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Prediction and Validation**

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## CHALLENGES IN ROTORCRAFT ACOUSTIC FLIGHT PREDICTION AND VALIDATION

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### Abstract

Challenges associated with rotorcraft acoustic flight prediction and validation are examined. First, an outline of a state-of-the-art rotorcraft aeroacoustic prediction methodology is presented. Components including rotorcraft aeromechanics, high resolution reconstruction, and rotorcraft acoustic prediction are discussed. Next, to illustrate challenges and issues involved, a case study is presented in which an analysis of flight data from a specific XV-15 tiltrotor acoustic flight test is discussed in detail. Issues related to validation of methodologies using flight test data are discussed. Primary flight parameters such as velocity, altitude, and attitude are discussed and compared for repeated flight conditions. Other measured steady state flight conditions are examined for consistency and steadiness. A representative example prediction is presented and suggestions are made for future research.

### Introduction

By all accounts, rotary wing vehicles are noisy. This important class of vehicles includes conventional helicopters with a single main rotor and tail rotor, tandem helicopters, coaxial helicopters, and tiltrotors. These rotorcraft vehicles have the unique abilities to take off vertically, land vertically, and hover for long periods of time. They have become an indispensable part of the military, air medical transport, police support, and news/traffic reporting. Rotary wing vehicles such as the tiltrotor are also being examined for integration into future civilian and military transportation systems. Tiltrotors can take off and land vertically and also can fly like conventional propeller-driven aircraft during cruise. These capabilities provide a potential alternative

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means of civilian transportation that could increase airport capacity without the need to build new runways.

For all of these roles, acoustics is an important factor in the viability of the vehicle. In the case of military helicopters, acoustic detection plays a pivotal role in the mission success. To become a successful part of the civil aviation market, rotorcraft must be perceived by the public as a quiet, safe, and economical mode of transportation.

As rotorcraft become a larger part of the civilian aviation transportation system, they will be subject to the same noise management strategies already in place for fixed wing aircraft. As part of the United Nations, the International Civil Aviation Organization (ICAO) advocates a “balanced approach” to aircraft noise management policy<sup>1</sup>. This policy states that, once a noise problem is identified, four elements should be examined to alleviate the problem in the “most cost-effective manner”<sup>1</sup>. These elements are (1) source noise reduction, (2) land-use planning, (3) noise abatement procedures, and (4) operational restrictions. Ideally, source noise reduction and noise abatement procedures should be exploited first as a means to minimize the impact of land-use planning on surrounding communities and to avoid the need for operational restrictions.

To develop vehicles with reduced noise and to study various noise abatement procedures, designers need a variety of tools to be at their disposal. These tools include both noise prediction analyses and flight tests. The most cost effective way to use these tools is to flight test only the most promising low noise designs, as determined by noise prediction analyses or model testing in wind tunnels. For prediction analyses to be a viable tool to evaluate low noise designs or flight procedures, they must be accurate. However, development of accurate noise prediction tools relies on code validation<sup>2</sup>, which requires comparison to flight data. Comparisons of both primary flight parameters and acoustics present significant challenges that must be addressed in order to improve predictive capabilities. These challenges are the subject of this paper.

The remainder of the paper is divided into three parts. The first part discusses the general constitution of a state-of-the-art rotorcraft acoustic prediction methodology. The second part provides an analysis of measured data from an XV-15 flight test for representative flight conditions. The third

part presents a representative prediction example, followed by suggestions and recommendations for future research.

### **Prediction Methodology**

Current general rotorcraft acoustic prediction methodologies rely on the coupling of several different components: (1) rotorcraft aeromechanics, (2) high resolution reconstruction, and (3) acoustics. Rotorcraft aeromechanics is used to determine the aerodynamic and dynamic state of the vehicle at a low resolution. This information is then used to reconstruct the high resolution aerodynamic and dynamic state of the vehicle. This high resolution information is then utilized in an acoustic prediction method to determine the acoustic characteristics of the vehicle. A more detailed discussion each of these items will now be presented.

#### **Rotorcraft aeromechanics**

One of the primary noise sources for rotorcraft vehicles is the rotor system. As such, accurate prediction of rotorcraft acoustics depends on accurate information about the rotor state. This requires detailed information about the rotor blade motion and aerodynamics, which are strongly coupled to one another. For flight vehicles, this coupling is further complicated by the fact that the fuselage is also in motion. Determination of the rotor/vehicle state information is the job of “rotorcraft aeromechanics.”

Many aeromechanics tools, with a wide range of capabilities, have been developed in the last several decades. The objectives of these codes are to compute the vehicle “trim state,” performance, and stability and control. These quantities are computed typically for a steady state flight condition. For steady state flight, computation of the trim state requires all of the average forces and moments to be balanced on the vehicle. These averages on the rotor can be computed accurately at a low resolution, typically with 15° steps around the rotor azimuth and 15 stations radially on the rotor blade. Balance of average forces and moments is accomplished by adjusting control parameters (*e.g.*, pilot controls, vehicle attitude, *etc.*) until the specified steady flight condition is matched.

One of the first, general, rotorcraft aeromechanics codes was the Comprehensive Analytical Model of Rotorcraft Aerodynamics and Dynamics (CAMRAD)<sup>3,4</sup>, introduced in 1980. This model provided, a general two-rotor analysis capability for an isolated rotor in a wind tunnel, conventional helicopters, tandem helicopters, tiltrotors, and coaxial helicopters. Many other codes (*e.g.*, CAMRAD/JA<sup>5</sup>, CAMRADII<sup>6</sup>, 2GCHAS<sup>7</sup>, COPTER<sup>8</sup>, CHARM<sup>9</sup>, TECH-01<sup>10</sup>, TECH-02<sup>10</sup>, *etc.*) have been developed to perform similar tasks. Though these codes incorporate a wide range of aerodynamic, dynamic, and flight dynamic technologies and methodologies, their purpose remains the same – obtain

state information about the vehicle for a given flight condition.

Because the rotor system contribution to the trim state primarily depends on the average forces and moments on the rotor system, and because of the averaging process, the trim state is not a strong function of the low frequency loading distribution on the blade. As such, aeromechanics codes aim to balance the average forces and moments, not their distributions. However, it is these low frequency loading distributions, not their averages, that determine the low frequency rotor noise. Accurate computation of low frequency rotor noise requires accurate computation of low frequency loading distributions.

Because aeromechanics codes are designed to determine the vehicle trim state, they are not concerned with computation of mid- to high frequency loading. However, accurate computation of mid- to high frequency acoustics requires as input loading at commiserate frequencies. The following section discusses a method of obtaining such loadings.

#### **High Resolution Reconstruction**

Mid- to high frequency rotor tone noise is a major component of rotorcraft noise. Many useful noise metrics, including dBA (A-weighted decibel), emphasize the effects in this frequency range because the human ear is especially sensitive in this regime. An example of a mechanism that generates mid- to high frequency tone noise is a blade-vortex interaction (BVI) event. During a BVI, a rotor blade and a rotor wake vortex pass very close to one another in a near-parallel encounter. This close encounter causes a very short duration, very strong pressure pulse that radiates into the surrounding area with the familiar “wop-wop” sound.

Accurate computation of this type of tone noise requires accurate computation of high resolution blade loading gradients in both space and time. Prediction of BVI noise requires prediction of the blade loading gradients at a much higher resolution than is used in aeromechanics codes. Research has shown<sup>11</sup> that BVI noise prediction requires rotor blade loading information at 1° steps (or smaller) around the rotor azimuth and at 75 stations (or more) radially on the rotor blade. Typically, this resolution is about 2 orders of magnitude higher than that used for aeromechanics.

Several methods are available to compute information at the resolution necessary for acoustic computations in this frequency range. One technique is to use an aeromechanics code to directly compute information at a high resolution. Operating an aeromechanics code in this manner is extremely inefficient because the trim state must be established first. Because the trim state can be efficiently determined at a low resolution, direct computation of the trim state at a high resolution requires many unnecessary iterations. Another technique is to employ the aeromechanics

model at a low resolution, then kinematically interpolate the system to a higher resolution. The general technique of kinematic linear interpolation to a high resolution between low resolution data is described by Boyd, *et al.*<sup>12</sup>

The basic procedure of kinematic interpolation of data to a high resolution involves interpolation of blade position, rotor wake strength, and rotor wake position between known low resolution data. With these interpolated data, influences of all newly positioned rotor wake segments can be determined on each newly positioned blade location. These influences are used to compute the high resolution induced velocities at rotor blade locations. These velocities are then used in an unsteady aerodynamic model (*e.g.*, indicial aerodynamic method, full potential solver, *etc.*) to compute high resolution unsteady airloads. It is these high resolution airloads, along with the blade positions and geometry, that are used in the acoustic formulations to compute rotor impulsive noise and tone noise.

### Rotor acoustics

Rotor noise can be generally classified into broadband noise and discrete frequency noise. Discrete frequency noise includes impulsive noise and tone noise. Historically, due to its character, rotor broadband noise has been computed by schemes that are based on empirical methods<sup>13</sup>. Only recently have time domain approaches<sup>14</sup> shown success for simple non-rotating problems.

Routine computation of discrete frequency noise is often accomplished using Farassat Formulation 1A<sup>15</sup>. Implementations of this acoustic analogy, such as in the computer code WOPWOP<sup>16</sup>, often account for the thickness and loading noise components, but not the volumetric (or quadrupole) noise component. Thickness noise is associated with the pressure (exerted on the air) that is required to move the air out of the path of the blade. Accurate computation of thickness noise requires accurate knowledge of blade motion and blade geometry. Loading noise is associated with the pressure (or loading) that generates net forces (*e.g.*, lift and drag) on the rotor blades. Spatial and temporal variations in these pressures result in acoustic pressure signals at observer locations.

It is important to note that acoustic computations, such as in WOPWOP, typically make a number of assumptions. First, the rotor is usually assumed to be an isolated, single rotor with rigid, hinged blades. These rigid blades are usually assumed to be undergoing periodic flap, lead-lag, and pitch motion. It is also normally assumed that the rotor is undergoing constant linear translation consistent with the assumption of steady state flight, that there are no atmospheric attenuation effects, and that there are no reflection or refraction effects.

In summary, the combination of these three elements – rotorcraft aeromechanics, high resolution reconstruction, and rotor tone noise model – constitutes a current, state-of-

the-art rotorcraft acoustic methodology. A specific example of this methodology can be found in NASA Langley's TiltRotor Aeroacoustics Code (TRAC)<sup>11</sup>. Validation of any computational model of this type requires comparisons to measured data.

## Flight Measurements and Analysis

### Introduction

When comparing predicted data to measured data, a number of issues arise. One extremely important issue is the suitability of a given measured data set for a given comparison. Another issue is that a higher level of variability (relative to that found in wind tunnel testing) of data occurs in flight testing due to "real world factors" that are difficult to control. In the following sections, a subset of a well documented, well conducted flight test is examined. This examination illustrates issues that must be considered in a prediction validation study.

### Test Description

Test data examined are from an XV-15 acoustic flight test<sup>17</sup> conducted in the summer of 1995 near Waxahachie, Texas. One of the primary objectives of this test was to measure noise changes at an array of ground microphone locations due to changes in vehicle configuration and to changes in operational procedures.

The XV-15, seen in figure 1, is a tiltrotor aircraft with a conventional wing and H-tail configuration. It has two counter-rotating prop-rotors that are mechanically synchronized and that are powered by engines at the tips of the main wing. The main wing is at a fixed angle and is rigidly attached to the fuselage; the entire engine/nacelle/prop-rotor assembly can be rotated at the wing tips. This rotation, known as the "nacelle tilt angle," ranges from a vertical orientation ("helicopter mode") to a horizontal orientation ("airplane mode"). In helicopter mode, the vehicle can take off vertically as a conventional helicopter would. Once airborne, the nacelles are rotated forward ("transition mode") to forward flight, then ultimately to airplane mode for cruise flight. Further detailed design information can be found in a NASA vehicle familiarization document<sup>18</sup>.

The first phase of this test consisted of simultaneous, synchronized measurements of aircraft state data from on-board instrumentation, aircraft position data from a laser tracking system, and acoustics at an array of microphones. Data were measured for numerous flight conditions, including level fly-overs and descents. Here, a subset of the measured flight conditions will be examined. The cases studied here are listed in table 1.

Table 1: Nominal flight conditions examined.

Description	Flight # / Run #	V[kts]	$i$ [°]	$\gamma$ [°]
Level Flight 1	171a/175	90	60	0°
Level Flight 2	171b/185	90	60	0°
Level Flight 3	173a/201	90	60	0°
Level Flight 4	173b/212	90	60	0°
3° Descent	162/127	90	60	3°
6° Descent	171a/181	90	60	6°
9° Descent	168b/154	90	60	9°
9° Descent BVI	168b/148	70	85	9°

The first column of table 1 gives a descriptive label of the flight condition. The second column provides the “Flight Number” and “Run Number”. The remaining columns provide the nominal vehicle velocity,  $V$ , in knots, the nacelle tilt angle,  $i$ , in degrees, and the flight path angle,  $\gamma$ , in degrees. The nacelle tilt angle is 90° in helicopter mode and 0° in helicopter mode. A flight path angle of 0° indicates level flight; positive flight path angles indicate a constant angle descent. All level fly-overs are nominally perpendicular to the linear microphone array (see figure 2) and are nominally flown at 394 feet above the centerline of the microphone array. For descent flights, the nominal target is to descend at a constant glide slope such that, when directly over the centerline microphone, the vehicle is at 394 feet. The linear microphone is composed of 17 flush mounted, ground-board microphones; acoustic data were acquired at 20 kHz.

The first set of four flight conditions in table 1 are the most consistent set of repeated level flight conditions. For this flight test, each “flight” consisted of several “runs,” or passes, over the microphone array. Of the 17 level fly-overs in the test, 4 were judged to be “acceptable” for this study. Acceptability was based on a lack of adverse comments in the test log or flight log for the particular case<sup>19</sup>.

#### Aircraft State Data

When performing a flight acoustic prediction or validation, one needs to have knowledge of the flight condition for which the prediction is to be made. Often, the nominal flight condition is used in the prediction. However, when comparing measured data to predicted data, the deviations between these measured and predicted flight conditions and state data must be assessed.

Because of its fundamental effects in the determination of the aircraft state, one of the most important items to be assessed is the vehicle airspeed. Figure 3 shows the airspeed for all four of the repeated level flight cases (“runs”) from

table 1. Flight time is shown on the horizontal axis in seconds. The origin on the time axis is synchronized with the aircraft being directly over the array. Negative time indicates that the aircraft has not reached the array; positive time indicates that the aircraft has passed. This time scale is consistent throughout the following figures of this type. The vertical axis shows the measured airspeed in knots. The nominal (intended) airspeed of 90 knots is shown in red. It can be seen that all of the repeated flights exceeded the nominal airspeed, and had variations of up to approximately 5 knots, or 5% of the nominal. Though this 5% difference in airspeed appears small, it can have a large impact on the vehicle state, loading, and thus, noise generation. For example, because forces on the wing are proportional to the square of the velocity, a 5 knot increase in speed at the same angle of attack can result in approximately 11% larger forces generated by the wing. If the wing loading changes by this amount, to maintain a steady flight condition, the rotor loading must be reduced by the same amount.

Another example of data variability is seen in figure 4. Using the same axis scales as in figure 3, the three constant angle descent cases with a nominal airspeed of 90 knots, plus the BVI flight condition at a nominal airspeed of 70 knots, are plotted. The 3° descent case has a slightly lower airspeed than the nominal, but is relatively constant. The 6° descent case has an average airspeed approaching the nominal, but has more variation than the 3° descent. The 6° descent condition exhibits some variation in airspeed; the 9° descent case (at 90 knots) shows an even larger variation. Near the overhead position for the 9° descent, the aircraft performs what can be considered a maneuver. More precisely, the aircraft slowly increases speed until about 4 seconds before being overhead. At that point, it decelerates at approximately 0.1g for nearly 8 seconds, then accelerates at approximately 0.1g for an additional 5 seconds. Because the XV-15 is a 13,000 pound aircraft, a 0.2g change in acceleration results in an effective rapid vehicle weight change of about 2,600 pounds. This, too, can drastically affect the vehicle state, loading, and thus, its noise generation.

The root of these rapid changes can be seen in figure 5, which shows the flight path of the descent cases. The nominal lines (in red) indicate the 3°, 6°, and 9° nominal descent paths. The black lines show the actual flight path flown. The 3° descent is seen to be nearly constant; it closely matches the nominal flight path. This is in concert with the constant, steady airspeed seen for this case in figure 4. Curiously, the 6° and 9° descent cases start at almost the exact same altitude. They both follow the nominal 9° descent profile from 20 seconds to 10 seconds prior to the overhead position. Examination of the 6° descent case reveals that, starting near 10 seconds prior to the overhead position, the aircraft begins a descent in excess of 9° in order to get onto the intended 6° flight path. This is the root of the acceleration seen at this time in the 6° run in figure 4. From the overhead position to the end of the run, the 6° case attains, then main-

tains, the appropriate  $6^\circ$  glideslope. Again, this matches well with the nearly constant airspeed seen in figure 4 for this time frame.

Examination of the  $9^\circ$  descent case shows that the aircraft initially follows the  $9^\circ$  descent profile until approximately 5 seconds before the overhead position. At that time, the vehicle attitude changes to a  $6^\circ$ , then to a nearly  $3^\circ$ , then back to a  $6^\circ$  glideslope, and maintains this  $6^\circ$  glideslope for the remainder of the descent. Again, these flight path changes closely correspond to the airspeed changes in figure 4.

Another stark feature in figure 5 is that the  $9^\circ$  BVI descent case actually follows the wrong flight path angle for the entire flight; the vehicle is flown on a  $6^\circ$  descent profile instead of a  $9^\circ$  profile. It should be noted that is not a criticism of the flight test or pilots. In fact, this was a well designed, well executed test flight. These discrepancies are emphasized here solely to illustrate that extreme care must be taken when deciding what flight conditions to use in prediction methodology development and validation. Here, for example, if only nominal conditions are considered for the prediction without close examination of the measured state data, and if only acoustic results and measurements were compared, a poor comparison is nearly assured.

Another aircraft state parameter that is closely related to the airspeed and flight path angle is the fuselage pitch angle. Because fuselage pitch can not be directly controlled by the pilot, there is no nominal target for it. However, for a given nominal steady state flight condition, this orientation should be a constant. Figure 6 shows the measured fuselage pitch angle for the repeated level fly-overs. For a given run, the maximum variation of pitch is approximately  $2^\circ$ . Comparing all of the repeated runs, the range of variation is as much as  $5^\circ$ . This variation can substantially affect the forces and moments on the vehicle and the force balance between the rotors and the wing (*e.g.*, “lift share”). Again, changing these state parameters and loading can have a substantial impact on the rotorcraft aerodynamics and acoustics.

A similar examination of the fuselage pitch for the descent flights is shown in figure 7. Because all of the descent flights conditions are nominally at different descent angles, the nominal pitch attitude should be constant for each case; this constant will be different for each case. Consistent with previous figures, the pitch attitude for the  $3^\circ$  descent and the  $9^\circ$  BVI descent are well behaved and do not show much variation. Also consistent with previous airspeed and flight path discussions, the  $6^\circ$  and  $9^\circ$  cases show large, rapid changes in fuselage pitch. The combination of rapid fuselage pitch and the rapid velocity changes results in rapid angle of attack and loading changes. For a prediction methodology, this is troublesome because aeromechanics prediction methodologies currently trim the aircraft for steady state flight conditions. More detailed discussions of for this particular test can be found in Boyd, *et al.*<sup>19</sup>.

Other flight measurement issues can hamper efforts to develop a validated comprehensive analysis or noise prediction model. One issue is that it is not practical, or is impossible, to measure some parameters that are needed to validate computations. For example, as stated before, the vehicle trim state depends on balancing forces and moments on the entire vehicle. But, current prediction models determine the vehicle trim by computing forces and moments on each component, then trim the entire vehicle by adjusting individual components. There is currently no practical method to measure, for example, the thrust from just the rotor system. So, if there is a discrepancy in the measured and predicted trim state, there is no method available to isolate the source of the discrepancy, even on a component basis. At this time, predictive improvements for each component (rotor system, fuselage aerodynamics, *etc.*) rely on wind tunnel tests of isolated components.

A similar problem arises when trying to measure interactions between aircraft components. For example, there are no practical methods currently available to directly measure the aerodynamic influence of the rotor wake on the wing lift and drag for a flight vehicle. In the past, it has been necessary to obtain this information by inference from measurements made on models in a wind tunnel.

### Acoustics

The focus of the acoustic measurements for this 1995 test was to compare acoustic footprints for a variety of terminal area operations. Of particular importance were footprint changes due to different flight conditions. These comparisons are typically made using contoured, integrated noise metric maps, such as A-weighted, overall sound pressure level (OASPL). Conner, *et al.*<sup>17</sup> shows results in this format which illustrate the community impact of aircraft operations. It is also important to remember that these footprints are derived using integrated spectra from the acoustic pressure time histories. Due to this integration, the waveform of the acoustic pressure time history is not retained. Because many different acoustic pressure time histories and spectra can result in the same integrated metric, hinging a prediction validation on the integrated metrics can be misleading. The prediction method may produce the correct integrated quantity while completely missing the physics of the problem.

To elucidate this, acoustic pressure time histories will be examined. Instantaneous pressure time history samples are used here to show blade-to-blade variability in the data. In each repeated run, the flight conditions are not duplicated exactly due to “real world” effects. To facilitate comparisons on an equal footing, all time histories have been processed as described by Boyd, *et al.*<sup>19</sup> to (1) remove Doppler effects by accounting for the local flight Mach number and the acoustic emission angle, and (2) remove spherical

spreading effects by normalizing the acoustic pressure amplitude to a constant distance (394 feet) from the vehicle.

First, the centerline microphone (#9) will be examined for the level fly-over runs listed in table 1. For illustration purposes, the examination will be limited to a representative acoustic pressure time history that begins at 10 seconds before the aircraft is overhead. This corresponds to “time[sec] = -10” location on the horizontal axis of figure 3. Figure 8 shows the measured acoustic pressure time history at microphone #9. With the aircraft in its nominal state at this location, the acoustic time history theoretically (1) should be dominated by thickness noise because the microphone array is nearly in the plane of the rotor, (2) should only show a single thickness noise peak due to the identical phasing of the rotor thickness noise from each rotor, and (3) should be identical between the repeated runs.

It is clear from figure 8 that there are substantial differences between the repeated runs. In run 1, for example, there are two pulses present inside the large pulse. Run 2 exhibits a similar behavior, but not to the same extent as run 1. Run 3 displays a completely different acoustic waveform than any of the other three runs and it is clear that these two signals are not in-phase with each other. Only run 4 exhibits the expected behavior with only a large single pulse. For comparison purposes, table 2 lists a modified OASPL, for all four repeated level flight runs. These values are dubbed “modified” OASPL because they are computed using the normalized, de-Dopplerized acoustic pressure time histories.

Table 2: Measured decibel levels for time histories at microphone #9 at 10 seconds prior to overhead.

Description	dB
Level Flight 1	106.7
Level Flight 2	106.6
Level Flight 3	99.6
Level Flight 4	107.1

From the standpoint of comparing a predicted acoustic time history to one of these measured time histories, conclusions drawn from a prediction would depend greatly on which of the four repeated runs were chosen for the comparison. Boyd, *et al.*<sup>19</sup> shows that this finding is not limited to the “10 second prior to overhead” location. Similar differences persist throughout the measured time range and similar variations appear in the symmetrically placed side-line microphones<sup>19</sup>. Boyd, *et al.*<sup>19</sup> explained the double pulse character by including the effects of vehicle orientation and position on the acoustic pressure amplitudes and phasing of the signals from each rotor.

## **Representative Example**

To illustrate the effects that the non-nominal parameters can have on the prediction of an acoustic pressure time history, an example result<sup>19</sup> from the TiltRotor Aeroacoustic Code (TRAC) system is given here. Figure 9 shows two plots. The first plot (labeled “Nominal”) is a uses only the nominal conditions in a prediction of the acoustic pressure time history corresponding to the same location as shown in figure 8. A single acoustic pulse is seen, as is expected from a symmetric aircraft flying over a centerline microphone.

The second plot in figure 9 (labeled “Nominal + Orientation”) is used to illustrate the effects of accounting for vehicle/rotor position and orientation in the acoustic prediction. In this case, the identical loading from the nominal prediction has been used. However, each rotor has been positioned and oriented in the acoustic prediction using the measured vehicle spatial location from laser tracking data, and the measured pitch, roll, and yaw angles from on-board measurements of the Level Flight 3 (“Run 3”) case. This illustrates that predicting the vehicle orientation correctly is necessary to properly account for phasing of rotor signals.

Table 3: Predicted decibel levels for time histories at microphone #9 at 10 seconds prior to overhead.

Description	dB
Nominal	103.8
Nominal + Orientation	98.8

Table 3 shows the predicted decibel levels for the plots in figure 9. The level for the prediction using nominal data is shows approximately a 3 dB under-prediction compared to the level flights numbered 1, 2, and 4. However, if Run 3 had been “blindly” chosen for comparison with a nominal prediction of decibel levels, a 4 dB over-prediction would be obtained. Without examination and comparison of the acoustic pressure time histories and vehicle state data, the source of the over-prediction would be difficult, if not impossible, to find.

## **Future Research**

### **Prediction methods**

There are a number of issues that need to be addressed in order to improve the state-of-the-art in rotorcraft acoustic prediction methodologies. These issues exist in all three areas that constitute a comprehensive rotorcraft acoustic model: comprehensive analysis, high resolution reconstruction, and acoustics.

### Comprehensive Analysis

Almost universally, comprehensive analyses use a trim flight condition that is assumed to be steady state or quasi-steady state. However, many important flight conditions involve a maneuver that can not be considered as steady state or quasi-steady state. Comprehensive analysis methods that do deal with maneuvers normally handle them as “transient” events. That is, after the steady state flight condition is achieved, the vehicle response to a changes in a system parameters is computed. For example, a transient analysis can compute the vehicle response to a pilot inputs to the collective stick, cyclic stick, or rudder pedal.

Transient computations typically rely on the accuracy of the steady state trim computation. To obtain a steady state trim, the vehicle equations of motion are iterated until changes in trim variables fall below a specified tolerance. Even though, computationally, the vehicle is considered “trimmed,” because this tolerance is non-zero, there are undetermined (*i.e.*, residual) forces and moments acting on the vehicle. Though this is acceptable for an iterated trim solution, these residual forces and moments will eventually cause a time-marching transient solution to diverge in an unpredictable manner. The solution to this problem is still a subject of research.

The transient problem discussed above is known as a “direct problem”. That is, the vehicle response to system inputs is computed. Determination of the vehicle state based on a given flight condition is known as the “inverse problem”. Iterated trim solutions normally fall under this category. Though the capability to compute a trim state using the inverse method has been used for many years, a general method to solve for the vehicle configuration needed to fly a particular maneuver does not currently exist. For example, given a general maneuvering flight path, it is not currently possible in general to determine the pilot controls necessary to fly that given path. Though, some initial progress has been made recently in this area<sup>20</sup>, much research still remains to develop a general method.

One difficulty in implementing the inverse problem for maneuvering flight in current comprehensive codes is that techniques used to solve the trim problem normally take advantage of the simplifications associated with an assumption of periodicity that arises from the steady state assumption. In a maneuver, the assumption of periodicity for the rotor is no longer valid. For many comprehensive codes, removal of this assumption would require substantial code revision and possibly even requiring a complete re-write of the code itself.

### High resolution reconstruction

Current methods used to reconstruct a high resolution solution from a low resolution trim solution<sup>11</sup> also rely on the assumption of periodicity inherent in the low resolution

trim solution. To have a general reconstruction method applicable to maneuvers, a method is needed that could reconstruct a general time history of blade motions, wake trajectories, wake strengths, influences, *etc.* without including the assumption of periodicity. This method should be modular and independent of the underlying low resolution comprehensive aeromechanics.

In addition, current methods to reconstruct the high resolution solution use linear interpolation for nearly all aspects of the reconstruction process. Linear interpolation, though straightforward to implement in a complex problem such as this, can lead to problems when interpolating highly curved structures. An example of this is the kinematic motion of, and linear interpolation of, a tip wake structure past a rotor blade. This situation can lead to artificial flow features<sup>12,21</sup> that would not exist in the continuous system. Many of these issues, dealt with on a case-by-case basis currently, could potentially be resolved with a higher order interpolation scheme that more closely mimics the continuous system while retaining the information from the low resolution analysis.

### Acoustics

The current routine-use, and arguably state-of-the-art, rotorcraft acoustic prediction method for rotor tone (discrete frequency) noise is embodied in WOPWOP. Recent research<sup>22,23</sup> at the Pennsylvania State University and NASA Langley has led to the development of a completely new version of WOPWOP, known as WOPWOP-PSU. This new methodology now includes the capability to compute rotor discrete frequency noise from multiple rotors undergoing arbitrary, rigid-blade motion. That is, the periodicity assumption has been removed from the acoustic computation and many computational enhancements have been added.

Even though the capability to compute rotor tone noise for maneuvering aircraft is emerging, there are still a number of unresolved issues in rotorcraft acoustics that need further research. For example, for a flight vehicle, propagation of an acoustic signal through the atmosphere, especially over long distances, can have a profound effect on the signal itself. Atmospheric effects such as wind, temperature, moisture, and absorption can greatly influence the propagation path and frequency content. For several years, the Rotor Noise Model<sup>24</sup> (RNM) has been widely used to compute atmospheric propagation effects for rotorcraft acoustics. RNM originally was designed to compute noise footprints for steady state flight conditions by propagating noise from a “noise hemisphere” to the ground along straight acoustic ray paths. Future planned research includes adding the capability to account for the effects of wind. Though RNM includes ground reflection, echoes from surfaces such as canyon walls or buildings is not included. This can be an important factor for future urban heliports, but will require

further research. Continued work is also needed to improve the capabilities for maneuvering flight computations in RNM. Currently, RNM propagates noise from a hemisphere below the rotorcraft to an observer. Maneuvering computations will require knowledge of noise on a sphere surrounding the entire vehicle.

### Flight measurements

As future maneuver codes are developed, they will need to be validated with flight data. However, development of techniques to measure aircraft state and acoustic data for maneuvering flight vehicles is even more difficult than for steady state conditions. Innovative measurement techniques will be required.

Currently, the state-of-the-art in acoustic flight measurements can provide simultaneous acquisition of vehicle state information from on-board systems and acoustic data from a linear microphone array or a distributed microphone array. Using the linear array, and the assumption of a steady state flight condition, noise hemispheres can be measured for use in, and validation of, RNM. However, for a maneuver, the linear array technique is no longer valid because the flight condition is continuously changing. A distributed array could be used, however, this presents many difficulties also. For example, because a distributed array needs to be deployed over large tracts of land, practical issues arise involving microphone deployment methods, microphone layout geometries, transmission of signals in long cables, calibration of many distributed microphones, *etc.*

Even with a distributed array, measurement of maneuver acoustics would require multiple repeated fly-overs at the same maneuver condition to completely map the acoustic field. It has been seen already that accurately repeating flight conditions, even under the best of circumstances, can only be expected to occur within certain limits or tolerances. Because these flight condition parameters can only be controlled to a certain degree, the prediction method or comparison method must be able to account for these variations.

Another issue involves measurement of, and use of, noise hemispheres. As mentioned above, noise spheres are needed for maneuvering aircraft in RNM. These noise spheres encompass the entire vehicle and measurement of such a sphere would require either that the vehicle to fly through a three dimensional distributed microphone array or that the vehicle have an array moving with it.

In addition to these difficulties, data processing techniques must be developed to routinely handle non-periodic information. Currently, methods used to compute acoustic spectra from acoustic time histories rely some form of averaging and periodic analysis (*e.g.*, Fourier analysis, *etc.*). Future research in flight acoustics should include exploration of new data analysis techniques that use time-frequency analysis techniques, such as found in wavelet analyses<sup>25</sup>.

### Conclusions

This paper has explored some of the general issues related to development of, and validation of, a rotorcraft acoustic prediction methodology. Development of a prediction methodology was discussed in general terms and was described as a component-wise outline of a state-of-the-art methodology. The outline discussed rotorcraft aeromechanics, high resolution reconstruction, and acoustic computations.

A specific flight test of an XV-15 tiltrotor aircraft was examined. It was seen that primary flight parameters such as velocity, altitude, glideslope, and attitude can only be controlled to a certain degree, even under controlled conditions. It was also shown that acoustic pressure time histories, which are necessary to validate an acoustic prediction method, can vary greatly even for nominally repeated flights. It was seen that extreme care must be exercised when choosing data to be used in a validation exercise.

Some future research topics were also discussed. For low frequency acoustic predictions, continued research is necessary in rotorcraft aeromechanics to improve low frequency blade loading. This will require not only improvements in predictive capabilities, but also improvements and innovations in flight measurement technologies. Improvements in flight measurement technologies need to include methods to measure individual component loads, such as rotor loading, wing loading, fuselage loading, *etc.* Improvements in high resolution reconstruction techniques should include higher order kinematic interpolation methods that are independent of the underlying rotorcraft aeromechanics method.

Rotor discrete frequency noise computations were also discussed. Though current state-of-the-art codes can compute rotor tone noise for rather general rotor configurations, accurate prediction of acoustics depends on accurate prediction of the loading and state information from the underlying rotorcraft aeromechanics code.

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## Figures



Figure 1: XV-15 tiltrotor aircraft.

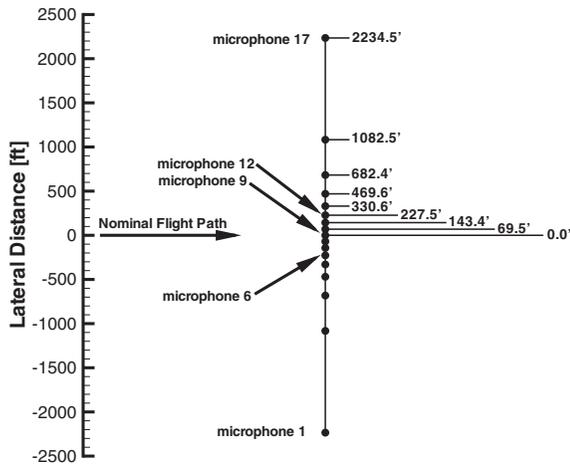


Figure 2: Linear microphone array for Phase I of 1995 XV-15 flight test.

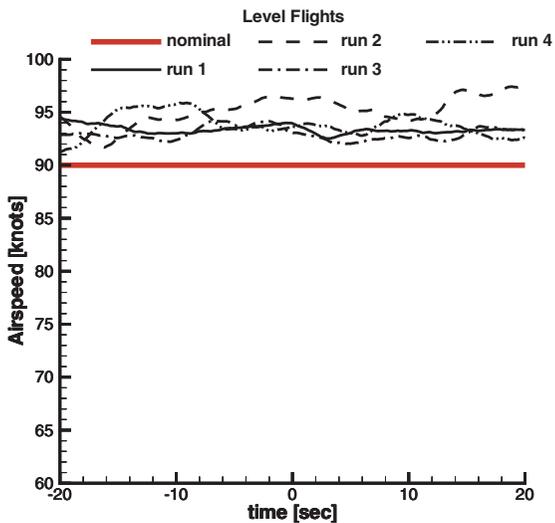


Figure 3: Measured airspeed for level flight runs.

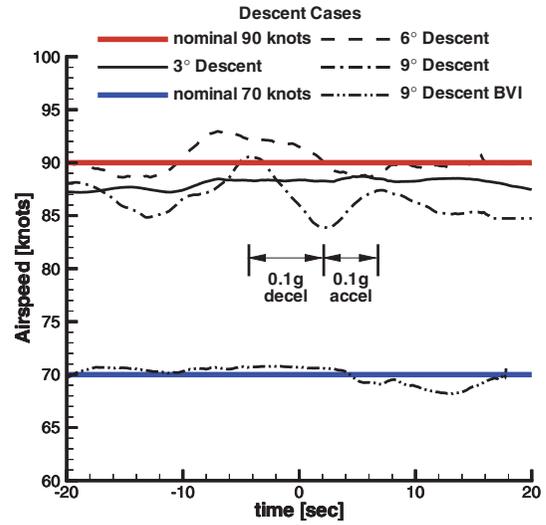


Figure 4: Measured airspeed for descent flight runs.

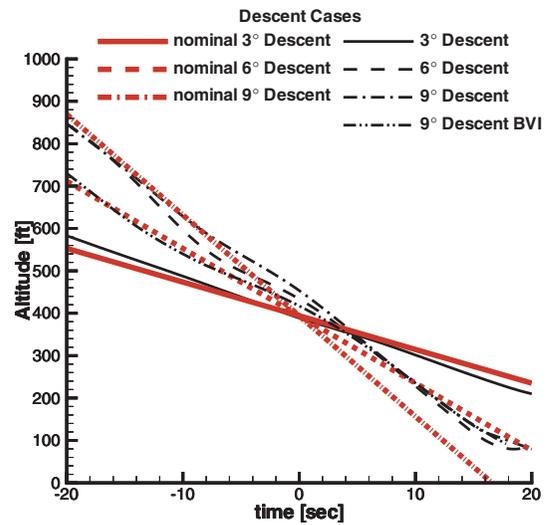


Figure 5: Measured altitude (flight path) for descent runs.

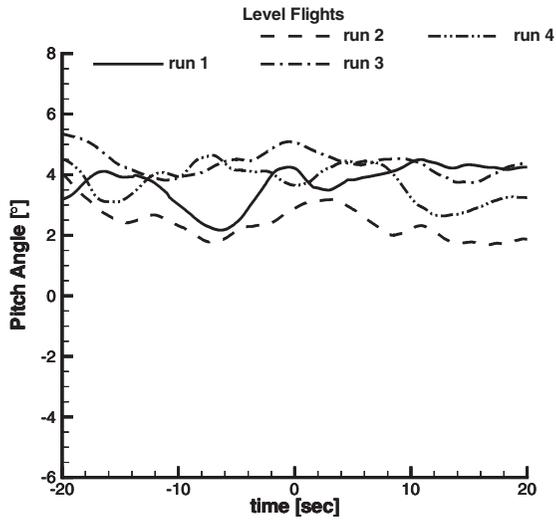


Figure 6: Measured pitch angle for level flights.

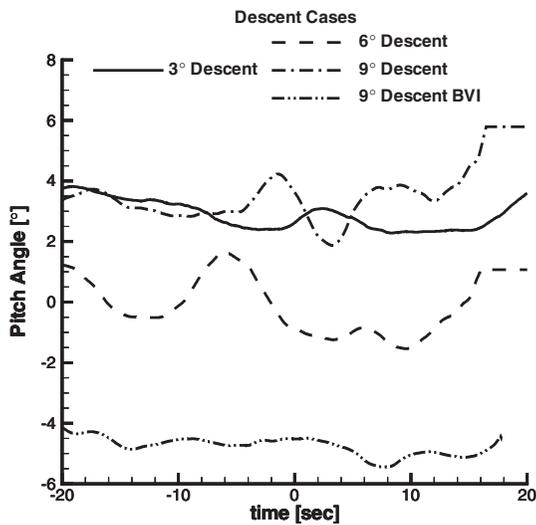


Figure 7: Measured pitch angle for descent flights.

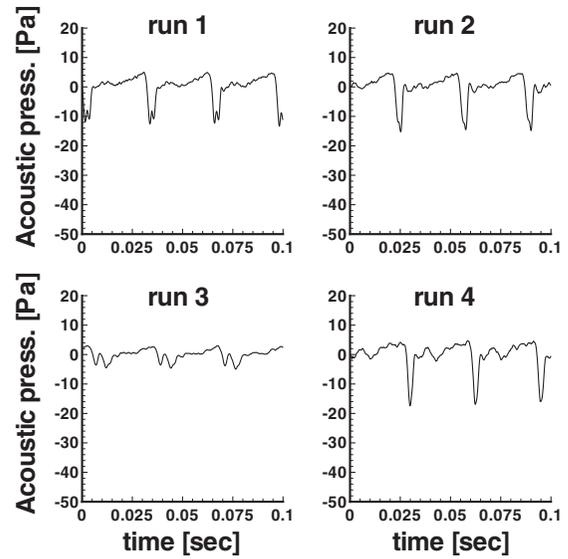


Figure 8: Measured acoustic pressure time histories for repeated level flight runs at microphone #9 at 10 seconds prior to the overhead location.

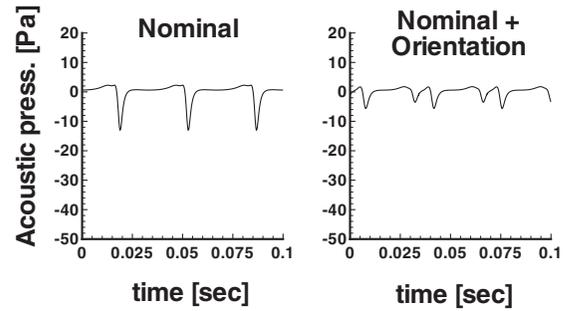


Figure 9: Predicted acoustic pressure time histories. “Nominal” uses only nominal conditions. “Nominal + Orientation” includes measured vehicle orientation from Run 3.