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Helicopter Vibration Reduction and Damping Enhancement Using Individual Blade Control

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A control law is presented for a BO-105 helicopter to reduce vibration and to increase damping by the use of individual blade control. H_∞ control synthesis is used to develop a robust controller usable in different operating conditions with different helicopter flight speeds. In simulation, hub load vibration can be canceled (−99%) in three outputs simultaneously, for example in all three hub forces. A reduction in hub vibration, however, does not necessarily lead to reduced vibration in the cabin. Therefore, a finite element model of the flexible fuselage is coupled with the aeromechanical rotor model. The resulting coupled rotor–fuselage model allows vibration to be calculated and controlled at locations in the cabin. A simultaneous vibration reduction of −89% is achieved at the pilot and copilot seats. To reduce gust sensitivity, lag damping must be enhanced, which requires the lag rate of the blades to be fed back. However, the control law is developed with the constraint of no sensors and, consequently, no measurements in the rotating blades. The use of a model-based control strategy enables lag damping to be enhanced from 0.5 to 2–3% critical damping by feedback of the observed lag rates, requiring only measurements of the hub loads.

Nomenclature

A, B, C, D	=	state-space matrices
a	=	acceleration (with subscript)
$\boldsymbol{d}$	=	disturbance at plant output
$\boldsymbol{d}'$	=	disturbance at plant input
F	=	force (with subscript)
F_∞	=	state feedback matrix
G	=	plant
K	=	controller
L_∞	=	H_∞ control matrix
M	=	moment (with subscript)
N	=	number of rotor blades
$\boldsymbol{r}$	=	reference signal
S_y	=	output sensitivity
T	=	period
t	=	time
$\boldsymbol{u}$	=	input vector
$\boldsymbol{u}'$	=	input vector with disturbance
$\boldsymbol{v}$	=	control deviation
W	=	weighting function (with subscript)
$\boldsymbol{w}$	=	exogenous input
X_∞	=	H_∞ control matrix
$\boldsymbol{x}$	=	state vector
$\boldsymbol{y}$	=	output vector
Z_∞	=	H_∞ control matrix
$\boldsymbol{z}$	=	controlled output
γ	=	H_∞ norm of augmented closed-loop system
ζ	=	critical damping
θ	=	blade pitch angle

Subscripts

K	=	controller
x, y, z	=	x, y, z axis
$(1), \ldots, (N)$	=	blade degrees of freedom

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I. Introduction

HELICOPTERS suffer from high vibration. This vibration is mostly caused by the main rotor, which operates in a complex aerodynamic flowfield. Changing aerodynamic effects, pilot inputs via the swashplate, and interactions of blades and vortices of preceding blades are causes of oscillations in the flexible rotor blades that are transmitted through the rotor hub and cause vibration in the fuselage. Fuselage vibration leads to human discomfort and fatigue damage of structural components, both of which give reason to aim for helicopter vibration reduction.

Concepts for active vibration control are higher harmonic control (HHC) and individual blade control (IBC). Both methods aim at modifying existing and/or inducing additional forces and moments at the rotor that are opposite in phase and equal in amplitude with the original forces and moments, leading to destructive interference. The original vibration is consequently reduced and ideally canceled out. By IBC, the blades are individually controlled in the rotating frame above the swashplate. In the concept of individual blade *root* control, the lift of the blade is varied by changes in the pitch of the blade at its root. Therefore, the pitch link rods are substituted by hydraulic actuators, allowing a blade pitch control to be superimposed to the swashplate commands. A four-blade BO-105 helicopter equipped with an individual blade root control system is considered here.[1]

One advantage of individual blade root control over other IBC concepts via flaps, twist, etc., is that no changes to the blade are necessary. Thus, the blades do not need to be recertified. However, if the original blades are used, noblade sensors are available and, consequently, no measurements are available in the rotating frame. The availability of only hub load (and possibly fuselage) sensors imposes certain restrictions on the design of the control law, which this paper seeks to examine.

Helicopter vibration reduction belongs to the class of vibration control problems for plants with periodic coefficients. Here, periodicity is a result of the mechanics of the system and cannot be avoided. A linear time-periodic system responds to a sinusoidal input not only with a sinusoid at the excitation frequency, as linear time-constant systems do, but also at additional harmonic frequencies that are spaced by multiples of the plant-periodic frequency.[2] The introduction of multiblade coordinates[3] (MBC) opens up the possibility of the use of a wide range of time-invariant controller synthesis techniques because the periodicity of the plant is (partly) preserved in a time-constant MBC representation.

The lightly damped lag motion of the rotor blade is susceptible to various aeroelastic and aeromechanical instabilities. This is the reason why most rotors have mechanical lag dampers that provide additional damping to suppress the occurrence of such aeromechanical phenomena.[4] Ham et al.[5] describe how IBC can be used to increase damping actively. However, if the advantage of individual blade root control, namely, the usability of unchanged blades, is to be exploited, control strategies without lag rate sensing are required. One of the main contributions of this paper is the development of a robust observer-based controller to increase lag damping without blade sensors. Preliminary results of this research have been published in Ref. 6.

Friedmann and Millott[7] give a comparison of various approaches for vibration reduction in rotorcraft by the use of active control. Linear quadratic control is used in Ref. 8 to control six hub forces and moments, and in Ref. 9 to control the hub force in the thrust direction by the use of collective pitch only. Optimal output feedback control strategies are presented in Ref. 10. Aspects of periodic system control are presented in Ref. 11. Active control of fuselage vibration is described in Refs. 12 and 13. Methods for helicopter simulation are presented in Refs. 14 and 15.

An elementary issue in helicopter vibration control is the selection of outputs to be controlled. For a four-blade rotor, typically three hub forces/moments are chosen, for example, the force vector or a combination of the force in thrust direction and the roll and pitch moment.[10] However, a reduction of vibration in selected hub loads can lead to reduced or increased vibration in the remaining hub loads, depending on the dynamic properties of the system. However, vibration is typically increased in outputs not considered for vibration reduction, which is obviously counterproductive to the aim of reduction of vibration that is transmitted to the fuselage.

The final goal of helicopter vibration reduction is to reduce vibration, not necessarily at the rotor hub, but at specific points in the fuselage, for example, at the pilot and copilot seats and in the load compartment. IBC inputs aimed at the reduction of hub loads may not necessarily lead to a simultaneous reduction in the accelerations at specific locations in a flexible fuselage. In Ref. 7, an increase in fuselage acceleration by a factor of two to five from its baseline value was observed when higher harmonic control inputs aimed at minimization of hub shears were introduced. This shows that the model on which the controller design is based is required to include both rotor dynamics and fuselage dynamics. To cope with this, a finite element model of the BO-105 helicopter is integrated with the aeromechanical model of the rotor to develop controllers aimed at minimization of vibration at various locations in the fuselage.

The paper is organized as follows: First, the models used for analysis and design are described, followed by an explanation of the control law design. Next, vibration reduction results are presented, and lag damping enhancement is discussed. Finally, some conclusions are presented.

II. BO-105 Camrad II Rotor Model

Here, a four-bladed BO105 helicopter equipped with an individual blade root control system is considered.[1] The actuation system consists of a conventional swashplate with the pitch rod links substituted by hydraulic actuators. The primary flight control works traditionally via swashplate. The secondary antivibration control is superimposed via the hydraulic actuators in the rotating frame (Fig. 1). Hydraulic power is transported to the rotating frame via a hydraulic slip ring. The effect of IBC on rotor power is difficult to measure because it is only a small percentage of baseline rotor power. Friedmann and Millott[7] calculate control power required for IBC to be approximately 3–4% of rotor power, which might serve as an upper-bound estimate. In contrast to that, other references suggest IBC as a means to *reduce* rotor power.[1,16,17]

The aeromechanical analysis software Camrad II[18] is used to derive a state-space model of a BO-105 helicopter rotor with four flexible blades.[10] The rotor blades are modeled as beams by finite elements, which allows flap (out-of-plane), lag (in-plane), and torsion motion. Aerodynamics are calculated by the use of blade element theory, combined with experimental airfoil tables, a wake model, and uniform inflow theory. The structural blade modes considered are four flap modes, two lag modes, and the first torsion mode. Eight finite elements per blade were used to model the rotor. The highest frequency considered here is the fourth flap mode with 7.97/rev. This was found to be sufficient to predict results for frequencies of up to around 10/rev. Higher-frequency structural and aerodynamical modes, as well as flight mechanical modes, are neglected. Aerodynamic effects of the fuselage and stabilizer are considered by lookup table data. The tail rotor is treated as a rigid rotor without degrees of freedom. The actuators are modeled with first-order dynamics.

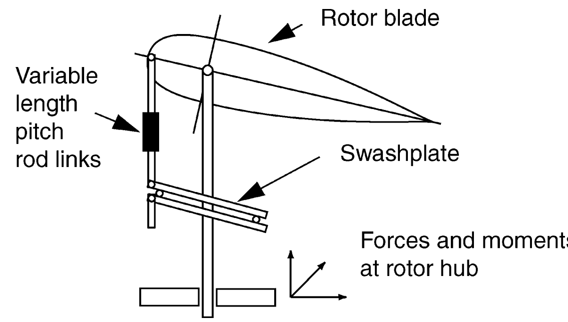

Fig. 1 Individual blade root control.

The resulting model is linearized at specific operating conditions and can be either linear constant or linear time periodic. The IBC inputs $\boldsymbol{u}$ in the model represent the hydraulic actuator commands. Outputs $\boldsymbol{y}$ are the forces in all three directions and the roll and pitch moments at the rotor hub caused by IBC inputs. The yaw moment is usually not considered in helicopter vibration control.[19] The model includes two elastic blade lag modes, four flap modes, and one torsional mode, for a total of 56 first-order states $\boldsymbol{x}$. The forces and moments at the hub due to flight, called baseline vibrations, are modeled as output disturbance $\boldsymbol{d}$. Because of a mechanical filtering effect,[20] these baseline vibrations occur at frequencies of iN/rev, with i as integer and N as the number of blades. Vibrations with $i = 1$ are most dominant and solely considered here.

The state and input vectors in the model equation

$$\dot{\boldsymbol{x}} = A(t)\boldsymbol{x} + B(t)\boldsymbol{x}, \qquad t \in [0, T]$$

$$\boldsymbol{y} = C(t)\boldsymbol{x} + D(t)\boldsymbol{u} + \boldsymbol{d}$$

are given in MBC that result from a linear Fourier coordinate transformation from single-blade coordinates (SBC).[3] This coordinate transformation partly retains the periodicity of the system when time-constant models are used.[20]

III. BO-105 Fuselage Model

This section presents the fuselage model used to calculate and control vibration at locations in the cabin, such as at the pilot and copilot seats and in the load compartment. An existing finite element model of the fuselage[21] is used to obtain mode shapes of the flexible structure. The following summary outlines how the mode shapes are implemented into the model in Camrad II and result in a coupled rotor–fuselage model. (See Ref. 22 for a detailed description.) The rotor and fuselage are coupled via pitch/mast bending.[18] This is implemented by specification of the mode shapes (both linear and angular) at the hub. The contribution of the swashplate to coupling is neglected. The fuselage model was developed with the finite element software NASTRAN.[23] The fuselage model consists of 812 points with three translatory and three rotatory degrees of freedom, for a total number of 4872 degrees of freedom.

Based on an analysis[22] of the transfer function from IBC inputs to accelerations in the fuselage, there are no significant gain/phase changes from consideration of more than 17 modes. Therefore, 17 fuselage modes are used for the coupled rotor–fuselage model. The highest fuselage mode frequency considered here is 5.68/rev.

In Camrad II, the rigid motion describes the center of mass motion of the helicopter. The rigid-body degrees of freedom are orthogonal to the free-vibration modes.[18] The modes are calculated for a free body, inclusive of rigid motion. Small motion is assumed, consistent with the linear modes. Consequently, the center of mass motion does not affect the elastic modes, and the elastic motion does not affect the inertial properties of the rigid modes.

In the NASTRAN model, the rotor is modeled with a point mass to calculate the center of mass and approximate the influence of the inertia on the modes. Only the elastic modes need to be imported from NASTRAN into Camrad II. The rigid-body modes, therefore, must be truncated. Aerodynamics influences of the rotor on the damping and stiffness properties of the fuselage modes are neglected.

IV. Controller Design

The two main objectives of the controller are 1) vibration reduction and 2) damping enhancement. In helicopter flight, forces and moments generated by the rotor act on the rotor hub and cause vibration in the fuselage. The antivibration controller is supposed to calculate suitable commands for the IBC inputs to modify existing and/or provoke additional forces and moments that are equal in amplitude but opposite in phase to the original forces and moments at the hub. If successful, the forces and moments cancel each other out. The sum of the original forces and moments and of the IBC-induced forces and moments transmitted to the fuselage is reduced, and the objective of vibration reduction is achieved. The second design objective is to increase damping; this is demanded for the weakly damped lag modes. An increase in damping results in an improved transient response to disturbances.

Vibration control is assumed to be independent of primary flight control. This is justified because the typical frequency range of primary flight control (approximately 0.3–12 rad/s, corresponding to 0.007–0.27/rev for the BO-105 helicopter[24]) and the frequency range of interest for vibration control (around 3–5/rev) are clearly separated. Experimental evidence for this is provided by the fact that during the BO-105 flight-test series, none of the pilots reported any noticeable impact of HHC on helicopter trim or handling qualities.[25]

The objective of increased damping in the lag modes requires the lag rate of the blades to be fed back. In Ref. 5, two accelerometers, one at the tip and the other at midspan of the rotor blade, are used to measure the lag rate for damping enhancement. Because here no sensors are available in the blades, an observer-based control architecture is chosen. This allows one to feed back the (observed) lag rate and, thus, to increase damping without dedicated sensors.

To take into account deviations of the physical plant from the design model already in the design phase and to make the controller robust against changes in flight speed, H_∞ control methods are chosen for design. This approach is selected to overcome shortcomings reported in Ref. 11, where helicopter vibration reduction controllers did not perform satisfactorily (and even caused instabilities) when simulated at helicopter flight speeds other than those they were designed for. Closed-loop shaping is favored because of the flexibility in the translation of design requirements into weighting functions.

Only output feedback is possible for the structure of the controller because the state variables are not measurable. Furthermore, the output disturbance is not directly available to the controller because only the superposition of baseline vibration and IBC-induced vibration can be measured. Figure 2 shows the closed-loop structure, set up for disturbance rejection. The reference signal is set to $\boldsymbol{r} \equiv 0$ and is omitted in Fig. 2. The signal $\boldsymbol{d}$ stands for output disturbances that cannot be influenced by the controller output $\boldsymbol{u}$. The output disturbance is the periodic baseline vibration. These quantities are summed up with the vibration $\boldsymbol{y}$ caused by IBC. The sum $\boldsymbol{v}$ corresponds to the overall vibration that acts on the helicopter. Uncertainties of the plant G are modeled as input and output uncertainties. The sum of controller output $\boldsymbol{u}$ and uncertainty $\boldsymbol{d}'$ represents the IBC input of the plant in MBC.

To investigate the possibility of account taken of periodicity during design, the controller design is performed at multiple, equally spaced positions of the rotor around the azimuth with identical setups. The resulting controllers are then developed in a Fourier-series analog to the time-periodic model itself. This yields a time-scheduled controller with a period equal to one rotor revolution.

The controller can be realized as an observer with state feedback. The additional term $\gamma^{-2}B_1^T X_\infty \boldsymbol{x}_K$ can be interpreted as an estimate of the worst-case disturbance.[26] The state equation of the observer is given by

$$\dot{\boldsymbol{x}}_K = A\boldsymbol{x}_K + B_1\gamma^{-2}B_1^{-2}B_1^T X_\infty \boldsymbol{x}_K + B_2\boldsymbol{u} + Z_\infty L_\infty(C_2\boldsymbol{x}_K - \boldsymbol{y})$$

and the state feedback equation by

$$\boldsymbol{u} = F_\infty \boldsymbol{x}_K$$

A, B_1, B_2, and C_2 are matrices from the augmented system's state-space realization, whereas F_∞, L_∞, X_∞, and Z_∞ are matrices and γ is a scalar of the controller, all in standard H_∞ control notation. (See Ref. 26 for details and the underlying assumptions.) Figure 3 shows the realization of the controller as an observer with state feedback and the gain-scheduled matrices.

V. Vibration Reduction

The controller is simulated in closed-loop with the model at high flight velocity (design case) and at low flight velocity (off-design case). The baseline vibration from Camrad II acts on the system as a disturbance at the plant output. The controller cancels out vibration in the selected three hub quantities (−99%), as shown in Fig. 4. The results of the simulation for low flight velocity are similar. Although performance is slightly degraded in the off-design case, the controller still considerably reduces vibration in all three selected hub quantities (−96%). This demonstrates the robust usability of the controller at both operating points.

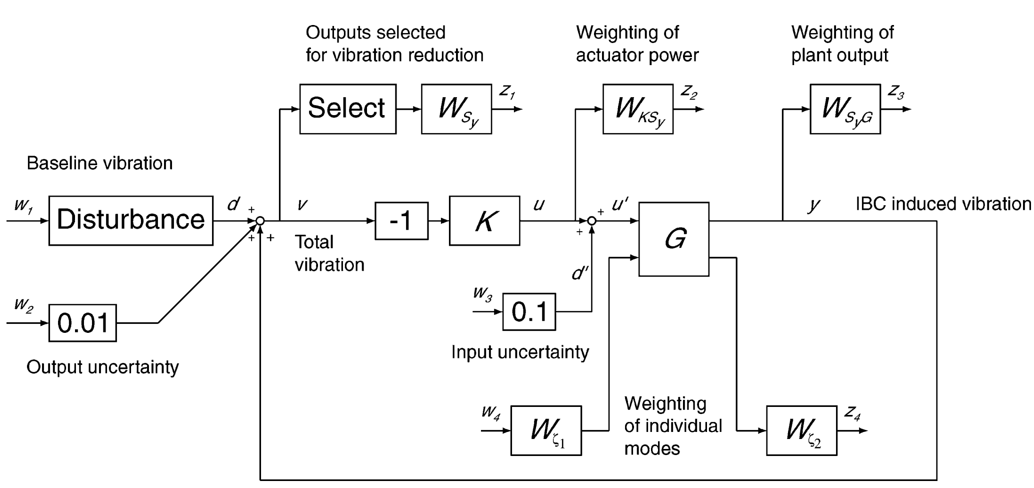

Fig. 2 Closed-loop structure used for controller design.

The final goal of helicopter vibration reduction is to reduce vibration, not necessarily at the rotor hub, but at specific points in the fuselage, for example, at the pilot seat or in the load compartment. The model, on which the controller design is based, is then required to include both rotor dynamics and fuselage dynamics. (A complete cancellation of all six forces and moments at the rotor hub would be ideal, because no vibration would then be transmitted to the fuselage. However, it is not possible to cancel out completely six outputs due to the limited number of degrees of freedom.[27])

A reduction in vibration at one specific location in the fuselage can lead to reduced or increased vibration at other locations, depending on the dynamic properties of the system. Depending on the mission definition of the helicopter, different design objectives are conceivable: For surveillance and transport helicopters, vibration reduction at the pilot/copilot position may be of prime importance, whereas for a rescue helicopter, the load compartment, which is the location of the injured person, becomes more important.

Two different cases are examined: First, vibration reduction is examined at the pilot position only. (For the copilot position, measurements of the vibration level were only available in the y direction for instrumentation reasons.) Second, simultaneous vibration reduction at both the pilot and load compartment positions is considered.

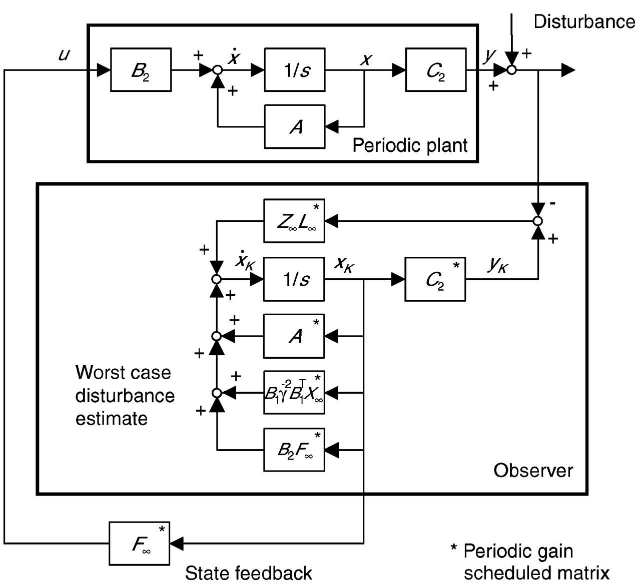

Fig. 3 Observer-based realization of the controller.

In all cases, during the controller design, both off-design locations in the fuselage and at the rotor hub have to be monitored closely to avoid violations of structural load limits.

The results of the simulation given in Fig. 5 show that it is possible to reduce vibration considerably at the pilot position in all three directions (accelerations in the x, y, and z directions). Given the only available measurement at the copilot seat position in the y direction, a simultaneous vibration reduction of -89% at the pilot and copilot position is achieved. Simultaneously, vibration at the load compartment is increased by $+32\%$.

Figure 6 shows the results of the simulation for simultaneous vibration reduction at the pilot seat and in the load compartment. Vibration is reduced in all six outputs, as well as at the copilot position. As expected, because more outputs are considered now than degrees of freedom are available, performance is degraded in comparison to the results for a single location. The average vibration reduction achieved simultaneously at the pilot and copilot seat, as well as in the load compartment, is -47%. Although some fine tuning and adjustment of the weighting functions for the individual outputs might help to distribute the vibration reduction more equally among the outputs, the key finding remains the same: If more independent outputs are considered for vibration reduction than degrees of freedom are available, vibration cannot be canceled, but only reduced moderately.

The result of increased hub loads agrees with those obtained in previous studies. The average increase in hub vibration is $+293\%$. Similar findings have been obtained in Ref. 7, where vertical hub shears increased by a factor of three to six when HHC inputs aimed at minimizing fuselage accelerations were introduced. This indicates that vibration reduction in the helicopter fuselage does not necessarily lead to reduced vibration at the rotor hub and vice versa: That is, a vibration reduction at the rotor hub does not necessarily lead to reduced vibration in the fuselage.

VI. Lag Damping Enhancement

An important issue in the control design is the sensitivity of the closed-loop system to disturbances, for example, gusts affecting one or several rotor blades. The control strategy to increase damping is to use an observer-based controller and to feed back the (observed) rates of the modes to be controlled. The physical mechanism of lag damping augmentation is as follows: The observed lag velocity is fed back to the individual blade pitch control. A blade flapping velocity is, thus, generated, which results in an in-plane moment due to the Coriolis force opposing the lag motion. Because the opposing Coriolis force is proportional to the lag velocity, blade damping is augmented.[5]

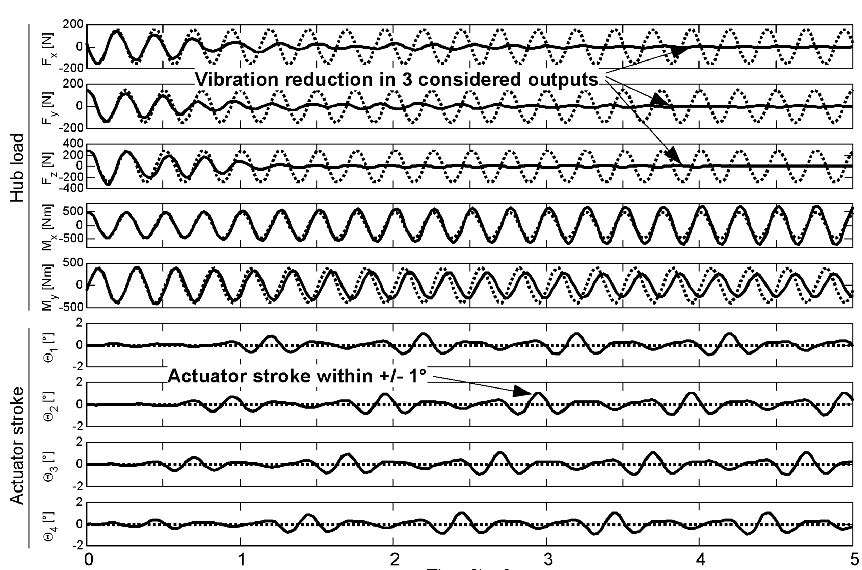

Fig. 4 Vibration reduction in hub force vector: - - - -, baseline vibration vs ——, controlled vibration.

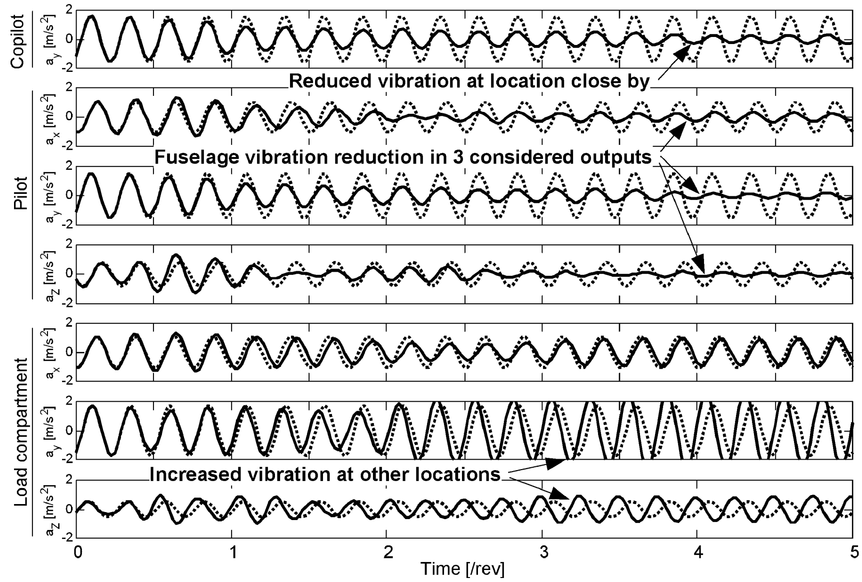

Fig. 5 Fuselage vibration reduction at pilot position: - - - -, baseline vs ——, controlled.

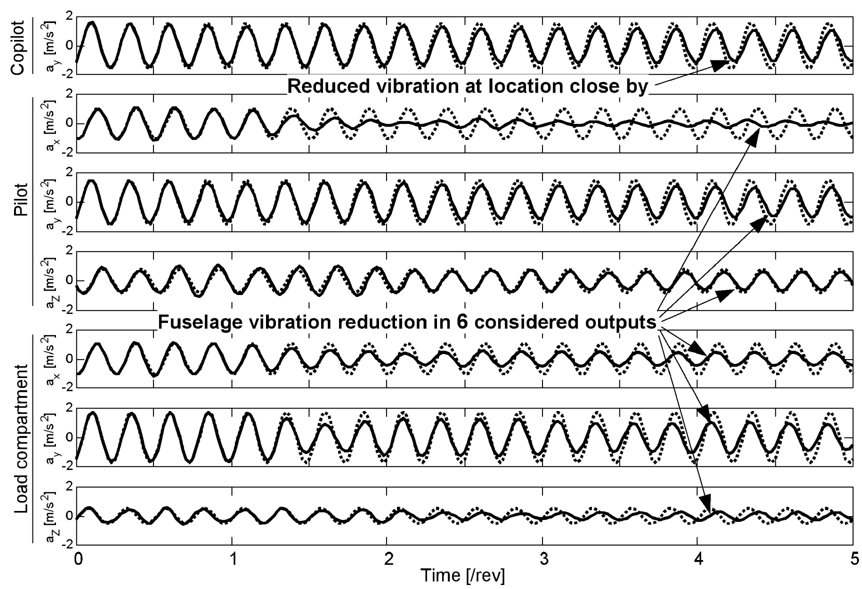

Fig. 6 Simultaneous vibration reduction pilot position and load compartment: - - - -, baseline vs ——, controlled.

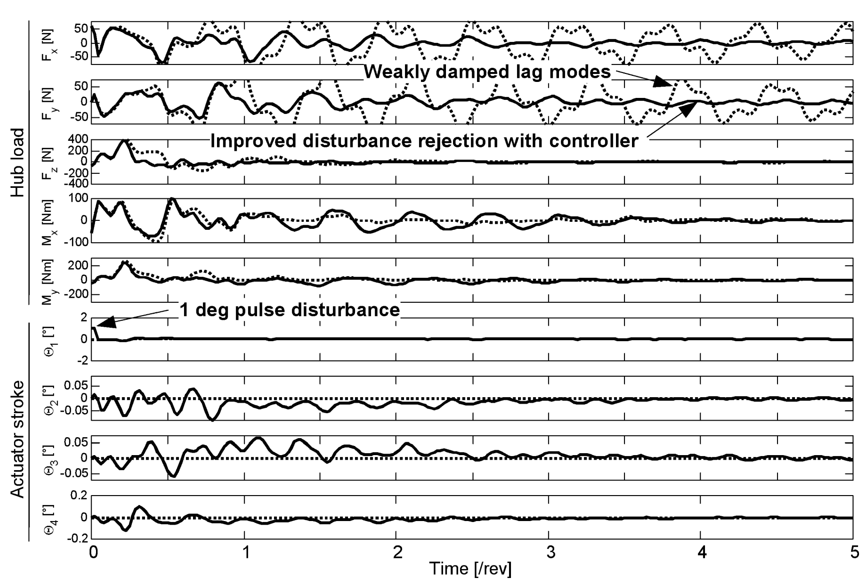

Fig. 7 Disturbance rejection with controller: - - - -, uncontrolled response vs ——, controlled response.

To simulate a gust maneuver, one rotor blade is pitched up one degree for the duration it takes the blade to advance 10 deg, that is, a pulse input to one blade is simulated. The results are compared for the uncontrolled and the controlled cases. Here, the forces in the x and y direction are most interesting because the weakly damped lag modes mostly affect F_x and F_y. Figure 7 shows the improvements achieved by the controller with respect to decay times.

By definition, the observer requires that the modes to be controlled are observable in the measured outputs. A frequency domain analysis of the plant with respect to controllability and observability revealed that the cyclic forms of the lag modes are not expected to cause any problems, whereas the collective lag form is only observable in M_z. Moreover, the differential lag mode cannot be controlled by a time-constant controller based on a time-constant approximation of the time-periodic plant because the differential mode is not observable ("reactionless" mode) in the time-constant system.[6]

To analyze the influence of the availability of M_z as a measurement, two time-constant controllers are designed for the plant. Both controllers use all three forces and the moments about the pitch and yaw axes, whereas the second controller is also able to utilize a measurement of the moment about the torque axis M_z. The section of the pole map containing the most relevant second lag mode is given in Fig. 8. N roots are shown. The locations of the collective and differential roots correspond to the natural frequency and damping of the mode, whereas the imaginary parts of the roots of the cyclic modes are shifted by $\pm n$/rev in MBC. Both controllers increase damping in the cyclic forms. With both controllers, the critical damping of the cyclic modes is increased from the minimum 0.5% open-loop damping to 2% in the closed loop. However, collective lag mode

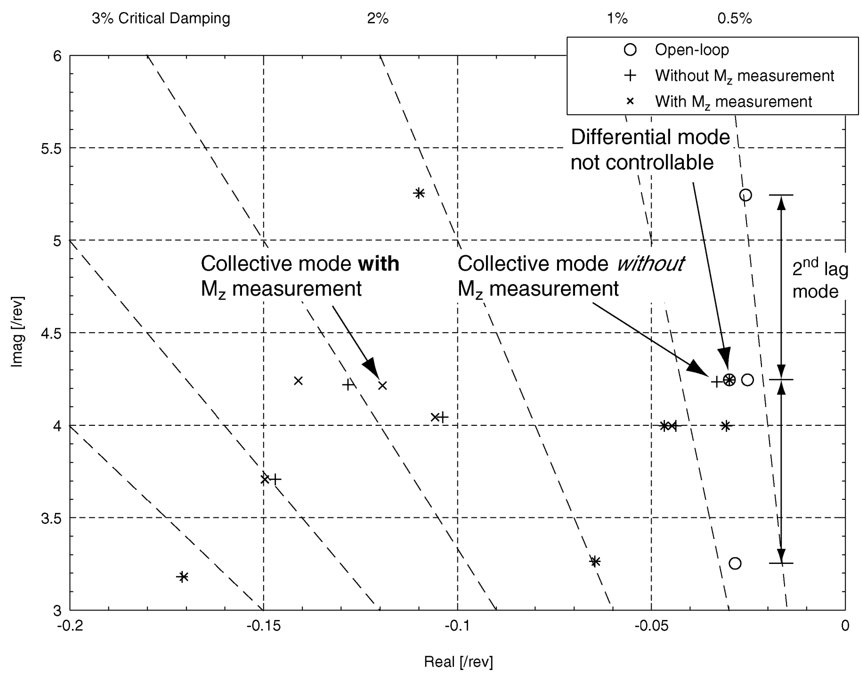

Fig. 8 Open-loop pole locations vs closed-loop pole locations, comparison of controller with and without measurement of M_z available for feedback.

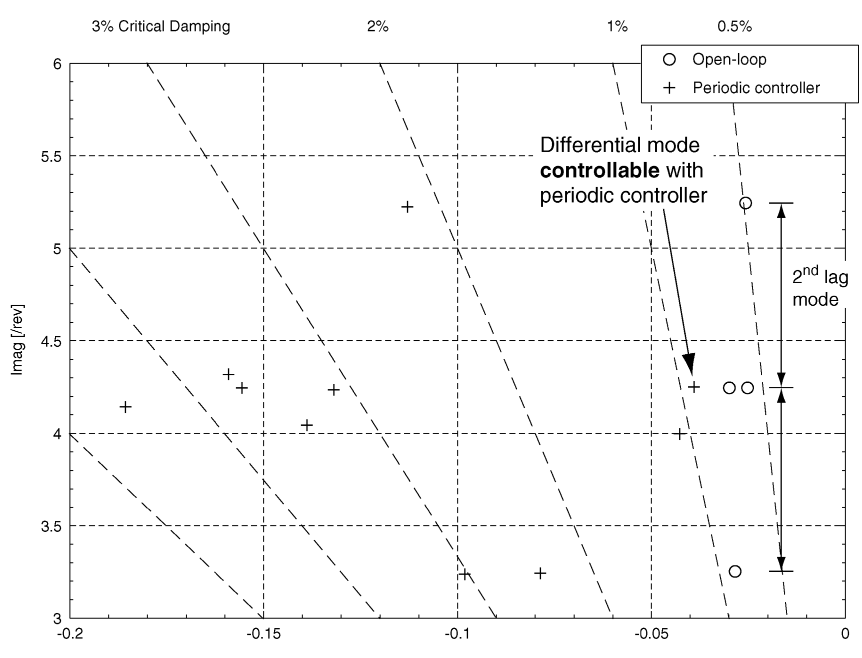

Fig. 9 Open-loop pole locations vs Poincaré exponents (closed-loop pole locations of the Floquet-transformed) periodic closed-loop system with time-periodic controller (with M_z feedback).

damping can only be increased to $> 2\%$ when a measurement of M_z is available, whereas the collective pole remains almost unchanged when M_z cannot be fed back. In both designs, the differential mode remains unchanged.

Figure 9 shows the results for a time-periodic controller. The controller is based on the preceding design parameters, including feedback of M_z, with the exception of time periodicity. The open-loop pole locations of the plant are compared with the closed-loop system Poincaré exponents (closed-loop pole locations of the Floquet-transformed time-periodic closed-loop system). Although the results of the cyclic and collective modes are only slightly improved, the key result is that the time-periodic controller allows one to control the differential mode. Differential second lag mode damping is increased to approximately 1% critical damping. The possible differential damping enhancement is not as large as for the cyclic or collective form. This is due to the dynamic properties of the plant and not to periodicity or the gain-scheduled time-periodic implementation of the controller because time-constant controller designs for fictitious time-constant rotors "fixed" at specific azimuthal positions yield similar results. Although the differential damping enhancement is smaller than that for the cyclic and collective forms, the time-periodic controller allows one to increase second lag mode damping from minimal 0.5% critical damping to 1% in the differential form and to 2–3% in the collective and cyclic form. With a controller design based on the constant coefficient approximation, it is not possible to control the differential form from the nonrotating system.

VII. Conclusions

A control law to reduce vibration and increase lag damping in helicopters via IBC has been developed. H_∞ control synthesis was used to design one robust controller based on a reduced-order model that can be used to alleviate vibration in different operating conditions at different helicopter flight speeds. The damping enhancement achieved leads to a significantly reduced gust sensitivity.

With regard to hub loads, vibration can be canceled (-99%) in three outputs simultaneously, for example, all three hub forces. By the use of the same controller as designed for high-speed flight at a different operating point with low-flight speed, oscillatory loads were also reduced by -96%, demonstrating the robustness of the controller.

As in previous studies, a reduction in hub vibration does not necessarily lead to reduced vibration in the cabin. Fuselage accelerations increased by a factor of up to three when IBC inputs aimed at minimizing hub loads were introduced. Therefore, a finite element model of the flexible fuselage was coupled with the aeromechanical rotor model. The resulting coupled rotor–fuselage model was used to calculate and control vibration at locations in the cabin, such as at the pilot and copilot seats and in the load compartment. With the focus of cabin accelerations, simultaneous vibration reduction at the pilot and copilot seats of -89% was achieved, at the expense of an increased level of vibration of $+32\%$ in the load compartment.

The use of a model-based control strategy enabled lag damping to be enhanced from 0.5 to 2–3% critical damping without the use of dedicated lag rate sensors in the blades. Only the hub loads must be measured. However, because the lag rate of the blades in the rotating frame has to be reconstructed from measurements in the nonrotating system, some restrictions apply. To increase damping in the cyclic modes is straightforward, whereas damping enhancement in the collective mode requires measurement of the torque moment. Finally, differential lag mode damping can only be increased via periodic control.

Flight tests of the proposed control law are planned on the BO 105 helicopter equipped with the individual blade root control system.

References

[1] Teves, D., Klöppel, V., and Richter, P., "Development of Active Control Technology in the Rotating System, Flight Testing and Theoretical In estigations," *Proceedings of the 18th European Rotorcraft Forum*, Association Aéronautique et Astronautique de France, Verneuil sur Seine, France, 1992, pp. 89.1–89.13.

[2] Cole, D. G., "Harmonic and Narrowband Disturbance Rejection for Linear Time-Periodic Plants," Ph.D. Dissertation, Dept. of Mechanical Engineering, Virginia Polytechnic Inst. and State Univ., Blacksburg, VA, 1998, Chap. 1.

[3] Hohenemser, K. H., and Sheng-Kuang Yin, "Some Applications of the Method of Multiblade Coordinates," *Journal of the American Helicopter Society*, Vol. 17, No. 3, 1972, pp. 3–12.

[4] Leishman, J. G., "Principles of Helicopter Aerodynamics," Cambridge Univ. Press, Cambridge, England, U.K., 2000, Chap. 4.

[5] Ham, N. D., Behal, B. L., and McKillip, R. M., Jr., "Helicopter Rotor Lag Damping Augmentation Through Individual-Blade-Control," *Vertica*, Vol. 7, No. 4, 1983, pp. 361–371.

[6] Mannchen, T., and Well, K. H., "Helicopter Vibration Reduction Using Periodic Robust Control," AIAA Paper 2001-4034, Aug. 2001.

[7] Friedmann, P., and Millott, T., "Vibration Reduction in Rotorcraft Using Active Control: A Comparison of Various Approaches," *Journal of Guidance, Control, and Dynamics*, Vol. 18, No. 4, 1995, pp. 664–673.

[8] de Terlizzi, M., and Friedmann, P. P., "Active Control of Vibration due to BVI and Experimental Correlation," *Proceedings of the 25th European Rotorcraft Forum*, Associazione Italiana di Aeronautica e Astronautica, Rome, Italy, 1999, pp. G6.1–G6.16.

[9] Bittanti, S., Lorito, F., and Strade, S., "An LQ Approch to Active Control of Vibration in Helicopters," *Journal of Dynamic Systems, Measurement and Control*, Vol. 118, Sept. 1996, pp. 482–488.

[10] Dieterich, O., "Application of Modern Control Technology for Advanced IBC Systems," *Proceedings of the 24th European Rotorcraft Forum*, Association Aéronautique et Astronautique de France, Verneuil sur Seine, France, 1998, pp. AS03.1–AS03.21.

[11] Arcara, P., Bittanti, S., and Lovera, M., "Active Control of Helicopter Rotors for Attenuation of Vibration in Forward Flight," *IEEE Transactions on Control Systems Technology*, Vol. 8, No. 6, 2000, pp. 883–894.

[12] Yeo, H., and Chopra, I., "Fundamental Issues Related to Prediction of Coupled Rotor/Airframe Vibration," *Proceedings of the 25th European Rotorcraft Forum*, Associazione Italiana di Aeronautica e Astronautica, Rome, Italy, 1999, pp. G9.1–G9.18.

[13] Cribbs, R. C., Friedmann, P. P., and Chiu, T., "Coupled Helicopter Rotor/Flexible Fuselage Aeroelastic Model for Control of Structural Response," *AIAA Journal*, Vol. 38, No. 10, 2000, pp. 1777–1788.

[14] Bertogalli, V., Bittanti, S., and Lovera, M., "Simulation and Identification of Helicopter Rotor Dynamics Using General-Purpose Multibody Code," *Journal of the Franklin Institute*, Vol. 366, No. 5, 1999, pp. 783–797.

[15] Hablowetz, T., Mannchen, T., and Well, K. H., "Advanced Helicopter Flight Simulation with Controller in the Loop," *Proceedings of the 26th European Rotorcraft Forum*, Netherlands Association of Aeronautical Engineers, Amsterdam, The Netherlands, 2000, pp. 25.1–25.9.

[16] Kretz, M., and Larché, M., "Future of Helicopter Rotor Control," *Vertica*, Vol. 4, No. 1, 1980, pp. 13–22.

[17] Quackenbush, T. R., "Testing of a Stall Flutter Suppression System for Helicopter Rotors Using Individual Blade Control," *Journal of the American Helicopter Society*, Vol. 29, No. 3, 1983, pp. 38–44.

[18] Johnson, W., *CAMRAD II Comprehensive Analytical Model of Rotorcraft Aerodynamics and Dynamics*, Johnson Aeronautics, Palo Alto, CA, 1999.

[19] Miao, W.-L., Twomey, W. J., and Wang, J. M., "Challenge of Predicting Helicopter Vibration," *Proceedings of the American Helicopter Society International Meeting on Advanced Rotorcraft Technology and Disaster Relief*, American Helicopter Society, Alexandria, VA, 1998, pp. 487–500.

[20] Johnson, W., *Helicopter Theory*, Dover, New York, 1994.

[21] Stoppel, J., "Idealisierung des Hubschraubers PAH1 nach der Methode der Finiten Elemente," Messerschmitt-Bölkow-Blohm Technical Report, Munich, Germany, 1987.

[22] Sorour, M., "Implementierung von Rotor-Zellenkopplungen für Regelkonzepte zur Vibrationsminderung bei Hubschraubern durch Einzelblattregler," M.S. Thesis, Inst. of Flight Mechanics and Control, IFR_SR_00_016, Univ. of Stuttgart, Stuttgart, Germany, 2000.

[23] Lahey, R. S., Miller, M. P., and Reymond, M. (Eds.), *MSC/NASTRAN Reference Manual*, Ver. 68, MacNeal–Schwendler, Santa Ana, CA, 1994.

[24] Tomashofski, C. A., and Tischler, M. B., "Flight Test Identification of SH-2G Dynamics in Support of Digital Flight Control System Development," *Proceedings of the 55th American Helicopter Society Annual Forum, American Helicopter Society*, Alexandria, VA, 1999, pp. 1141–1167.

[25] Müller, M., Arnold, U. T. P., and Morbitzer, D., "On the Importance and Effectiveness of 2/REV IBC for Noise, Vibration and Pitch Link Load Reduction," *Proceedings of the 25th European Rotorcraft Forum*, Associazione Italiana di Aeronautica e Astronautica, Rome, Italy, 1999, pp. G4.1–G4.12.

[26] Skogestad, S., and Postlethwaite, I., *Multivariable Feedback Control*, Wiley, Chichester, England, U.K., 1996, Chap. 9.

[27] Mannchen, T., and Well, K. H., "Influence of the Number of Rotor Blades on Helicopter Active Vibration Reduction Potential," *Proceedings of the 28th European Rotorcraft Forum*, Royal Aeronautical Society, London, U.K., 2002, pp. 81.1–81.11.