Industrial measurement campaign on fully equipped helicopter model

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Abstract The flow field around a helicopter is characterised by its inherent complexity including effects of fluid-structure interference, shock-boundary layer interaction, and dynamic stall. Since the advancement of computational fluid dynamics and computing capabilities has led to an increasing demand for experimental validation data, a comprehensive wind tunnel test of a fully equipped and motorized generic medium transport helicopter was conducted in the framework of the GOAHEAD project. In this paper the results of the three-component velocity field measurements are summarized. The effect of the interaction between the main rotor wake and the fuselage for cruise/tail shake conditions was investigated, detecting and analysing the flow characteristics downstream the rotor hub and the rear hatch. The results indicated a sensible increment of the intensity of the vortices shedding form the lower part of the fuselage and a strong influence of the main rotor in the upper region. Furthermore, the pitch up phenomenon was considered, detecting the blade tip vortices impacting on the horizontal tail plane. For high speed forward flight the shock wave forming on the advancing blade was investigated, measuring the location on the blade chord and the intensity. Furthermore, dynamic stall on the retreating main rotor blade in hight-speed forward flight was observed at 40% and 50% blade radius. The analysis of the substructures forming the dynamic stall vortex revealed an unexpected spatial concentration suggesting a rotational stabilization of large scale structures on the blade.

1. Introduction

Nowadays, the characterization of the flow field around a fully equipped helicopter still remains a challenging task. The complexity of the helicopter aerodynamics are characterized by unsteady flow-structure interaction, in particular, between the main rotor wake and the rear of the fuselage as well as the tail rotor. Phenomena know as pitch up and tail shake seriously interfere with handling quality and structural safety of the machine. Furthermore, phenomena such as shock wave-boundary layer interaction on the advancing blade or dynamic stall on the retreating blade, under-high speed conditions, greatly affect the rotor performance and the structural integrity of the main rotor. During the past decade considerable progress has been made in developing and advancing aerodynamic prediction capabilities for isolated helicopter components. The isolated main rotor downwash structure, for example, has been investigated mainly by means of optical methods under hover conditions (e.g. Heineck et al. 2000, Martin et al. 2002, Lee et al 2008) or low-speed forward flight (e.g. Raffel et al. 1998, Le Pape et al 2006 and van der Wall et al 2006).

Today, cutting edge computational fluid mechanics (CFD) tools are capable of predicting the viscous flow around rotor-fuselage configurations advancing towards complete helicopters. However, a detailed experimental validation data base to qualify methods and industrial design tools is still lacking. In order to fill this gap, a comprehensive experimental database for validation purposes was created within the European Union funded project GOAHEAD (see Pahlke 2007) with special emphasis on unsteady viscous flow phenomena like flow separation, transition and including rotor dynamics.

Therefore, an extensive wind tunnel test campaign was performed in spring 2008 in the Large Low-speed Facility of the German-Dutch wind tunnels (DNW-LLF) using a Mach-scaled model of a modern medium transport helicopter comprising all control surfaces on the fuselage as well as main and tail rotor (figure 1). Special care was taken to ensure well defined boundary conditions closely comparable to free flight conditions. Therefore, a closed test section was used. Additionally,
the in-flow velocity profiles were quantified by means of hot-wire anemometry and wall pressure measurements in order to provide accurate boundary conditions for CFD simulations. The experiments comprised measurements of the global forces of the main rotor and fuselage, normal force and bending moments acting on the horizontal stabilizer, axial force and torque of the tail rotor, steady and unsteady pressures, transition positions, stream lines. Furthermore, the position of flow separation, the velocity fields in the model wake, around the model fuselage and on the upper surface of the advancing and retreating rotor blades, the vortex trajectories and the elastic deformations of the main and tail rotor blades were investigated (see Raffel et al 2009). This paper summarizes the flow field measurements of the GOAHEAD test campaign jointly performed by CIRA, DLR and DNW.

Two model configurations were investigated by means of planar three-component particle image velocimetry (PIV). Both the isolated fuselage with rotating hub mounting stubs without blades and the fully equipped model configuration were considered. A variety of flight conditions and flow field regions were investigated within and beyond the generic helicopter mission envelope. Besides cruise flight conditions, pitch up, tail shake and dynamic stall phenomena were addressed. Pitch up is a low speed aerodynamic interference phenomenon which occurs during transition from hover flight to medium cruise speed or vice versa (Barbagallo et al 2000). The main rotor flow impinges on the horizontal tail unit, resulting in pitching moment fluctuations constraining the handling quality.

Tail shake is an aerodynamic phenomenon resulting from the interaction of rotor hub wake with the tail boom and the vertical tail invoking low frequency vibration, impacting on fatigue cycles of the vertical stabilizer and the tail rotor (de Waard et al 1999).

Dynamic Stall is an unsteady highly non-linear aerodynamic phenomenon caused by rapidly changing angle of attack. At high horizontal speeds, high local angles of attack on the retreating blade may be sufficient to form the well-known large scale dynamic stall vortex on the blade which eventually detaches. Thereby, the blade’s lift and pitching moment in this region are greatly altered by the life cycle of the vortex. A sudden drop of lift and pitching moment, can result in a twisting of the entire rotor blade exciting blade torsion modes associated with high dynamic pitch link loads. In summary, dynamic stall induces a periodically stalled region over the main rotor generating high frequency force oscillations which might potentially be safety-relevant (Leishman 2000). Finally, the shock wave occurring on the upper surface of the advancing blade under high-speed conditions was quantified.
2. Experimental Configuration

2.1 LLF-DNW wind tunnel
The tests were carried out in the Large Low-speed Facility (LLF) of the German-Dutch wind tunnels (DNW). The LLF is a closed loop industrial wind tunnel for the low-speed domain with a 8 x 6 m² closed test section operated at a maximum Mach number of Ma=0.34. The tests were performed at Ma=0.059 for the pitch up condition, at Ma=0.209 for cruise and tail shake conditions and at Ma=0.259 for the dynamic stall condition.

2.2 Model description
The model was composed of a NH90 fuselage scaled by a factor of 1:3.88 (figure 1). The fuselage was equipped with 300 steady pressure taps, 38 hot films sensors and 130 unsteady pressure transducers. The fuselage was integrated with the ONERA 7AD main rotor including the rotor hub. The four-bladed rotor had a radius of 2.1m rotating clockwisely (as seen from above). The entire test campaign was performed at constant rotational main rotor speed of \( \Omega_{MR}=956 \text{ rpm} \), corresponding to a blade tip Mach number of \( \text{Ma}_{MR}=0.617 \). The blades were instrumented with 118 dynamic pressure transducers for pressure distribution measurements at multiple span wise position of the blades and with 40 hot film sensors for transition localization. The rotor shaft was inclined by \( \alpha_s=-5^\circ \) with respect the fuselage vertical axis (figure 1).
A scaled two-bladed MBB Bo105 tail rotor with a radius of 0.383m and a S102 airfoil was used. The tail rotor was equipped with 36 dynamic pressure sensors. The rotational speed of the tail rotor was kept constant throughout the tests at five times the main rotor speed \( \Omega_{TR}=4780 \text{ rpm} \). Furthermore, static and dynamic loads of the main and tail rotor, on the fuselage, and the horizontal stabilizer were recorded by means of piezoelectric internal balances.

2.3 PIV instrumentations
For the measurements five double cavities Nd-Yag lasers with pulse energies of 280 mJ per pulse were used in connection with four 4Mpx cameras installed by motorized Scheimpflug adapters. High quality lenses with 180mm and 200mm focal length and f-numbers of 2.8 and 2 respectively were used. The data acquisition was synchronized with the main rotor by means of a one-per-revolution signal and a phase shifter enabling the adjustment of the phase-angle of the main rotor \( \Psi_{MR} \) (figure 2). The delay time was computed from the actual rotor period and a given phase shift. Due to the constant rotor frequency, phase jitter was found to be almost negligible.

![Fig. 2 Diagram of trigger system for rotor-laser synchronization](image)

Di-Ethyl-Hexyl-Sebacat (DEHS) particles atomized by 64 Laskin nozzle particle generators were used as flow tracers. The particles were distributed through a rake mounted in the settling chamber of the wind tunnel. The rake was remotely traversed to guide the homogeneous stream of tracers to the region of interest. The mean particle diameter was below 1 \( \mu \text{m} \) as confirmed by previous tests.

2.4 Measurement regions and experimental set-up
The flow field measurements involved various different locations in the vicinity of the main rotor (figure 3), in particular, on the advancing blade (\( \Psi_{MR}=90^\circ \), black regions) and on the retreating
blade ($\Psi_{MR}=270^\circ$, bleu regions) for high forward flight speed ($M_{WT}=0.259$). Furthermore, the cross flow region downstream from the rotor hub (red regions) and downstream from the fuselage hatch (grey regions) under cruise and tail shake conditions ($M_{WT}=0.209$) were acquired. The vertical region above the horizontal stabilizer (green region) was considered with regard to the pitch up phenomenon ($M_{WT}=0.059$). In order to evaluate the interference between the main rotor and the fuselage, the cruise flight conditions were analysed for the isolated fuselage and for the fully equipped model.

Three component velocity measurements were acquired on several parallel cross planes downstream from the rotor hub (field size $1 \times 0.35m^2$) and downstream from the fuselage rear hatch (field size $0.5 \times 0.35m^2$) under cruise flight and tail shake flight conditions. The upper wake region was obtained measuring separately on the right and left of the fuselage. The measurements provided a flow field space resolution varying from 2.6mm to 3.6mm. The region of interest was illuminated from both sides of the wind tunnel walls by two aligned laser light sheets providing partial forward scattering to both cameras (figure 4). The laser light sheet thickness was approximately 8mm. The region of interest was recorded by two cameras located inside the circuit close to the side walls at about 3.4m downstream of the measurement plane. The angle between the camera sight views and the measurement plane normal direction was $50^\circ$. For the purpose of calibration the camera, calibration target, and model positions were acquired by a theodolite system.

Optimizing the measurement system handling and minimizing set-up times, lasers and cameras were installed on traversing systems and rigidly translated avoiding additional calibration procedures. Aiming at the flow field beneath the fuselage the same configuration was used.
Another stereo PIV configuration with the region of interest above the horizontal tail plane was used for the pitch up case. A laser was mounted on the wind tunnel ceiling and the light sheet was directed toward the horizontal tail plane through an optical window. The upper region above the horizontal tail plane was illuminated by a vertical light sheet parallel to the mean flow velocity. The two cameras were installed on the left hand side of the wind tunnel. One camera was mounted inside the circuit on a two-dimensional traversing system, the other one was located outside the wind tunnel wall (figure 5). The angle between the camera sight view and the normal direction to the plane was 48°. The adopted configuration provided a field size of 0.46x0.33 m² at a spatial resolution of 3.5mm.

![Fig. 6 Dynamic stall Exp. Set up](image)

The dynamic stall measurements targeted the flow field on the suction side and in the near-wake of the retreating main rotor blade (Ψ_{MR}=270°) at various radial positions (figure 6). The laser light sheet was provided from the ceiling of the wind tunnel. In order to reduce the light reflection on the upper surface of the blade, the light sheet access was located far downstream the model in order to achieve an approximately tangential impingement on the surface of the blade. Due to the limited optical access from above the scan planes were tilted with respect to the incident flow resulting in an inclination angle β with respect the wind axis (figure 6). The measurements were performed at three radial positions (at 50%, 60% and 90% of the blade radius R_{MR}) with field sizes varying from 0.23x0.22m² to 0.16x0.16m² and corresponding spatial resolutions varying from 1.4mm to 1mm.

The shock wave investigation on the advancing blade consisted of two-component PIV measurements. The data acquisition was performed simultaneously with the dynamic stall measurements on the retreating blade. The region of interest covers two locations at 85% and 95% of the main rotor blade radius R_{MR} at Ψ_{MR} = 90°. A measurement region of 0.2x0.18m² with a spatial resolution of 3 mm was obtained.

Again the upper surface of the rotor blade was illuminated by a laser located on the ceiling. The laser light sheet was inclined by 11° with respect to the vertical axis in order to illuminate a field orthogonal to the blade axis while compensating the blade deformation (figure 7).

In summary 37 different measurement regions were recorded for three different flight conditions and two model configurations.

### 2.5 Accuracy Estimation

The PIV data evaluation follows the standard procedures given by Raffel et al (2004). However, given the large geometry and the complexity of the set-up, a measurement accuracy estimation should be given here.

The random noise of the displacement is smaller than 0.05px for all observation areas, except for vortex core region. Within the strong velocity gradients of the cores, errors reach up to 0.1px. Minor bias errors were expected, including minor effects of peak-locking. However, based on the
displacement histograms, the bias error was estimated to fall below random noise, i.e. less than 0.05px. The resulting velocity error $\varepsilon_u$ was estimated as:

$$\varepsilon_u = \varepsilon_x / (\Delta t M)$$  \hspace{1cm} (1)

where $\Delta t$ is the pulse-separation time and $M$ is the optical magnification. Scaling with the maximum in-plane velocity component $U_{\text{max}}$ the relative error $\varepsilon_{u, \text{rel}} = \varepsilon_u / U_{\text{max}}$ was determined as

$$\varepsilon_{u, \text{rel}} < W_{\text{max}} / U_{\text{max}} \cdot 3 / (\Delta Z \cdot M) \cdot \varepsilon_x$$  \hspace{1cm} (2)

where, $W_{\text{max}}$ is the maximum out-of-plane component of the flow velocity, $\Delta Z$ is the light-sheet thickness, and $\varepsilon_x$ is the displacement uncertainty in pixel dimensions.

Due to the recording geometry and the measurement uncertainty, the out-of-plane measurement error $\varepsilon_w$ was approximately a factor of $\sqrt{2}$ higher than that of the in-plane measurement.

The largest measurement error in the velocity domain and the relative measurement errors for all the test cases, for the displacement error of $\varepsilon_x = 0.1\, \text{px}$ are reported in table 1.

<table>
<thead>
<tr>
<th>Cruise Upper fuselage</th>
<th>Cruise Lower fuselage</th>
<th>Pitch up</th>
<th>Shock wave</th>
<th>Dynamic stall at 60% of R</th>
<th>Dynamic stall at 50% of R</th>
</tr>
</thead>
<tbody>
<tr>
<td>$M$ [px/m]</td>
<td>6.30E+03</td>
<td>5.95E+03</td>
<td>5.80E+03</td>
<td>1.07E+04</td>
<td>1.00E+04</td>
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<tr>
<td>$Dz$ [m]</td>
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<td>9.00E-03</td>
<td>9.00E-03</td>
<td>5.00E-03</td>
<td>5.00E-03</td>
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<tr>
<td>$U_{\text{max}}$ [m/s]</td>
<td>1.00E+01</td>
<td>3.50E+01</td>
<td>3.00E+01</td>
<td>2.00E+02</td>
<td>1.30E+02</td>
</tr>
<tr>
<td>$W_{\text{max}}$ [m/s]</td>
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<td>7.00E+01</td>
<td>1.10E+01</td>
<td>1.00E+01</td>
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</tr>
<tr>
<td>$Dt$ [s]</td>
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<td>2.00E-05</td>
<td>1.50E-05</td>
<td>4.00E-06</td>
<td>7.00E-06</td>
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<tr>
<td>$\varepsilon_x$ [px]</td>
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<td>0.10</td>
<td>0.10</td>
<td>0.10</td>
<td>0.10</td>
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<tr>
<td>$\varepsilon_w$ [m/s]</td>
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<td>$\varepsilon_{u, \text{rel}}$ [%]</td>
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<td>$\varepsilon_{u, \text{rel}}$ [%]</td>
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<td>3.79E-01</td>
<td>2.81E-02</td>
<td>9.23E-02</td>
</tr>
</tbody>
</table>

Table 1 Maximum measurement deviation estimation

2.6 Data Post Processing

In the framework of data evaluation, several different vortex identification schemes were used. The vortex centre locations were determined by either calculating the out of plane vorticity component ($\omega_z$); and evaluating the Eigenvalues ($\lambda_2$) of the velocity gradient tensor (Vollmers 2001). Once the vortex centres were detected, maximum vorticity, vortex radius, maximum tangential velocity were derived along with an evaluation of the out of plane vorticity ($\omega_z$), circulation $\Gamma$, radial ($V_r$) and tangential ($V_\theta$) velocity components versus the vortex radius ($r$).

![Fig. 8 WT reference system](image1)

![Fig. 9 Flow field behaviour for $\Psi_{MR}=0^\circ$](image2)
All the data relevant to this paper refer to the wind tunnel reference system (figure 8). The origin is located in the centre of the wind tunnel test section symmetry line, the x axes is directed parallel to the mean flow velocity, the y axes is horizontal and positive versus starboard direction and the z axis is orthogonal to xy plane and positive upward.

3. Results and discussions

3.1 Cruise and Tail shake
The analysis of the dynamic pressure and load data on the tail plane indicated that the selected flight cruise conditions coincided with tail shake. The test were performed at an incident Mach number of \( Ma = 0.209 \) and a fuselage incidence of \( \alpha = -2.5^\circ \). Nine parallel planes were measured starting from the zone immediately downstream the engine exhausts down to the vertical tail plane. The flow field was acquired for five different azimuth angles (\( \Psi_{MR} = 0^\circ, 22.5^\circ, 45^\circ, 67.5^\circ \) and \( 90^\circ \)). This region is characterised by highly unsteady flow due to separation starting form engine exhausts, the rotor hub wake and the main rotor tip vortex. Unfortunately, during these measurements a malfunctioning of the acquisition system (especially for the isolated fuselage model configuration) occurred which caused the loss of most of the instantaneous velocity fields preventing a comparison between the two model configurations.

![Flow field Vorticity map on the third scan plane at \( \Psi_{MR} = 45^\circ \) (left) and \( \Psi_{MR} = 67.5^\circ \) (right)](image)

The velocity magnitude and the stream lines for \( \Psi_{MR} = 0^\circ \) clearly indicated the presence of two major counter rotating vortices emerging from the fuselage exhausts (figure 9). The flow field characteristics depended on main rotor azimuth angles as shown in the out-of-plane vorticity and the calculated stream lines of the scan planes at \( \Psi_{MR} = 45^\circ \) and \( \Psi_{MR} = 67.5^\circ \) (figure 10). The two major counter rotating vortices (indicated by A and B in figure 10) were clearly visible, along with a secondary pair of counter rotating vortices (bottom of figure 10 C and D) detaching from the tail boom. At \( \Psi_{MR} = 45^\circ \) an outstretched zone of high vorticity (indicated by E in figure 10) indicated the presence of shear layer or the cut of a vortex parallel to the measurement plane. The vorticity strip was encountered on all the parallel measurement planes for some rotor azimuth angles (\( \Psi_{MR} = 22.5^\circ, \Psi_{MR} = 45^\circ \) and \( \Psi_{MR} = 67.5^\circ \)), following a slightly descendent path from the top right to the left of the flow region as the azimuth angle increased. The vorticity strip might be attributed to the wake of the fourth blade.

The lower fuselage zone was investigated in ten parallel planes at \( \Psi_{MR} = 0^\circ \). The region downstream from the fuselage hatch was characterized by two counter rotating vortices shedding from the fuselage (figure 11).
The mean velocity components were measured for the two model configurations. The full equipped model presented for the cross flow (v) velocity component higher values whereas the vertical velocity component (w) indicated lower values respect to the isolated fuselage as consequence of the main rotor wake (figure 13). Furthermore the axial velocity component (u) presented a reduction indicating an increment of the fuselage wake with consequent increment of the fuselage drag.

The counter rotating vortices for the full equipped model, respect the isolated fuselage model configuration, were drag toward the fuselage centre line and down respect to the fuselage (figure 12).

The effect of the main rotor wake behaved on the vortex characteristics with a substantial increment of the strength of the vortex. The vortices maximum tangential velocity increased of about 75% and the peak of vorticity presented an increment varying from 75% to 350% (figure 14). The vortex centre location was detected on the instantaneous velocity fields showing a concentrate distribution. The centres distribution was characterised by a standard deviation varying from a minimum of 3mm to a maximum of 40 mm.

Fig. 11 Velocity magnitude below the tail boom

Fig. 12 Vortex path isolated fuselage (red) full model (bleu)

Fig. 13 Mean velocity components comparison for isolated fuselage and full model
3.2 Pitch-up

The Pitch up phenomenon was initially investigated at \(Ma=0.059\) and fuselage incidence of \(\alpha=1.9^\circ\) on the base of previous pressure and loads measurement on the horizontal stabiliser.

The main rotor azimuth angle was selected on the base of the presence in the PIV field of view of the main rotor blade tip vortex. The tip vortex was detected at \(\Psi_{MR}\) in the rage between 56° and 85°.

The PIV measurements (figure 15) clearly showed the tail plane fully immersed in the main rotor wake and the tip vortex passing far away above from it. The vortex trajectory, with the horizontal tail plane position, is shown in figure 16. The results indicated that the pitch up condition was not fully reached for the selected test condition. In order to get the tip vortex impacting on the tail plane the wind tunnel Mach number was decreased respectively down to \(Ma=0.043\) and \(Ma=0.031\) and the fuselage incidence varied to \(\alpha=-1.1^\circ\).

The results indicated that the tip vortices moved closer to the tail plane for \(Ma=0.043\) (figure 17) and impacted on the leading edge for \(Ma=0.031\) (figure 18). Furthermore counter rotating vortices
shedding by the lower surface were detected, indicating that the stabilizer was stalled as consequence of the main rotor vertical velocity.

The tip blade vortex path behaviors (Figure 19) showed a larger dispersion in the vortex centre distribution for the case at Ma=0.031 due to the interference with the stabilizer leading edge. The vorticity distribution of the blade tip vortex for the different wind speed showed a substantial similar behavior (figure 20). Only for Ma=0.031 the vortex presented a discordant behavior caused by the location of the vortex too close to the edge of the PIV field of view for extracting enough useful information.

3.3 High-speed

Within the high-speed case, at a free-stream and rotor Mach number of Ma=0.259 and Ma\textsubscript{MR}=0.617, respectively dynamic stall was clearly identified at r/R=0.5 and 0.6. At the most outward radial position r/R=0.9, however, fully separated flow along with strong three-dimensionality indicated the formation of the blade tip vortex. The dynamic stall vortex is discernable in the phase averaged velocity field at $\Psi_{\text{MR}} = 272.3^\circ$ (figure 21) where the in-plane velocity components $u$ and $w$ are scaled with the free-stream velocity $U_\infty$ while the out-of-plane component is normalized as $(v-\nu)/U_\infty$ with $\nu$ taken far from the blade surface.

The vortex centres axes were identified using the Galilean invariant form of the scalar function introduced by (ref Graftieaux2001). Unlike alternative criteria for vortex centre identification such as vorticity peak detection, $\lambda_2$, etc., the Galilean invariant does not require velocity field derivatives or the out-of-plane component reconstructed during evaluation and, therefore, it is less susceptible to experimental noise.

Notably, the dynamic stall vortex, formed by the roll-up of the shear layer, is observed to be
unexpectedly compact and spatially stable. Considering the distribution of center positions of individual (small-scale) shear layer vortices extracted from the instantaneous velocity fields, the macro-structure is found to be strongly localized (figure 21). The concentration of the vortex centre positions is much more pronounced than usually observed in two-dimensional measurements, suggesting that the rotation stabilizes the large-scale structures.

In parallel the flow field behavior at 85% and 95% of the advancing rotor blade radius was investigated. The measurements suffered by lack of seeding concentration, because the seeding rake was aligned with the retreating blade zone and by strong surface reflection losing the region immediately close the surface. The mean velocity fields showed a strong shock wave located at about the 37% of the blade chord.

For each single velocity field the effective azimuth angle position was measured. The analysis of effective azimuth angles provided a mean value equal at $\Psi_{MR}=92.73^\circ$ and a value of the standard deviation equal to 0.166° confirming a real stable rotational speed of the rotor. However the fluctuation of 0.166° induced a shift of blade in the measurement region of about 3.7% of the chord length. The main rotor rotational speed jitter affected the ensemble average results attenuating the shock velocity drop.

The ensemble average data presented in concomitance of the shock a mild slope in comparison with the instantaneous velocity behaviour (figure 22). The shock locations of the instantaneous velocity fields were apart of about a distance correspondent to 13% of the blade chord whereas the azimuth shift between the two data set corresponded to a chord shift of 10%, the remaining 3% was attributed to slightly different blade pitch angles and to different blade deformations.

4. Conclusions
An extensive and successful measurement campaign was carried out on a full equipped and motorised helicopter model. Three component PIV measurements were performed on 37 different regions for two models configurations (isolated fuselage and Full model) and four different flight conditions. The optimised experimental set-up allowed for efficient calibration and measurement procedures yielding a large number of different configurations, allowing an increment of the investigated regions.

The influence of the main rotor wake on the fuselage was evaluated. The vortical structures in the wake region downstream the rotor hub were localised and their sources detected. The region below the fuselage for the full equipped model presented an increment of the strength of the detached vortices with a variation of their trajectories and an increment of the fuselage drag.

The pitch up and dynamic stall condition were reached and the flow phenomena investigated. The PIV measurement allowed to select the appropriate flight condition for reaching the pitch up
condition. The location and characteristics of the tip vortices impacting on the horizontal stabiliser were measured. The dynamic stall behaviour was analysed, noticing a pronounced vortex concentration with respect to previous two-dimensional experiments suggesting a rotational stabilisation of the large-scale structures. The compressibility effects occurring on the advancing blade were detected, and the location and strength of the shock wave was measured. The results are now available to the GOAHEAD data base, providing a unique tool to the European helicopter industries and to the scientific community. The GOAHEAD project has been a particularly successful example of cooperation between numerous European partners involved in instrumentation integration, measurements procedures, data exchange and data post processing procedures.

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References