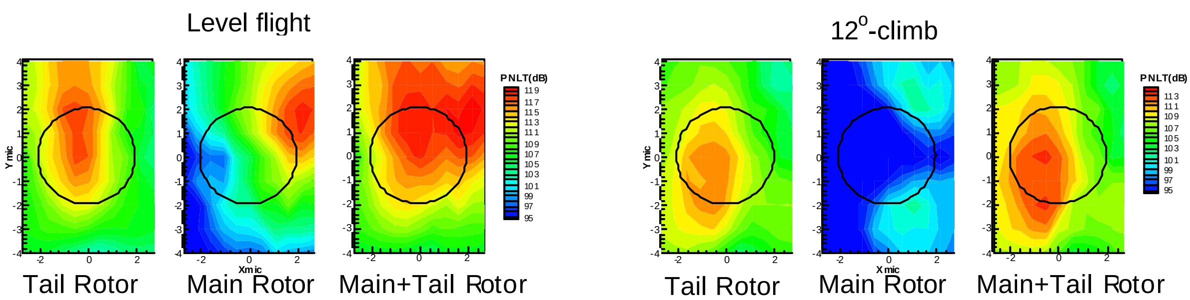

Figure 5.3: Sound field prediction below the helicopter in level flight and 12°-climb.

Task 6 Helicopter Rotor Transfer Functions for Unsteady Inputs on the Controls

Task 6 of Brite Euram HELIFLOW project is dedicated to the measurement of isolated helicopter rotor dynamic transfer functions and to the validation and improvements of flight mechanics codes. The main objective of this HELIFLOW task 6 is to obtain a transfer functions data base for an isolated rotor in a frequency range interesting for handling quality studies. In particular to look at the off axis responses and to validate and improve flight mechanic codes.

The wind-tunnel tests have been performed in the ONERA S1MA wind-tunnel in December 1997. For these tests, a 4-bladed rotor of 4.2 meters in diameter is used and sinusoidal inputs are introduced successively on collective, lateral and longitudinal rotor pitch angles. The excitations have a maximum of 1° in amplitude and the frequencies range between .5 and 15Hz (rotor RPM : 16 Hz) and up to 22 Hz for one configuration. In order to obtain the transfer function of the rotor balance corrected of the inertia effects, a dynamic calibration of the balance has been performed. The inertia correction is obtained with the measurements of 8 accelerometers located on the non rotating part of the rotor hub (Figure 6.1).

Figure 6.1 : Balance dynamic calibration

A specific program has been developed in order to introduce sinusoidal inputs at fixed frequencies into the control systems successively on each control axis.

The following procedure was adopted for the tests :

- Wind-off measurements without blade : all the frequencies requested were successively realised. From this, the inertia influence of the actuator motions was obtained.

- Wind-on tests with the blades : when the requested steady conditions were reached, a point without excitation was acquired as dynamic zero and then, all the frequencies requested were realised.

The test configurations performed correspond to the following conditions :

$\mu = \frac{V}{\Omega R} = 0$	$\frac{CT}{\sigma} = 0.075$	
$\mu = 0.15$	$\frac{CT}{\sigma} = 0.075$	$\frac{(CxS)_f}{Ss} = 0.1$
$\mu = 0.3$	"	"
$\mu = 0.375$	"	"
$\mu = 0.425$	"	"
$\mu = 0.375$	$\frac{CT}{\sigma} = 0.10$	"

The tests started on December 18th and all the configurations were performed between the 18th and the 23rd of December 1997. The data base has been distributed to the partners at mid April 1998.

The data analysis performed confirms the high quality of the data base. For example, Figure 6.2 shows the evolution with the advance ratio of the effect of inputs in longitudinal cyclic on the rolling moment transfer function. These results demonstrate an increase of the off axis coupling when the advance ratio decreases and they show that the problem of off axis coupling on an helicopter at low speed appears already on an isolated rotor.

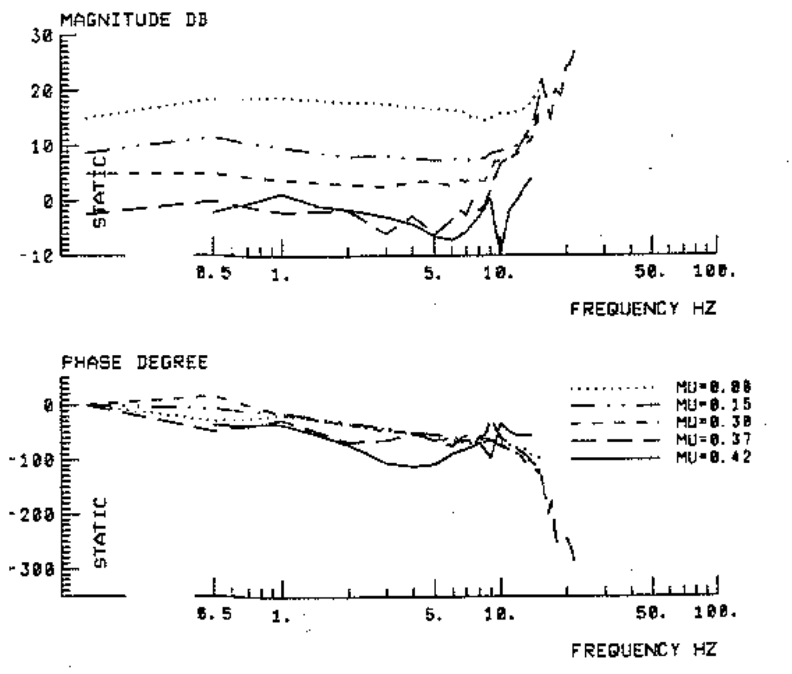

Figure 6.2 : Transfer function M_X/q_{1S}

Code validations were performed by the partners (DLR, EC, ECD, ONERA) participating in this task by comparisons between wind tunnel test results and computed results. For the on-axis transfer functions the computed results compare relatively well with the test data with however an over-prediction of the static gain for β_0/θ_0 and **Fz/**θ_0 which is largely improved when non-linearity, blade flexibility and vortex wake are taken into account (Figure 6.3).

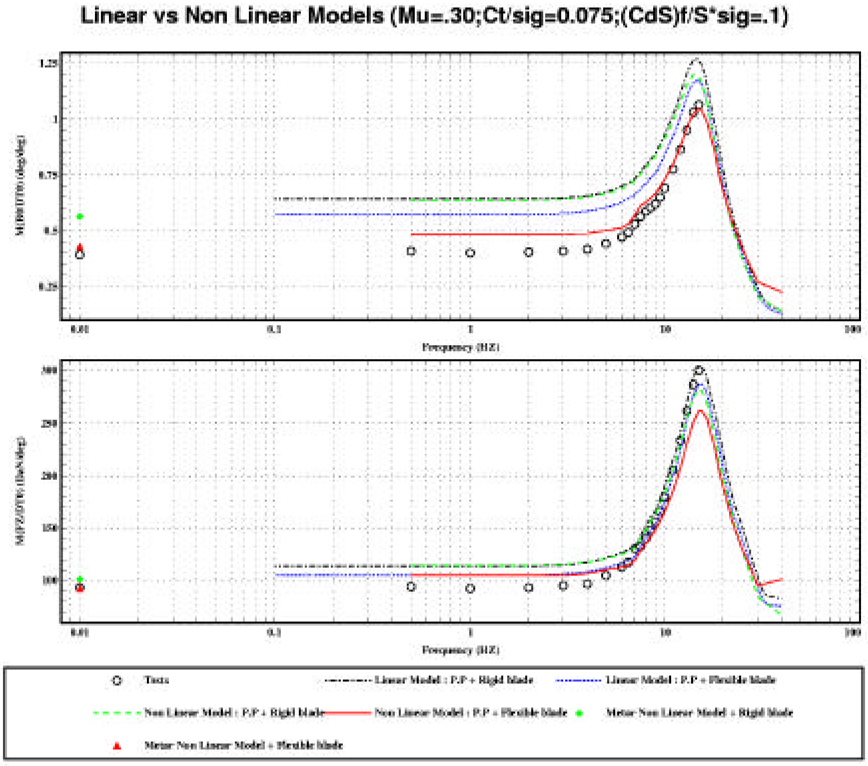

Figure 6.3 : Influence of blade flexibility (μ = 0.3)

For the off-axis transfer functions at low speed, in particular for the flapping and the moment responses to cyclic inputs, the results obtained are not good (Figure 6.4). Neither the effects of non-linearity, blade flexibility or vortex wake nor the ones of wake distortion improve these coupled responses.

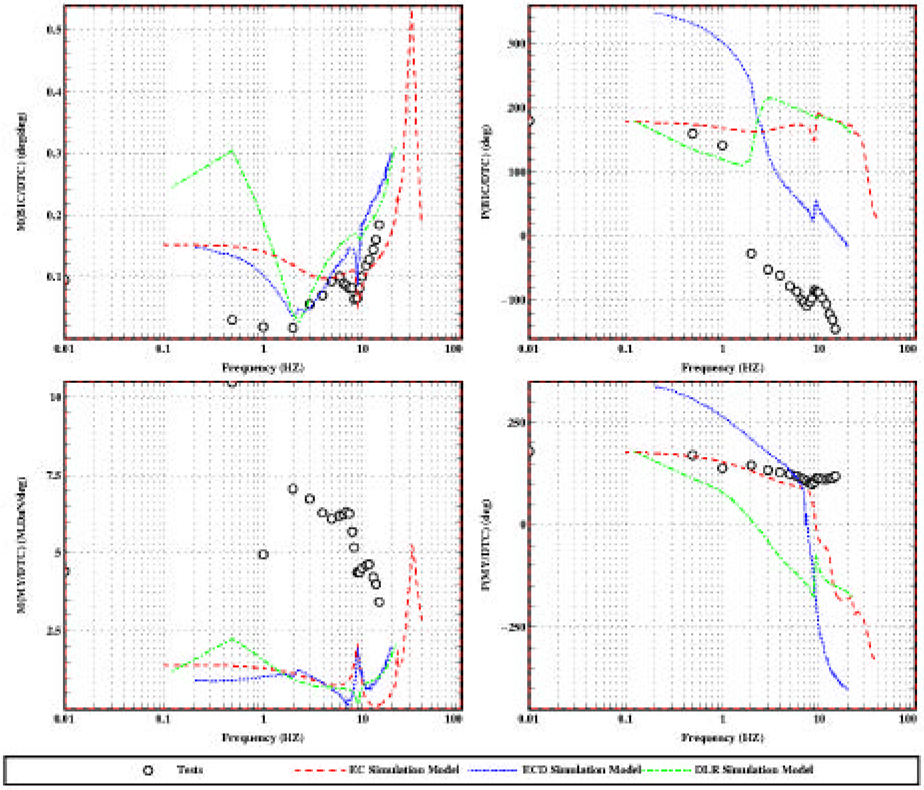

Figure 6.4 : Off-axis responses in hover

An identification method has also been applied in the frequency domain to obtain a parametric model built from stability and control derivatives. The off-axis results obtained with this identified model are largely improved in comparison with the ones given by a linear model (Figure 6.5).

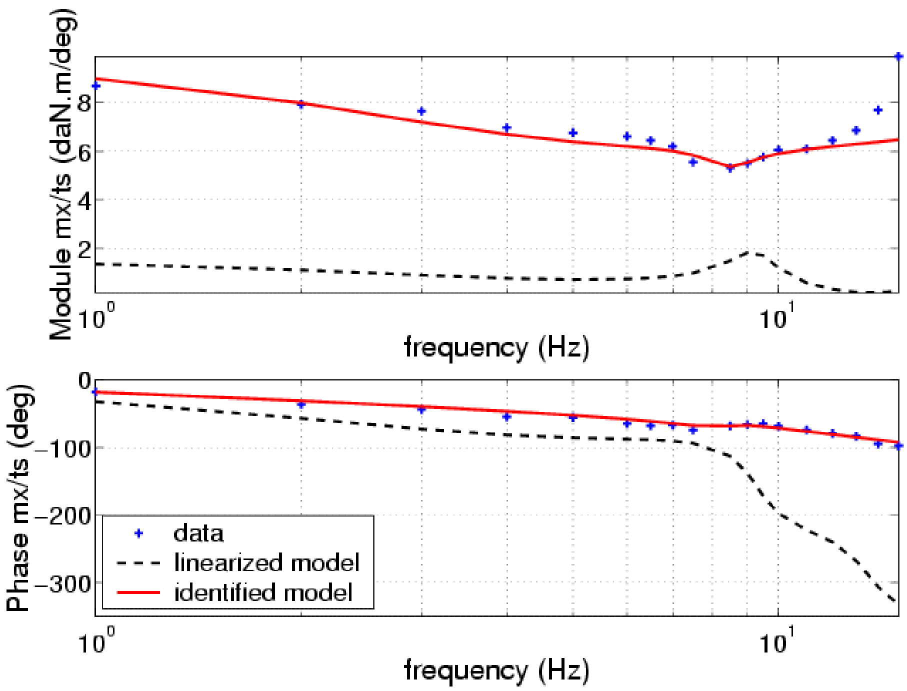

Figure 6.5 : Identification results in hover

The data bank obtained during this HELIFLOW programme will still be used for further code improvements after the end of the programme.

Conclusions

In spite of severe problems encountered during the wind tunnel experiments, very valuable results coud be attained:

- So, it could be demonstrated that the wind tunnel is the appropriate tool to simulate:
- complex interference phenomena as well as
- the complete helicopter trimmed as if it flew freely.

  This has to be seen in the light of testing in the past which focussed only on isolated components.
- The tests gave insight in the physics of the different phenomena addressed, which is impossible during conventional full scale flight testing
- In addition, extremely valuable data bases have been installed for the validation of flightmechanical (comprehensive) prediction codes. So, for the phenomena considered, the appropriateness of the partners' prediction codes could be demonstrated and deficiencies could be identified.
- Finally, the appopriateness of measurement techniques such as pressure sensors and Particle Image velocimetry has been demonstrated.

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