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HELICOPTER ROTOR BLADE FLUTTER

BY

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ABSTRACT

The stability problem of one isolated rotor blade in hover or in axial flow is considered. The blade has hinged or rigid connection with rotor hub and experiences coupled flap,lag and torsion deformation. Control system is modeled by some equivalent stiffness. Besides, the blade stability in forward flight is considered. The blade model permits to analyze blade oscillations with variable boundary conditions, when blade strikes its hub stop.

The experimental technique to evaluate blade oscillation attenuation in flight tests is presented.

Nomenclature

x and y -main central inertia axises of blade section,
Y - axis parallel to rotor axis,
X - axis laying in-plane of rotation and normal to blade longtitudinal axis,
W - undisturbed flow velocity, m/s
V - horizontal flight speed of helicopter, m/s
φ_0 - blade section pitch angle, rad
a_0 - blade section angle of attack, rad
ϑ_0 - blade section inflow angle, rad
b - blade chord, m
r - blade section radius, m
ω - angular speed, rad/s
$\varpi = \omega / \omega_{nom}$ -nondimentional rotor speed
ω_{nom} - nominal speed
χ - lateral and longitudinal tilt of swashplate, rad
$\Delta\beta$ - blade flap (out-of-plane) motion, rad
ξ - blade inplane motion,rad
k_φ - coefficient of friction in blade pitch hinge
p_{fl} - flutter frequency, 1/rev
ω_{fl} - rotor speed under flutter, 1/rev
I_x, I_y, T_t - blade section (second) moments of inertia about x,y and blade torsion axises respectively
N - centrifugal force in current blade section,
x_0 - distance between blade leading edge and pitch hinge axis, m
x_A - distance between blade pitch hinge axis to section aerodynamic centre, m
m - mass of blade section of unit length, kg
I_m - blade section second moment of inertia
k_β -coefficient of blade pitch-flap motion coupling (so-called "flap compensator"),
k_ξ -coefficient of blade pitch-lag motion coupling,
C_{con} - stiffness of leg from blade to swashplate connecting elements of control system
ρ - air density,
γ_0 - angle of blade turn in pitch hinge due to control system flexibility, rad
M_φ -friction moment in pitch hinge, N-m
M_b - pitch moment due to elastisity of control system, N-m
M_0^f, M_0^{in} - moments in flap and lag hinges due to elastisity of control system
a_0 - angle of blade constructive cone,rad

$\varphi = \vartheta + \kappa_\beta\beta + \kappa_\xi\xi$ -angle of blade turning in pitch hinge

$U_Y^{(j)}, U_X^{(k)}, \vartheta^{(m)}$ -blade shape (modes) of natural bending and torsion oscillations in vacuum

δ_j, l_k, γ_m -coefficients of bending and torsion deformations for j, k and m modes of blade natural oscillations

$\bar{s}$ - vector-function with δ_j, l_k, γ_m components

A,B,C - n x n rectangular matrices of mass, damping and stiffness

n - number of bending and torsion modes,accepted in analysis .

$(\dot{})$ – time derivative.

INTRODUCTION

This paper considers the problem of motion of twisted and flexible in bending and torsion isolated helicopter rotor blade in axial or skewed flow at hover and in forward flight. Blade motion is described by the system of three differential equations with partial derivatives. Two of these equations (1,2) describe blade motion in planes of flapping and rotation under blade bending in planes of minimal and maximal stiffnesses [3], and equation (3) describes torsional oscillations with blade turn due to control system flexibility [1].

Couplings between torsional oscillations and bending in both planes are provided because of inertial and aerodynamical forces [2,5]. Any combinations of hinged or rigid blade attachments to rotor hub are possible. Blade pitch control system is simulated by some equivalent stiffness and by kinematical couplings (k_β, k_ξ) between blade pitch angle and blade deflection angles in out-of-plane and in-plane of rotation.

Aerodynamic loads on blade element under blade oscillations are derived on the basis of hypothesis of stationarity with formulae of M.V.Keldish [4]. Conditions of air flow through the rotor are derived through blade element impulse theory for undisturbed flow.

The system of linear differential equations is formed on the basis of Galerkin method with decomposition of blade reaction in row of natural modes of oscillations in vacuum. In its calculation there can be used up to 5 bending modes and up to three torsional modes. Bending modes are coupled in planes of minimal and maximal stiffnesses because of blade attachment conditions at the rotor hub and non-coincidence of main axises of blade section stiffnesses with planes of flap and rotation under natural twist and nonzero blade pitch angle.

As a result there is the following system of differential equations of blade motion (similar systems of differential equations are considered in papers of other authors) :

$$\left[\left(EI_x\cos^2\varphi_0 + EI_y\sin^2\varphi_0\right)U_Y''\right]'' + \left[\tfrac{1}{2}\left(EI_y - EI_x\right)\sin(2\varphi_0)U_X''\right]'' - \left[NU_Y'\right]' +$$

$$+m\ddot{U}_Y - m\sigma_{cg}\ddot{\varphi}\cos\varphi_0 + \omega^2\left[m\sigma_{cg} r\,\varphi\cos\varphi_0\right]' - 2a_0\,\omega\,\dot{U}_X m + 2a_0\,\omega\,\dot{\varphi}\, m\sigma_{cg}\sin\varphi_0 = F_Y \qquad (1)$$

$$\left[\left(EI_y\cos^2\varphi_0 + EI_x\sin^2\varphi_0\right)U_X''\right]'' + \left[\tfrac{1}{2}\left(EI_y - EI_x\right)\sin(2\varphi_0)U_Y''\right]'' - \left[NU_X'\right]' +$$

$$+m\ddot{U}_X - \omega^2 mU_X + m\sigma_{cg}\ddot{\varphi}\sin\varphi_0 + \omega^2\left[m\sigma_{cg}\, r\,\varphi\sin\varphi_0\right]' - 2a_0\,\omega\,\dot{U}_Y m -$$

$$-2a_0\,\omega\,\dot{\varphi}\, m\sigma_{cg}\cos\varphi_0 + \omega^2 m\sigma_{cg}\varphi\sin\varphi_0 = F_X \qquad (2)$$

$$I_m\ddot{\varphi} - \left[GT_{\hat{e}\delta}\varphi'\right]' + \varphi\omega^2 I_m\cos(\varphi_0) + \omega^2 m\, r\,\sigma_{cg}U_X'\sin\varphi_0 - \omega^2 m\sigma_{cg} r\, U_Y'\cos\varphi_0 +$$

$$+\omega^2 m\sigma_{cg}U_X\sin\varphi_0 - m\sigma_{cg}\ddot{U}_X\sin\varphi_0 - m\sigma_{cg}\ddot{U}_X\sin\varphi_0 - m\sigma_{cg}\ddot{U}_Y\cos\varphi_0 -$$

$$-2a_0\omega\,\dot{U}_Y m\sigma_{cg}\sin\varphi_0 + 2a_0\omega\, U_X m\sigma_{cg}\cos\varphi_0 = M_h \qquad (3)$$

Solution of the system is found under the following boundary conditions:

$$M_0^f = \left[EI_X U_Y'''\right]_0 = \kappa_\beta (M_b + M_\varphi) \quad (4)$$

$$M_0^{in} = \left[EI_Y U_X''\right]_0 = \kappa_\xi (M_b + M_\varphi) \quad (5)$$

$$M_b = C_{con}\,\gamma_0 - M_\varphi \quad (6)$$

Right hand expressions in equations (1)-(3) for aerodynamic forses Fx, Fy and for moment M_h about the axis of blade pitch hinge are:

$$F_Y = C_y^\alpha \rho b\, W^2\{\varphi\cos\nu_0\cos\alpha_0 + \frac{\dot\varphi}{W}\left[\left(\tfrac{3}{4}b - x_0\right)\cos\nu_0 + \left(\tfrac{1}{4}b - x_0\right)\sin\alpha_0\sin\varphi_0 + \right.$$

$$\left. + \tfrac{1}{8}b\,\sin 2\alpha_0\,\sin\vartheta_0\right] - \frac{\dot U_Y}{W}\cos\vartheta_0\cos\varphi_0 - \frac{\dot U_X}{W}(\cos\vartheta_0\sin\varphi_0 + \sin\alpha_0)\} \quad (7)$$

$$F_X = C_y^\alpha \rho b\, W^2\{\varphi\sin\nu_0\cos\alpha_0 + \frac{\dot\varphi}{W}\left[\left(\tfrac{3}{4}b - x_0\right)\sin\nu_0 - \left(\tfrac{1}{4}b - x_0\right)\sin\alpha_0\cos\varphi_0 + \right.$$

$$\left. + \tfrac{1}{8}b\,\sin 2\alpha_0\,\cos\vartheta_0\right] - \frac{\dot U_Y}{W}(\sin\vartheta_0\cos\varphi_0 + \sin\alpha_0) - \frac{\dot U_X}{W}\sin\vartheta_0\sin\varphi_0\} \quad (8)$$

$$M_h = -\frac{\pi}{4}\rho b^2 W\left(\tfrac{1}{2}b - x_0\right)\dot\varphi + C_y^\alpha \rho b W^2\{\varphi\left[\left(\sigma_A + \frac{b}{4}\right)\cos 2\beta\alpha_0 - \frac{b}{4}\cos\alpha_0\right] +$$

$$+ \frac{\dot\varphi}{W}\cos\alpha_0\left[\tfrac{1}{4}b(2+\cos\alpha_0) - x_0\right]\left(\frac{b}{4} + \sigma_A\right) - \frac{\dot U_X}{W}\left[\left(\frac{b}{4} + \sigma_A\right)\sin(\varphi_0 + \alpha_0) - \right.$$

$$\left. - \frac{b}{4}(\sin\varphi_0 + \sin\alpha_0\cos\vartheta_0)\right] - \frac{\dot U_Y}{W}\left[\left(\frac{b}{4} + \sigma_A\right)\cos(\varphi_0 + \alpha_0) - \frac{b}{4}(\cos\varphi_0 - \sin\alpha_0\sin\vartheta_0)\right]\} \quad (9)$$

Solution of the system of equations is carried out on the basis of Galerkin method by decomposition of the solution in row of modes as:

$$U_Y = \sum_j \delta_j U_Y^{(j)} \quad (10)$$

$$U_X = \sum_k l_k U_X^{(k)} \quad (11)$$

$$\vartheta = \sum_m \gamma_m \vartheta^{(m)} \quad (12)$$

As a result of such substitution the system of ordinary differential equations of second order with unknown δ_j, l_k, γ_m is recived. This system can be expressed in matrix form as

$$A\,\ddot{\vec s} + B\,\dot{\vec s} + C\,\vec s = 0 \quad (13)$$

Analysis of eigenvalue problem for the system (13) gives the possibility to estimate blade stability.

ISOLATED BLADE STABILITY

The basic lows of isolated blade oscillation stability were eximined on the example of typical fiberglass blade.

Fig.1 and Fig.2 show the roots of characteristical equation of the system (13) as a functions of nondimential rotor rotational speed. One can see imaginary part"p" and real part"q" of eigenvalues for three blade section centre of gravity positions (x_{cg}=24.8%,25.7% and 26.5% of chord). There is also comparison with partial frequencies,which represent traditional "resonance" diagram as a function of rotor speed on figure 3.

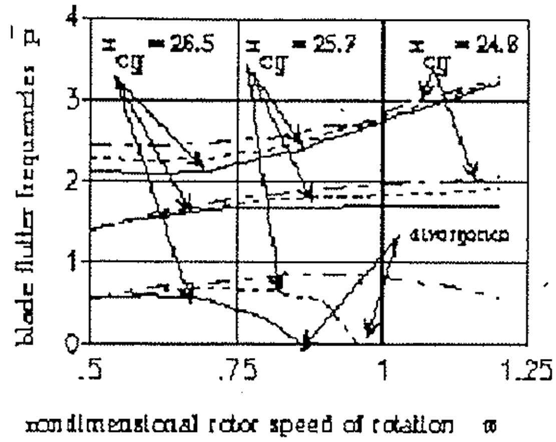

Fig 1. Blade centre of gravity effects on flutter frequencies

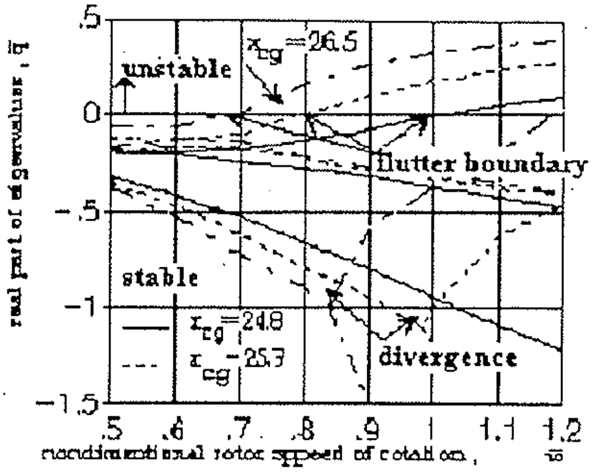

Fig 2. Influence of blade centre of gravity position x_{cg} on flutter damping

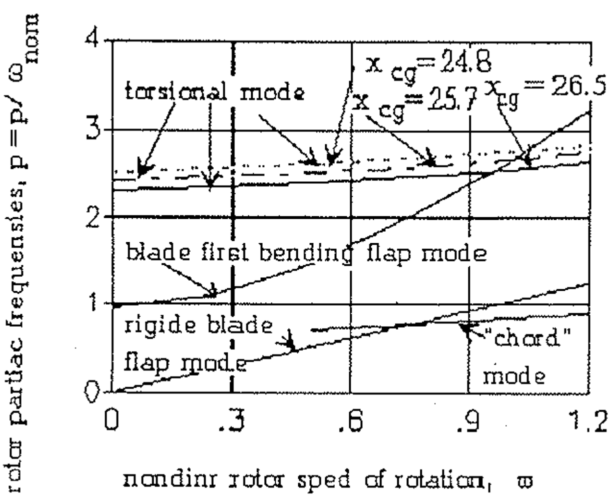

Fig 3. Rotor resonanse diagram

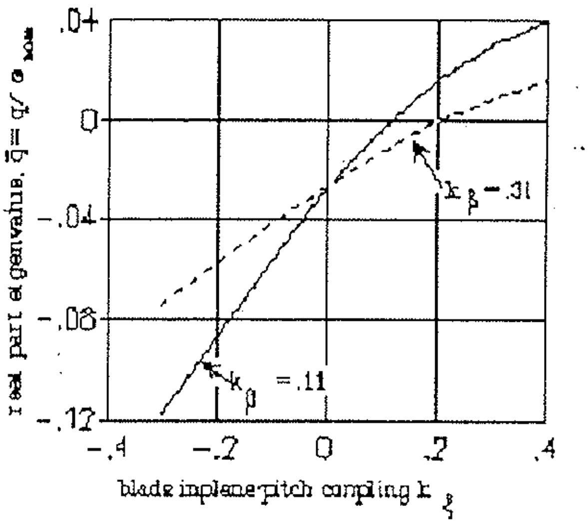

Fig 4. Effect of coupling k_β, k_ξ on damping

On Fig.1 one can notice the appearance of divergent instability region, when $\bar{p} = 0$ and $\bar{q} > 0$. Such type of instability for blades with typical parameters and torsional frequency arised from oscillation mode, which represents blade flap motion..

Second type of instability, named "classic flutter", arised under interaction of first bending mode and torsional mode of blade oscillation,Fig.1 and Fig.2.

Third type of blade instability was named "chord" flutter, because of the dominant blade oscillation form is in plane of rotation.

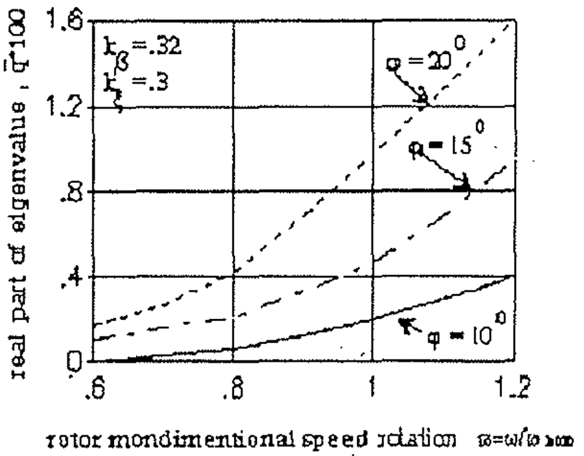

Fig 5. Unfluence of blade piten on blade damping

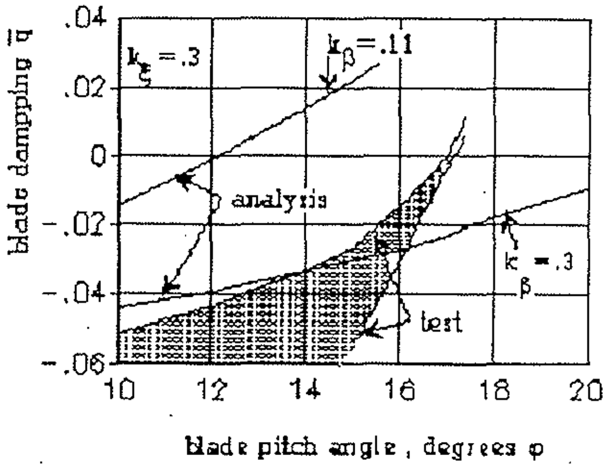

Fig 6. Effect of k_β conpling on blade damping

Figures 4-6 show the influence of kinematical coupling parameters k_β and k_ξ ,nondimentional rotor speed ϖ and blade pitch φ_0 on decrements or increments values $\bar{q}$ of blade inplane oscillations.

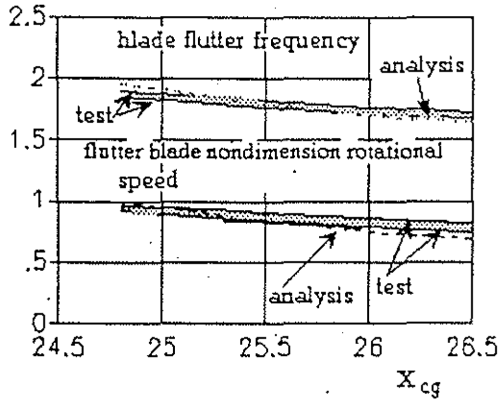

Fig 7. Rotor blade frequencies from analysis and test

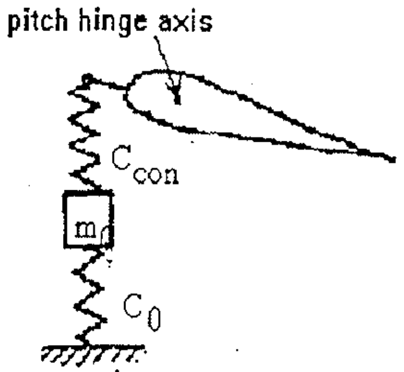

Fig 8. Oscillation loop model

Fig.7 compares analytical and experimental frequencies of blade bending-torsion flutter and influence of "efficient" blade centre of gravity position along its chord, x_{cg},% b.

COUPLED OSCILLATIONS OF BLADE AND "REVERSABLE" CONTROL SYSTEM

The method described above and the way of solution of isolated blade oscillation problem are applicable to air screws (rotors) with actuator or irreversable control system. When the control system is reversable, forces and moments from rotor blades can pass to pilot's stick after overcoming of friction forces or resistance of control system mechanisms,and then the control system will be switched in oscillation loop. Oscillation loop model can be presented in the form, shown on Fig.8.

In this case the equation of blade motion in oscillation loop together with control system will be the following:

$$m_0\ddot{x}_1 + C_0 x_1 - C_{con}(x_1 - x_2) = 0 \qquad (12)$$

where m_0 and Co are the efficient mass and stiffness of control system .

Equations (1-3) of blade oscillations and eq.(11) with some additional terms included are the system of equations, describing in this case the systems with reversable control.

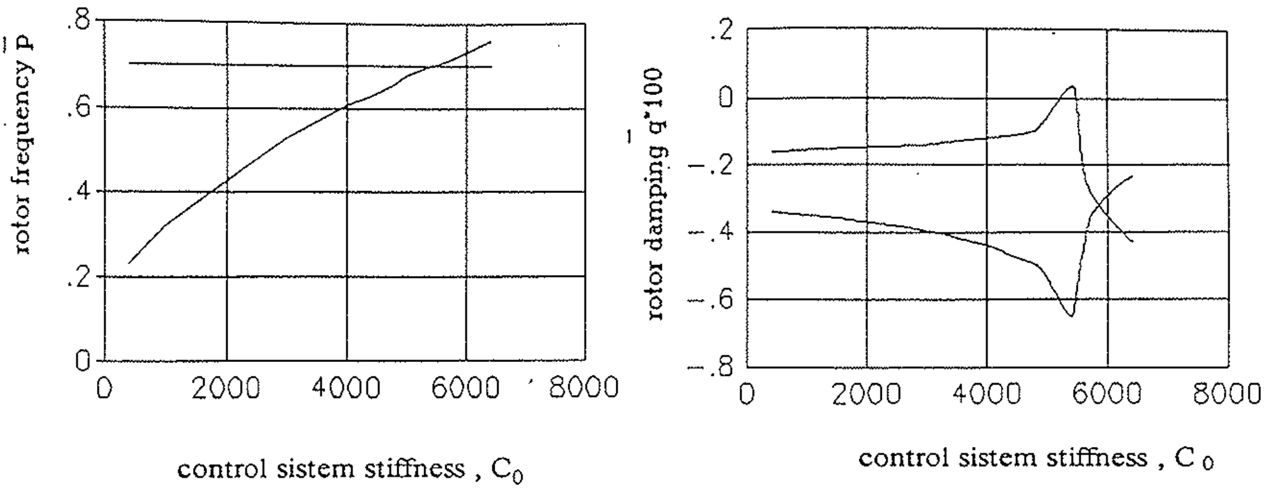

Fig 9. Blade and reversable control system coupled oscillations

Fig.9 shows eigenvalues (frequencies and demping) as a function of "equivalent" stiffness of reversable control system. As one can see in this case natural frequency can become close or equal to one of the blade oscillation frequencies or to combinational frequency $(p \pm \omega)$. This leads to essential change of oscillation decrements and possible appearence of instability area. Oscillations increment values to considerable degree depend on equivalent mass and damping of control system.

NONLINEAR OSCILLATIONS

In some cases of blade stability problem solution it is topical to consider nonlinear effects. One of these events is when rotor blade touches hub stop during its movement about of axis of flap or lag hinges or after blade deflection due to deformation of elastic element of blade damper. When blade touchs its stop on rotor hub, changing of boundary condition of blade attachment to the hub occurs. Thus the solution of blade equation in the form of modal decomposition (Galerkin method) covers both hinged and rigid types of blade natural forms. Transition from hinged to rigid conditions of blade attachement to rotor hub is fulfilled during the process of numerical integration, or by changing of kinematical conditions in hinges.

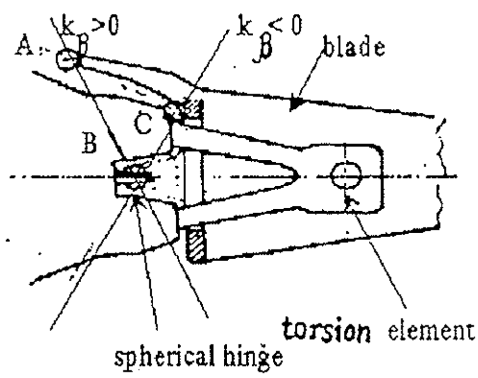

Fig 10. Nonlinear effect of blade touch with hub rest

Another important circumstance is the change of sign and value of flap compensator k_β , as shown on Fig.10.

Fig.10 shows, that in usual flight condition there is a blade combined flap-pitch motion around "basic" axis A-B, going through centre (A) of blade and control connection hinge and through spherical hinge (B) between blade and hub, and in this case kinematical coupling $k_\beta > 0$. In extreme case, when blade moves inplane and touchs the hub stop at point (C), new axis C-B of blade rotation appears, characterized by negative value of the flap compensator, $k_\beta < 0$.

This new axis of blade flap-pitch motion exists until the moment when blade leaves the stop in reversal inplane movement.

These two circumstances,that is the changes of boundary condition and of kinematic coupling k_β, were used in solution of the problem of blade nonlinear oscillation.

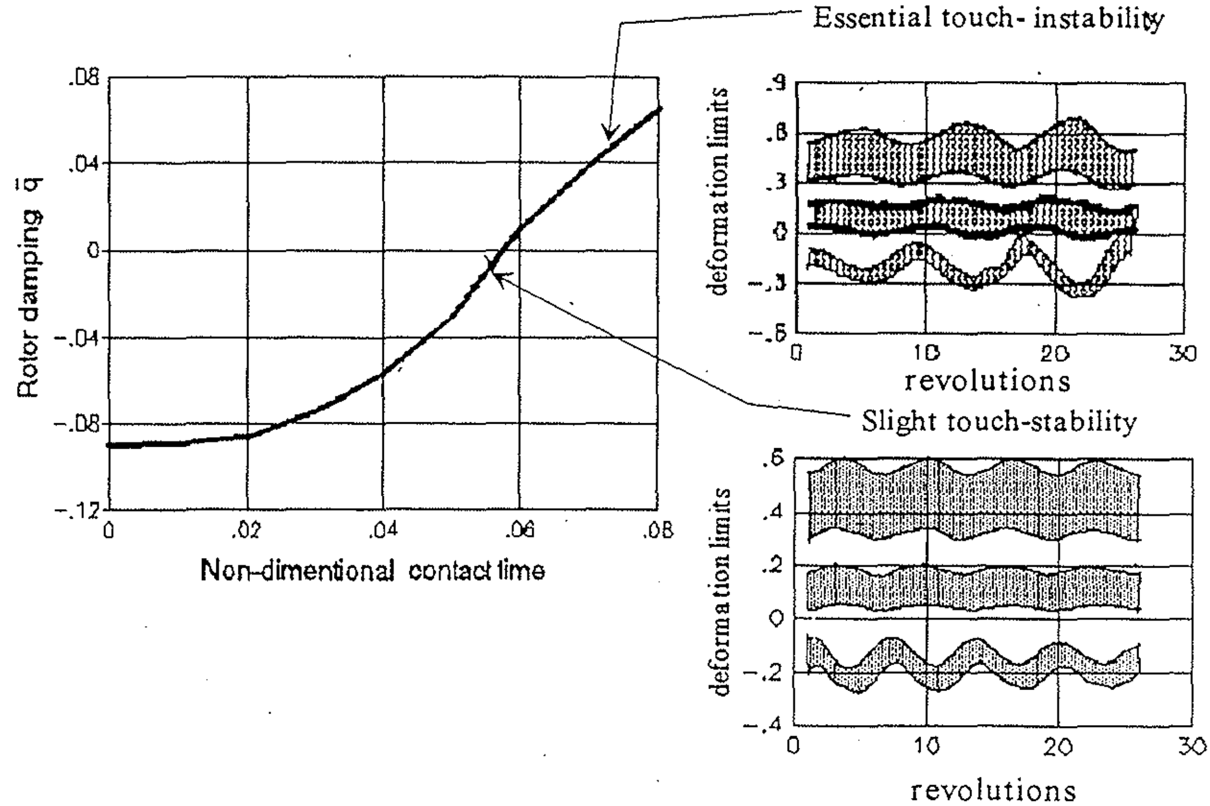

Fig 11. Blade damping as a function of hub rest contact time

Fig 11 presents blade damping as a function of blade-hub stop contact time and shows blade oscillation processes, corresponding to concrete damping. Such (divergent) regime can arise when blade reaches its hub stop, under increasing of exitation force impulse or changing of initial conditions of its movement.

Reduction of oscillation decrements and transition to blade instability condition after blade touchs its stop takes place because of that during a part of the period the system has negative value of flap compensator k_β ,(Fig.10). Analysis of linear system with such value of k_β shows the system unstable condition.

METHOD AND RESULTS OF FLIGHT EXPERIMENTS

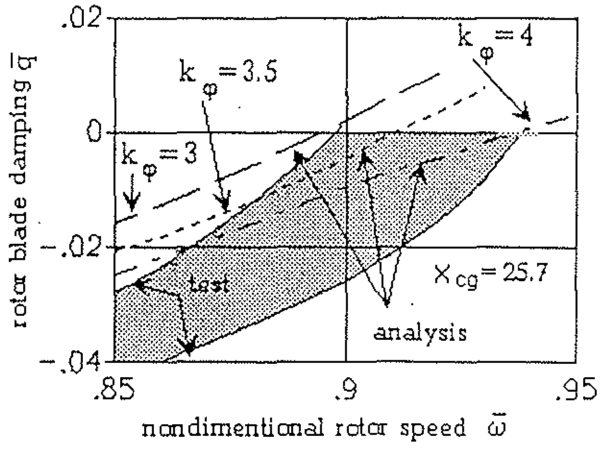

Fig 12. Comparison of test and analysis.

Analysis and estimation of experimental data represents independent and important part of investigations of aeroelastic effects. There are presented the results of such experimental inves tigations, performed for Mi-28 helicopter rotor blade to assess "chord" flatter mode of oscillations. These full-scale ground and flight investigations were performed with an exitement of investigated blade oscillation mode. The important part of the work was the selection of frequency and level of blade oscillations.

To provide full-scale ground and flight investigations of blade oscillation decrement the electric signal generator was used, which was included in auto-pilot channel of lateral control and through hydraulic actuator produced blade pitch changes with desired amplitude and frequency. At that time periodical signals from helicopter control system were switch-off and auto-pilot did not function in its normal manner.

The experiment had been performed in the following manner: at desired flight regime the generator was switched on and produced blade oscillation generation during about 5 second time period. After gererator was switched off, selected parameteres of transitional process were recorded. Analysis of these parameters permitted to estimate blade oscillation decrements.

Figure 12 shows comparison of decrements for prototype of the Mi-28 helicopter rotor blade, obtained from analysis and from bench and full-scale helicopter experiments. This comparison shows some qualitative correlation of the results and equal with analytical ones values of experimental influence of structural damping, pitch angle and flap compensator (flap-pitch coupling). Correlation of the results shown permits to get useful conclusions in prognosis and estimation of new designes.

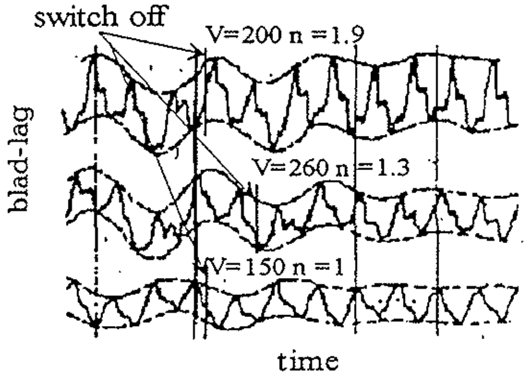

Fig 13. Transitional signals of blade inplane motion after generator switcht off

Fig.13 presents the examples of time histories of transitional processes after generator was switched off. Recorded parameter presents blade inplane motion. Dotted lines show envelope lines, corresponding to the process of oscillations exitation and fading after generator shutoff. In this example an estimation can be easily made without using of signal processing automatic means, but at other combinations of frequencies, excited oscillation levels and spectrumes of blade external loads the care is required in selection of useful signal and estimation of attenuation level.

CONCLUSIONS

1. Kinematical coupling k_ξ of blade inplane and pitch motions has significant influence on stability of "chord" flutter type of blade oscillations. For the main rotor blades, having partial frequencies near 1/rev out-of-plane and near 0,3/rev inplane,positive value $k_\xi > 0$ and absolute value $k_\xi \geq$ 0.1-0,2 can lead to instability arising. Decreasing of flap compensator k_ξ can lead to increased level of blade stability.

2. Reversability of control system creates additional loop of oscillation and can significantly influence on the blade flap-lag-torsion oscillations stability. Including of the pilot in this oscillation loop and small damping level can become the reason of arising of unstable oscillation in such system.

3. Nonlinear blade flap-lag-torsion oscillations in the case of blade touching hub stop in inplane motion have a tendency to become unstable when the time of contact of blade and hub stop increases or when the value of initial cycle amplitude increases. Main destabilizing factor is the change of sign of flap compensator k_β in that part of oscillation period,when the blade touches with the back hub stop.

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