A Multi-Diagnostic Approach to Testing V/STOL Craft

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ABSTRACT†§

The interactive aerodynamics of new V/STOL designs require a new approach to ground testing, where multiple properties are measured both on and away from surfaces during continuous changes in test parameters. Using a set of experiments of increasing complexity, a new capability is demonstrated for capturing surface pressure, velocity fields and vortex flow features over a range of test parameters. Two-dimensional velocity fields are captured over a full-scale UH-1 helicopter stabilator, a wing/canard configuration, and under the hub of a rotor in forward flight. Surface pressures and vortex flow features are captured over a wing/rotor configuration. Short tunnel run-times are achieved at the expense of large computational post-processing resources. It is shown that such experiments can be performed while the configuration geometry and flow conditions are continuously varied through a wide range of parameters. This experiment led to the discovery and capture of massive flow separation caused by vortex interaction on wings at moderate angles of attack, large-scale deflection of vortex wake trajectories by wake/wing interaction, unsteady flows over wings generated by canard interaction, and quasi-steady separation phenomena caused by the difference in time scales between separation and reattachment.

INTRODUCTION

There are several concepts being studied for high-speed rotorcraft†, commuter craft, and V/STOL fixed wing craft. In each of these concepts, the test engineers face a huge problem in the certification process: how to test for all of the variety of interactional phenomena that occur during transitions between hover and low-speed flight. For instance, the flows around the canards, wings, and rotors of proposed tiltrotor concepts can interact with large-amplitude, high-frequency unsteadiness. Integrated measurements of lift, drag, and quasi-steady stability derivatives are inadequate.

Why are multiple diagnostics needed?

There are two major reasons for such experiments. First is the multitude of interactional phenomena that cannot be reasonably predicted in such flows. The best way of finding these is by looking for them using efficient, rapid...
tests. A second purpose is to get reliable measurements to validate analytical methods, which can then be used with increased confidence to predict the behavior of full-scale craft. When one tries to find out why experiment and analysis do not match, one often runs into a dead end if the available information is insufficient. Global explanations based on steady-flow aerodynamics usually fail to anticipate the dominant features of unsteady interactional aerodynamics: there are too many spatial and temporal dimensions to permit "intuitive" solutions. Thus, to get an adequate set of data, one must measure several variables, with some redundancy if possible.

**Large test section size**

To perform useful tests on V/STOL configurations, the test section must be fairly large. Facility wall effects are always a problem, and hence, for a given model size, a larger test section is needed than for tests at cruise conditions. Thus, the capabilities demonstrated must be applicable to fairly large areas, with large working distances.

**PREVIOUS WORK**

Over the past ten years, the John J. Harper Wind Tunnel at Georgia Tech has been used for a systematic study of rotorcraft interactional aerodynamics. Early tests with a 2-bladed rotor and a hemisphere/cylinder progressed from measurements of time-averaged surface pressure distributions and wake boundaries to unsteady pressures, and then to vortex trajectories and velocity fields. In the late 1980s, we succeeded in performing extensive, detailed measurements of velocity fields of isolated rotors and rotor/body configurations one point at a time, using the periodicity of the rotor wake to enable accurate phase-averaged measurements. Typically, it took fourteen minutes of tunnel run-time to accurately measure two components of the periodic velocity variation at each location. We realized that it was relatively easy to visualize the rotor vortex trajectories by strobing a laser sheet using a pulse from a rotor shaft encoder. This soon led to a video-based technique where the dynamics of the vortex systems could be recorded by changing the pulse frequency slightly from the rotor frequency. These measurements led to the identification of the missing phenomena which would enable accurate prediction of the dominant features of the interactive flowfield and its effects. In the process, we learned four lessons:

1) Point-wise laser velocimetry is extremely tedious, and is expensive in terms of run-time. Away from flow boundaries, the effort involved in getting precise LDV data is wasted: computational techniques are a long way from being able to use such precision for validation.

2) The dynamic laser sheet visualization provided quantitative information which proved to be most useful in seeing what was really happening in the flow, and in identifying areas of improvement for prediction methods.

3) Despite years of "free" tunnel time, we were able to study only a very few configurations in detail. There was a myriad of other interactional effects of interest.

4) The real bottleneck was the lack of a quick way to quantify velocity fields. These lessons led us to search for techniques to cut down the run time needed to quantify flowfields.

**OBJECTIVES**

In this paper, we deal with three aspects. First, we experiment with the run-time actually needed to document the pressure and flowfield features in a rotor/wing interaction case over a range of parameters. Second, we discuss progress on large-area, time-resolved velocity measurement. Third, we discuss the phenomena uncovered during the rotor/wing experiment.

**MODELS**

The three items used as test models are shown in Fig. 1a. A "canard" with untwisted NACA 0012 section, 0.229m chord and 1.067m span, was held on a 3-axis traverse, able to move vertically, longitudinally, and rotate about its pitch axis under computer control. A full-scale UH-1 helicopter stabilizer with a 0.813m chord and 1.944m span was used to simulate an aircraft wing. A two-bladed teetering rotor of 0.914m diameter and constant NACA0015 blade section was attached to the end of a motor shaft through the roof of the John J. Harper 2.1m x 2.5m Wind Tunnel. The wing was held fixed at selected positions. These items were used in isolation and combination. Rotor downflow was varied by changing rpm (600 to 2100). The advance ratio (freestream speed divided by rotor tip speed) was varied by changing wind tunnel speed. The relative positions of the canard, wing, and rotor and the canard incidence setting were changed under digital control.

**THE INTEGRATED EXPERIMENT**

**Flow Visualization**

Prior to each run, a grid board used to align cameras was recorded in order to provide a spatially-accurate frame of reference for flow images. A set of smoke wires upstream created thin flat sheets of intense flow seeding. For more uniform seeding, a pair of atomizer seeders were used downstream to fill the closed-return tunnel with a mist of mineral oil droplets (nominal diameter: 4 microns). The beam from a pulsed 15-watt copper vapor laser was sent through a chopper to create a strobe effect, and then expanded into a sheet to illuminate the flow section of interest. Combinations of the video imaging rate and the strobing rate (or rotor rpm) were used to observe periodic phenomena with various time scales. Where necessary, a phase-delayed pair of intensified video cameras fitted with television zoom lenses permitted the capture of events occurring during just two successive flashes of the 6300-pulses/second laser, with each flash freezing the flowfield for just 25 nanoseconds.
Video signals were recorded on two VCRs. A camcorder mounted above the rotor shaft captured the instantaneous rotor azimuth from a protractor on the rotor shaft, and recorded this into a small window on one of the VCRs, allowing events to be related to rotor phase. In the experiments where advance ratio was varied, the third camera was used instead to record the digital display for tunnel dynamic pressure into a small window in order to relate events to advance ratio.

**Surface Pressure Measurements**

A 16-bit, 16-channel A/D converter digitized the output from several condenser microphones flush-mounted on the wing surface. To protect the microphone diaphragms from stray seed particles, a thin wire mesh was placed over each microphone port. An HP1000 A700 computer was used to perform time series analysis of the data. Mean static pressure distributions were determined from pressure taps evenly spaced along the wing chordline and connected to a Scanivalve pressure switch. This technology is ancient. Modern versions use parallel-processing of large number of pressure sensor outputs, or, most recently, image processing of pressure-sensitive paint. Either of these techniques would enable quasi-steady measurement of surface pressure distributions while configurations are being changed, with the latter technique also eliminating the need for drilling holes and attaching sensors and cables. The ultraviolet lighting needed for the paint method can probably be synchronized in phase-delay with the lighting for the flow visualization and velocity measurements.

**Velocity Measurements**

Using Spatial Correlation Velocimetry, it has become possible to extract instantaneous, quantitative velocity field information using the same data that is used for flow visualization. Thus it is now possible to measure velocity field histories while the configuration is being continuously varied, or even during high-rate, large-amplitude transient maneuvers. Over the past two years, the technique has progressed from solid surfaces and water flows (50mm x 50mm, 150mm/sec, Ref. 11) to unsteady air flow over a plunging wing (300mm x 300mm, 3m/s, Ref. 13) to the wing-canard experiment detailed in Ref. 12 (600mm x 400mm, 10 m/s). Implementation of SCV begins with the grabbing and digitizing of two corresponding frames separated by a known time delay. A first-level velocity computation is performed on the 80486 computer, and subsequent iterative processing is done on the Thinking Machines Connection Machine at the Pittsburgh Supercomputer Center.

**Geometric Variation**

The ability to view a larger flowfield permitted us to video tape flow phenomena in the area of interest above the 0.813m chord wing as well as to record changes in the canard geometry. After a short wind tunnel run in which the canard’s vertical, longitudinal and pitch positions were varied, the two video tapes were played back through a pair of video players. Canard geometry was determined directly from the video monitor.

Frame numbers were coded onto the videotapes after the experiment, with a reference frame marked by the laser beam being turned on. Corresponding frames were determined and digitized onto a PC-compatible 80486-33MHz computer. Successive images separated by a time delay on the order of 50-90 micro-seconds were analyzed to obtain velocity fields as detailed in Ref. 12.

The integrated flow experiment thus seeks to shift the majority of the burden from the wind tunnel to the computer. Thus, it takes only a few minutes of tunnel run time to complete slow transition through all of the required combinations of geometric and flow parameters. Then comes the massive analysis effort, where we use the ever-increasing speed of computers. This phase can now be automated to the desired level.

**ISOLATED WING FLOWFIELD**

Fig. 2 shows the velocity field above the isolated wing (the UH-1 stabilator) at a moderate angle of attack. Iterative post-processing enables high chordwise resolution of the instantaneous velocity field. The flow is obviously attached and smooth over the wing at this condition.

**WING/CANARD FLOWFIELD**

Figures 3 and 4 (Ref. 12) show two instantaneous velocity fields over the wing/canard configuration, one with the canard at low angle of attack, and the other with the canard stalled. The flow in the first case is quite steady; that in the second is unsteady and appears to have a jet-like flow below the canard trailing edge. Figure 5 shows quantitative data on the velocity profile across this jet-like flow, obtained from the instantaneous velocity field above the wing. Figure 6 shows the instantaneous velocity along the first line of vectors parallel to and above the wing surface for three conditions. The isolated wing case shows the expected acceleration and deceleration of the flow. The case with attached canard flow shows a substantial, but smoothly-varying, interaction effect. The case with the stalled canard shows large-amplitude fluctuations in the velocity field. In Ref.12, we have shown a detailed history of the velocity field over the wing/canard configuration measured while the canard was continuously changing position.

**ISOLATED ROTOR**

The rotor flowfield is unsteady and contains very sharp gradients and strong 3-dimensional velocity fluctuations. It poses a very strong challenge even to single-point techniques such as LDV. To-date, our efforts with the Spatial Correlation Velocimetry technique have been limited to those regions of the rotor wake where the gradients are
relatively mild, and the range of velocity variations is small. Fig. 7 shows a sample velocity field below the hub region of the rotor. In the remainder of the rotor/wing/canard experiment, we are as yet unable to discuss velocity field results.

**ROTOR/WING/CANARD EXPERIMENT**

Canard geometry and rotor azimuth were determined as explained previously. Wing geometry remained fixed during tunnel runs and was recorded manually prior to wind tunnel testing. Leading edge distances for this experiment were measured from a fixed point in the configuration located at the center of the rotor shaft. Analysis of video recordings and pressure data was done extensively for four canard positions. Relative distances between the canard, wing and rotor are shown in Fig. 1b. The canard positions are listed below. Horizontal and vertical distances are distances from canard leading edge to a fixed point located on the rotor shaft located at the center of the rotor:

**Table 1: Canard Positions**

<table>
<thead>
<tr>
<th>Position #</th>
<th>Horizontal Distance (X/R)</th>
<th>Vertical Distance (Z/R)</th>
<th>Angle of Attack (deg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>-1.0764</td>
<td>-0.7361</td>
<td>0</td>
</tr>
<tr>
<td>2</td>
<td>-1.0721</td>
<td>-0.7034</td>
<td>15</td>
</tr>
<tr>
<td>3</td>
<td>-1.0721</td>
<td>-0.5367</td>
<td>15</td>
</tr>
<tr>
<td>4</td>
<td>-1.0764</td>
<td>-0.5694</td>
<td>0</td>
</tr>
</tbody>
</table>

In this experiment, flow field images, time-averaged surface pressure variations, and phase-averaged unsteady pressure variations were measured for each condition. We hope that the recorded flow images will also yield a substantial amount of velocity field information as our analysis progresses, but that remains to be seen.

**TESTING TIME**

The time taken to perform the experiments is as listed in Table 2. Tunnel time refers to the actual time the tunnel is on and running. Total test time is the time from tunnel start up to the tunnel shut down at the end of the experiment. Various adjustments must be made throughout the experiment. The smoke wires must be re-painted with wax at intervals, and listed below are assorted adjustments which must be performed to cameras, etc. The time required to set up the experiment is not included. This is at present a function of the level of experience, the level of sophistication of the camera alignment systems, the laser sheet alignment procedures, the class schedules and homework assignment deadlines of the experimenters, and other subjective parameters.

**Table 2: Testing Time**

<table>
<thead>
<tr>
<th>Test Performed</th>
<th>Tunnel Time</th>
<th>Total Test Time</th>
</tr>
</thead>
<tbody>
<tr>
<td>Preliminary Testing - includes checking camera settings, smoke wires, rotor RPM and advance ratio, configuration of canard/wing, laser alignment</td>
<td>55 min</td>
<td>2 hrs</td>
</tr>
<tr>
<td>Microphone/Pressure Tap/Flow Visualization - 4 canard positions at 3 advance ratios with 29 pressure taps and 3 microphones (6 for μ=0.075), Cp distribution with no rotor</td>
<td>7 hrs</td>
<td>8 hrs</td>
</tr>
<tr>
<td>SCV Testing - included changes in canard angle of attack, advance ratio and time delay, rotor only test</td>
<td>50 min</td>
<td>2 hrs</td>
</tr>
</tbody>
</table>

**RESULTS**

**Vortex Trajectories**

Quantitative data on vortex trajectories were measured by tracking each vortex core on the video monitor at individual rotor azimuths and comparing their positions with the recorded grid board. The entire range of rotor azimuths was covered, using the recorded azimuth reading on each frame for reference. X and Z coordinates for vortices at 45 to 60 degree increments provided an adequate amount of data to plot vortex trajectories here. Fig. 8a shows the subtle changes in vortex trajectory with varying canard position for an advance ratio of 0.06. It should be noted that the vortices are not seen to "hit" the surface of the wing. The vortices approach the wing surface, interact with flow over the wing, and appear to bounce off the boundary layer or the separated flow region. As advance ratio increased (Figs. 8b and 8c), the path which the vortex followed moved further off the wing until at μ=0.10, the vortex no longer interacted with the wing.

There appears to be little change in the trajectory of the vortices with a change in canard position. The only noticeable effect on the vortex trajectory is when the canard is level with the wing’s leading edge and at a zero angle of attack (Position 1), the freestream velocity convects the vortices further downstream towards the trailing edge. When the canard is moved to Positions 2-4, its vertical and/or rotational placement interferes with the freestream flow. Convection is less effective in these cases and the trajectory is moved closer to the leading edge.
Vortex-Surface Interactions

While flow visualization yields limited quantitative data, a great deal of qualitative features and phenomena can be observed. One such phenomenon was the separation of flow over the wing while the wing was fixed at a relatively small angle of attack. Most of the wind tunnel testing done for this report was performed with the wing fixed at a 10.4 degree angle of attack. We would expect that when the wind tunnel was run without the rotor with the wing placed at that small an angle of attack, the flow across the wing would remain attached. Fig. 9 shows the attached flow. When the rotor was added to the configuration, a downflow and a system of vortices was added to the flowfield. The downflow on the wing would in fact help keep the flow attached. However, as the vortices approach the surface, the flow in the boundary layer stagnates, and the boundary layer separates. Such stagnation has been seen and measured in Ref. 14 on a surface with no pressure gradient. Here, the situation is aggravated by the adverse pressure gradient on the wing surface. Figs. 10 a,b,c show the separated region created by vortex/wing interaction. In addition, if the vortex itself interacts with the surface, the stagnation of the flow in the vortex would create a high value of stagnation pressure, corresponding to the value of stagnation pressure seen by the rotor tip. In this case, such an interaction does not occur. Instead, the vortex appears to be flung far away from the surface.

Unsteady Separation

The separation point on the wing appears to be quite unsteady, as seen from the differences between parts (a), (b), and (c) of Fig. 10, which correspond to different values of rotor azimuth. The range of movement of the separation point is large, and must cause severe unsteady loads if it occurs on full-scale configurations.

Why does the flow not reattach?

No video frames were seen in this run where the flow was attached over the wing. One explanation (probably simplistic) is as follows. After the vortex shed by the first blade interacts with the wing, separation occurred. There was a finite time delay between the interaction of the first blade's vortex and the second blade's vortex in which the wing experienced the freestream velocity and the downflow only. During this time, the flow over the wing attempted to reattach. Before the flow was able to reattach, the vortex shed from the second blade interacted with the flow and once again separation occurred. It is concluded that the time required to reattach the flow over the wing was longer than the time between vortex/wing interactions. The video camera was able to capture this phenomenon at various stages.

Abruptness of Separation

Another interesting feature observed in flow visualization is the abruptness of the actual separation process. Fig. 10 (a) through (c) as well as Fig. 11 show the very clear demarcation between the regions of attached and separated flow. The boundary appears to be a near-vertical "jet" of fluid shooting far into the external flow from the boundary layer. We do not as yet know whether this phenomenon is two-dimensional or 3-dimensional, but it is reminiscent of the so-called "Van Dommelen - Shen singularity" which is postulated to cause the formation of abrupt 3-D jets during the interaction of vortices with boundary layers, as seen in the experiments of Ref. 15.

Post-Interaction Wake Structure

After interaction, the vortices are still visible, much more clearly in the actual video than in the hard copy, but they are more diffused. Obviously, there is some effect of the surface on the vortices and their strength must be affected. Another effect is that the wake shape is now greatly distorted, and it is quite likely that the inboard vortex sheet may exchange positions with the tip vortices. We have not yet looked for such effects in the data.

Time-Averaged Surface Pressure

Changes in the instantaneous pressure on the wing surface can be correlated with rotor azimuth and with wake features observed in flow visualization. The rotor wake adds large amounts of energy to the flow over the wing and the canard. As a result, mean static pressures on the wing cannot be determined from local flow velocities alone making the acquisition of steady and unsteady pressure data a necessity in rotor flow configurations. This distinguishes the prediction of airloads in rotor flows from predictions for fixed wings where only velocity or pressure is needed.

Static pressure taps evenly spaced along the wing chordline were used to measure the mean static pressure over the wing in the rotor wake and without rotor flow. Surface-mounted microphones were used to measure the fluctuations of pressure about the local mean at locations ahead, at and downstream of vortex interaction. In our test case, the microphones and pressure taps were interspersed along a single chordline. Ideally, a single chordline of static pressure taps and an adjacent line of microphone ports would enable the experimenter to collect both mean and unsteady pressure components at known spatial locations on the wing. To measure instantaneous pressure at a known spatial location for a given rotor azimuth, the unsteady component can be added to the interpolated value of the mean pressure at the microphone location.

Figs. 12 a - f show the mean pressure coefficient distributions for varying advance ratios and their equivalent freestream cases with no rotor flow. The mean static pressure for an advance ratio of 0.06, Fig. 12a, is dominated by the rotor downflow and the vortex/wing interaction. As would be expected from vortex interaction considerations, the rotor downflow caused an increase in the pressure on the wing's upper surface where the energized wake flow decelerates. Peaks occur in the distributions where the vortex interacted with the flow over the wing. Fig. 8a shows the vortex trajectory for an advance ratio of 0.06 approaching the
third and fourth pressure tap. A peak in \(C_p\) occurs at the third and fourth pressure tap for the 4 canard positions tested. With increased freestream interference from the canard, the greatest occurring at Position 3, the peak in \(C_p\) increases.

Fig. 12b shows a more complex mean pressure distribution over the wing for the case of \(\mu=0.075\). Near the leading edge, at the stagnation point, \(C_p\) is close to the expected value of one. As one moves over the wing the \(C_p\) decreases as if to approach a suction peak. Instead, there is a positive peak in \(C_p\) where the vortex and wing interact. The greatest peak again occurs at canard Position 3. Past the peak the pressure distribution attempts to flatten out as the pressure approaches the freestream value. It is unknown at this time why beyond the 25th pressure tap, \(C_p\) increases.

Note that as the advance ratio increases, there is a decrease in the effect of the rotor downflow on the pressure distribution and an increase in the effect of the freestream flow. For a \(\mu=0.06\), \(C_p\) is on the order of \(10^0\) while \(C_p\) for \(\mu=0.075\) and \(\mu=0.10\), is on the order of \(10^{-1}\). While Figs. 12a and 12b show a peak in the \(C_p\) distribution due to vortex/wing interactions, Fig. 12c shows the characteristic suction peak of an airfoil in freestream flow due to the convection of the vortex off and away from the wing. Flow visualization for the case \(\mu=0.10\) showed attached flow directly over the wing with a turbulent region containing the vortices above it.

**Phase-Resolved Pressure**

Figs. 13a - d show the unsteady pressure component acquired from microphone data at 6 locations for 4 canard positions for an advance ratio of 0.075. The data are synchronized with a one-per rev pulse from the rotor shaft, and are phase-averaged with a resolution of 6 degrees of rotor azimuth. This corresponds to a Nyquist frequency of 3150 Hz., far above the range of any expected pressure fluctuations. There are several unusual aspects seen here. Figure 13(a) shows 2-per rev peaks corresponding to the effects of the blade bound circulation distribution (the “Blade Passage Effect” of Refs. 2, 8, 16) as the blades pass over the microphone. These peaks are offset from 0 and 180 degrees rotor azimuth because the line of pressure sensors was offset from the longitudinal axis of symmetry of the rotor. More unusual is the small amplitude of the 2-per-rev fluctuations in most of the plots: the signal is dominated by a once-per-revolution variation, with a smaller 2-per rev. This is related to the observation that there is a mismatch between the time scales of separation and reattachment. However, if the time scales were very far apart, one would expect to see only a constant separated field, perhaps with a small 2-per-rev effect superimposed on it. The occurrence of the 1-per rev indicates that there is a severe interaction between the separation and reattachment phenomena. If the vortices from the two blades were indeed quite different in strength, their interaction points along the wing should be different, causing dual spikes in the steady pressure distribution. This is not seen.

Up until now, little has been said about the effect of the canard position on pressure measurements. When the canard is stalled, as in Figs. 13b and 13c, the magnitude of the change in \(C_p\) at the forward-most microphone is larger than for the unstalled canard.

**DISCUSSION**

This experiment was undertaken primarily to "see how long it takes" to do multi-faceted measurements while a configuration was changing. However, several surprising phenomena have emerged. The vortex-induced flow separation raises some interesting questions. A survey of current prediction methods for tilt-rotor aircraft interactional aerodynamics will reveal that the more sophisticated of the codes as yet use "uniform downflow” models of rotor wakes, or at best prescribed wake trajectories. These are far from the complex phenomena observed here. The separation is a possible contributor to the observed “download” on the wing during takeoff: the lift loss on the wing is much higher than what one would predict based on the reduction in angle of attack due to the “uniform downflow”. It must be pointed out here that the vortices in the present test case are probably much stronger than realistic tiltrotor vortices, and the wing shape and locations have certainly not been optimized for performance.

Much remains to be analyzed from the recorded information. In our first attempt at such experiments, we did "forget" a few items; most of these can be taken into account in planning for future experiments.

The wing pressure distributions and periodic variations are much more complex than expected, and appear to require explanations based on the time scales of separation and reattachment of three-dimensional boundary layers, an area where much more research is needed. We see that large excursions of separation points are to be expected, forcing vibrations. This is a problem that is rich with new fluid dynamic phenomena. Most satisfyingly, these were discovered in the process of developing measurement techniques. If such a motivation were absent, one would seek to run experiments on very carefully-designed models, with very limited areas of interest. The interesting phenomena may have remained unseen in such tests.

**CONCLUSIONS**

1. It has been demonstrated that multi-diagnostic testing using video imaging, the newly developed technique of Spatial Correlation Velocimetry, static pressure taps and surface mounted microphones is possible and yields extensive data from short-duration wind tunnel tests.

2. Parameters such as geometric configuration and advance ratio can vary continuously in the experiment. Quantities such as canard position, advance ratio and rotor azimuth are recorded on and measured directly from video images.
3. The experiments revealed massive flow separation on short-aspect-ratio wings at moderate angles of attack due to vortex interaction.

4. The location of the separation point moves rapidly over the surface during the course of each rotor revolution.

5. The lack of reattachment indicates a difference between the time scales for separation and reattachment, however, the nature of the unsteady pressure variation indicates that these time scales are of the same order of magnitude.

6. There is an increase in pressure on the wing’s upper surface due to the downflow from the rotor.

7. Peaks occur in the steady component of the pressure where the vortex interacts with the flow over the wing. The location of this pressure peak corresponds with vortex trajectory data.

8. The vortex interaction effect decreases with increasing advance ratio. For advance ratios greater than 0.1, there is no flow separation on the wing as seen from both flow visualization and pressure distributions.

9. In the wing surface region where vortex/wing interactions occur, the change in Cp reaches a minimum where the vortex and wing interact, increases as BPE occurs, decreases slightly as the other vortex is encountered and increases again as the second blade creates BPE.

10. In the wing surface region ahead of the interactions, the change in Cp is governed by the canard position. When the canard is stalled, the change in Cp is greater than for the unstalled canard configuration. Again, BPE increases Cp.

11. In the wing surface region after the separation point, changes in Cp are driven by BPE.

12. Instantaneous velocity field data for unsteady flow for a wide range of geometric configurations and flow conditions can be extracted from video images with information like canard geometry, advance ratio and rotor azimuth measured from the video.

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REFERENCES
