EIGHTEENTH EUROPEAN ROTORCRAFT FORUM

E - 03

Paper No 30

INVESTIGATION OF INDIVIDUAL BLADE PITCH CONTROL IN TIME DOMAIN

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September 15-18, 1992
AVIGNON, FRANCE

ASSOCIATION AERONAUTIQUE ET ASTRONAUTIQUE DE FRANCE

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Abstract

The algorithm was developed for active control of individual rotor blade in time domain. The general blade computer code was adapted as the plant model.
Full identificability of system is assumed. Quadratic with respect to state and control variables performance index was chosen. Differential Riccatti equations were derived using matrices of blade equations of motion linearized for chosen values of azimuth angle as well as steady state and control variables. Optimal controls were obtained from solution of these equations for the conditions given at the end of the chosen time period. Controls, updated after each assumed azimuth period, were applied to nonlinear case.
The results of numerical calculations show that algorithm converges and works efficiently.

NOTATION

$B(q)$ - inertia matrix
$C(q)$ - Coriolis loads matrix
$D_n(q)$ - gyroscopic loads matrix
$f_m(q)$ - inertia loads due to centrifugal forces
J - performance index,
n - number of periods of integration
N_h - higher harmonic number
Q - weighting matrix of state variables
Q_A - aerodynamic loads
Q_d - damping loads
Q_{Nj} - nonconservative forces in Hamilton's principle
q - vector of generalized coordinates
R - weighting matrix of controls
S - matrix of Riccatti equation
T_o - equation of motion period
T - blade kinetic energy
u - control variables
U - blade potential energy
W - work of nonconservative forces
x - coordinate along blade
x - system state variables
$\eta(x)$ - blade natural modes
$\theta_g(x)$ - blade pretwist angle
$\theta_h(\psi)$ - blade additional pitch control angle: $\theta_c\cos(N_h\psi)+\theta_s\sin(N_h\psi)$
$\theta_n(\psi)$ - blade nominal pitch control angle: $\theta_n=\theta_0+\theta_1\cos(\psi)+\theta_2\sin(\psi)$
μ - advance ratio
ψ - azimuth angle
$(')$ - derivation with respect to blade lenght
$(^\bullet)$ - derivation with respect to azimuth

Matrices are writen bold.

1. INTRODUCTION

There are inherent sources of helicopter rotor blades excitation: flow velocity changes in forward flight, pitch control or atmospheric turbulence. These, and also different instabilities of blades, can cause excessive vibrations of rotor system or/-and helicopter structure.

Concerning the broad spectrum of excitation frequencies it is a challenging task to develop a system which would cancel all

detrimental effects mentioned above.
The first step for vibration reduction is the proper aeroelastic structure tailoring but it could not be a remedy for all adverse phenomena. Passive vibration suppression devices [1], [2], like isolators, absorbers or anti-resonant devices have performance limited to narrow frequency band, so they can be tuned only to particular flight conditions.
Since 1975, when [3] was published, the active control of rotating aeroelastic structures is being investigated extensively. New achievements in microprocessors and mechanical technology allowed to adopt this control technique to rotorcraft. Expectations connected with active control concern: performance improvement [4], vibration suppression [5], stability augmentation [6] and noise reduction [7,8]. Usually active control is aimed to deal with only one of the detrimental effects mentioned above, although multifunctional systems [9] are also suggested.
The idea is to use the existing controls to obtain required improvements by implementing additional control of blade pitch. Because periodic excitations are to be suppressed, a periodic control should be applied. This leads to adding pitch harmonics of frequencies other than rotor shaft angular velocity. Blade pitch angle is changed according to the formula:

$$\theta = \theta_n + \theta_h = \theta_0 + \theta_1\cos(\psi) + \theta_2\sin(\psi) + \theta_c\cos(N_h\psi) + \theta_s\sin(N_h\psi)\,.$$

This method can be applied in two ways as:
- Higher Harmonic Control (HHC), when for all blades the same additional control is applied by excitation of swash-plate,
- Individual Blade Control (IBC) when additional control is different for each blade [10].

Individual Blade Control seems to be more difficult for application but offers greater flexibility in comparison to Higher Harmonic Control.

The latest review of active control application to helicopter rotors is given in [11].

The aim of this study was to develop algorithm for stabilization of a blade motion by active control of pitch angle. The algorithm performs control of individual rotor blade in time domain.

2. DEVELOPMENT OF ACTIVE CONTROL SYSTEM

The actively controlled system, usually designed as an optimal one, (Fig.1) consists of three main parts:
- plant, i.e. aeroelastic system to be controlled,
- observer, for determination the system states,
- control unit for computation of optimal controls.

In this study, the plant consists of blade computer model (described in the sequent) for which control unit is developed. Full identifiability of the system is assumed. Details of the observer properties are not considered, although some evaluations were done.

There are three steps in active control system development:

1. Investigation of uncontrolled system behavior,
2. Establishment of active control law and algorithm,
3. Validation of control efficiency.

The first step helps the designer in understanding the physical properties of the system. In the second step, control system is developed, based on gained experience. In the third step a validation of control system is done by computer simulation and experiments. First two steps are considered in this study.

Different control techniques and kinds of controllers are currently applied in rotorcraft. These are global or local control algorithms in frequency domain using close or open loop concepts [12], designed mainly on linear model basis. They are tested and tuned using computer simulation, wind tunnel experiments or in flight measurements.

In this study active optimal algorithm in time domain is applied to nonlinear case. The properties of algorithm were evaluated by computer simulation.

3. PLANT MODEL

The general blade model developed in [13] was adapted as the plant model for this study. The overview of this model will be given here for completeness.

The isolated blade of helicopter rotor in steady flight is considered (Fig.2). Air flow velocity can vary in time, which allows to include into analysis gusts and wind. The angular velocity of rotor shaft is constant.

In the most general case a blade hub can be composed of up to three hinges in arbitrary sequence connected by stiff elements. At the ends of the first three elements flap, lag or pitch hinge can be placed. At each hinge nonlinear damping and stiffness can be taken into account. These can be arbitrary functions of hinge rotation angles and velocities.

Angle of rotation in the hinge consists of:

- constant component, which corresponds to the design angles like: precone, droop, etc.,
- periodic component, which describes the steady blade motion, (in feathering hinge pitch control is included),
- unknown component which is the disturbed blade motion.

Pitch-flap coupling can also be taken into account.

The blade is attached to the last segment of the hub or directly to the shaft. The blade can be deformable. The stiffness loads calculations are based on beam model [14], which is derived for small deformations. Blade cross sections have symmetry of elastic properties about the chord and there is no section warping. The blade has straight elastic axis parallel to the axis of the last hub segment. The blade is pretwisted around the elastic axis or, if it is rigid, around the feathering axis. The viscous structural damping of blade deformations can be included.

The blade deflections are discretized by free vibration modes.

The vector of blade motion generalized coordinates is composed of elastic degrees of freedom obtained from discretization of blade deformations by natural modes, and rigid degrees of freedom which are rotations in hinges.

The aerodynamic loads are calculated using a two-dimensional model with steady nonlinear airfoil coefficients. The unsteady effects were included by applying the dynamic inflow model, with coefficients taken from [15]. The vector of blade loads in equations of motion was obtained by successive transformations of section loads starting from aerodynamic center.
For the set of generalized coordinates, the equations of motion derived from Hamilton's principle:

$$\int_{\psi_1}^{\psi_2} [\delta (U-T) - \delta W]\, d\psi = 0$$

can be divided into two groups:
- for elastic degrees of freedom:

$$\int_R [\frac{d}{d\psi} (\frac{\partial T}{\partial \dot{q}_j}) - \frac{\partial T}{\partial q} + \frac{\partial U}{\partial q_j}] \eta_j (x)\, dR + \int_R [\frac{d}{d\psi} (\frac{\partial T}{\partial \dot{q}_j}) - \frac{\partial T}{\partial q_j} + \frac{\partial U}{\partial q_j}] \eta_j' (x)\, dR + \int_R \frac{\partial U}{\partial q_j} \eta_j'' (x)\, dR = \int_R Q_{Nj} \eta_j (x)\, dR$$

- for rigid degrees of freedom:

$$\frac{d}{d\psi} [\frac{\partial T}{\partial \dot{q}_j}] - \frac{\partial T}{\partial q_j} + \frac{\partial U}{\partial q_j} = Q_{Nj}.$$

Inserting the expressions derived for blade aerodynamic, damping, inertia and stiffness loads, equations of motion were obtained in the form:

$$B(q)\ddot{q} = -2C(q)\dot{q} - q^T D_n(q) q - f_m(q) - Q_d(q, \dot{q}) + Q_A(t, q, \dot{q})$$

Algebraic manipulations for deriving the coefficients in equations of motion are performed within the computer program. Translation vectors and rotation matrices are rearranged according to chosen hub model. To avoid numerical difficulties, the derivatives of matrices and vectors were first calculated analytically and then placed into computer code. The blade generalized masses and stiffnesses are obtained from the separate computer program before solving (or analyzing) the equations of motion, so the inertial and structural loads need not to be integrated along the blade span during the computation of equation right hand sides.

The Gear's algorithm was used for numerical integration.

4. ACTIVE CONTROL SYSTEMS.

Optimal control algorithms have the firm theoretical background only for linear plant models. The algorithm developed in this study is based on methodology used for linear optimal quadratic controllers with Riccatti equation solved in time domain. We applied this method to nonlinear model of plant, which is somehow heuristic approach.

State vector is defined as:

$$x(\psi) = (\dot{q}, q),$$

and control variables as:

$$u = u(\theta_c, \theta_s).$$

Before applying this control algorithm, stability of the open-loop system should be checked first and then the steady blade motion $\mathbf{x}_0(\psi)$ for nominal control $\mathbf{u}_0(\psi)$ should be calculated.

Nonlinear differential equations

$$\dot{x} = f(\psi, x, u)$$

with periodic right hand sides derived for general blade model given in previous section were taken as the plant in this study.

Total aerodynamic loads at the blade root

$$y = g(\psi, x, u)$$

were treated as one of the system output quantities.
A quadratic performance index for finite azimuth interval nT_0 was chosen:

$$J = \frac{1}{2} \int_{\psi_o}^{\psi_o + nT_o} [x^T Q x + u^T R u] \, d\psi .$$

The computation of optimal controls could be performed for linearized equations:

$$\dot{x} = A(\psi) x + B(\psi) u.$$

Matrices **A** and **B** obtained for steady blade motion $\mathbf{x}_0(\psi)$ and nominal control $u_0(\psi)$ are functions of azimuth angle. To obtain optimal controls, Riccatti equation:

$$\dot{S} = -SA + SBR^{-1}B^T S - A^T S - Q$$

should be solved for values given at the end of the chosen period of time:

$$S(nT_o + \psi_o) = Eps.$$

Optimal controls are calculated as:

$$u = -R^{-1} B^T(\psi_o) S(\psi_o) x(\psi_o + nT_o).$$

The computation time needed for solving Riccatti equation could make this algorithm difficult for application. So the algorithm was modified. Its flowchart is given in Fig.3.
The computation steps are as follows:
1. Matrices **A** and **B** of linearized equations are obtained for assumed azimuth angle ψ_0, steady states $\mathbf{x}_0(\psi)$, and nominal control $\mathbf{u}_0$.
2. Matrix Riccatti equation based on matrices calculated in step 1 is solved for assumed number of rotations of rotor shaft and assumed "final conditions" Eps.
3. Blade motion $\mathbf{x}(\psi)$ (i.e. solution of nonlinear plant equations) are computed for given initial values of $\mathbf{x}_0$, u_0 and for assumed number n of rotor rotations (periods of equations of motion).
4. New control values are obtained using the state values $\mathbf{x}(nT_0+\psi_0)$ calculated in step 3.
5. The computation is stopped after assumed number of rotations.

The shortcuts in this method

are:
1. Application of linear methodology to nonlinear case,
2. Calculation of linearized equation matrices for one set values of state variables and controls at one azimuth angle.
3. Computation of Riccatti equation solution only once.

5. EVALUATION OF CONTROL ALGORITHM

Efficiency of this control algorithm depends on:
1. Parameters of the plant model linearization, i.e.
   - azimuth ψ_0,
   - initial values $\mathbf{x}_0$, u_0,
2. Number n of rotations for which the equations are linearized,
3. Weight matrices Q, R,
4. Flight conditions i.e. flight velocity, nominal pitch angles, etc.

The results of computer evaluation of influence of some of these parameters is given here.

To save computer time the most calculations for investigating algorithm properties was performed for rigid flap-pitch blade model.
No nominal cyclic pitch was applied.
The equations of motion were integrated for 20 rotations.

To investigate the properties of control algorithm, the control of first harmonic for blade flapping improvement was assumed. Zero *HHC* controls were taken as starting values of iteration.

First the control was applied at 0 azimuth angle.

In Fig.4 the influence of control on rotor motion for different advance ratios is shown. As could be expected the first harmonic control has almost no influence on blade pitch (theta). For flapping (beta), in all cases during the first five or six rotations some overcontrol appears. In hover control causes periodic disturbance over almost steady uncontrolled system response. For all higher advance ratios flapping angles are diminished.

The Θ_c (theta-c) and Θ_s (theta-s) components of control variables as functions of number of rotor rotations for all cases from Fig.4 are plotted in Fig.5.
Both control components converge to almost the same values despite of advance ratio.

The results of algorithm convergence investigation are shown in Fig.6 for advance ratio μ=0.25. There were four sets of starting values:

Θ_c	Θ_s
0.0	0.0
0.0466	0.0
0.0	- 0.0329
0.0466	- 0.0329.

Despite of starting values, the control variables converge to steady values after about 17 rotations.

The influence on flapping of azimuth at which control was applied is shown in Fig.7 for μ=0.25. The equations of motion were integrated for starting azimuth range from 0 to 360 deg in 45 deg intervals. The best flapping suppression has been obtained for 315 deg.

These best control results are compared with the uncontrolled case in Fig.8.

The values of control parameters for different control azimuth angles are shown in Fig.9.

The difference between azimuth angles for which matrices **A**, **B**, and **S** are computed and at which, integration of equations of motion is started can be treated as the time which can be used by "observer" and optimizer to work out optimal controls.
The best results were obtained for azimuth 315 deg and starting integration at azimuth 340 deg.
It gives the time for control computation about 25 deg of azimuth angle if the control was applied in real time.

CONCLUSIONS

The algorithm was developed for active control of rotor blade motion in time domain. For computing control values, Riccatti equations are integrated only once for chosen values of azimuth, state and control variables. The efficiency of algorithm depends on these parameters. No divergence in algorithm computations was observed.

Acknowledgments
This research was supported by Polish State Committee for Scientific Research, grant no 449/91.

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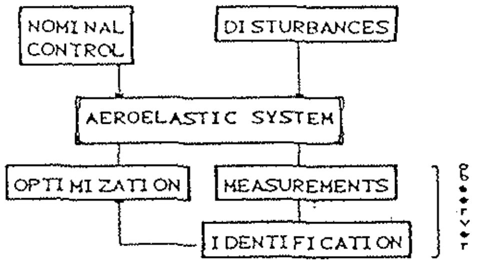


Fig. 1

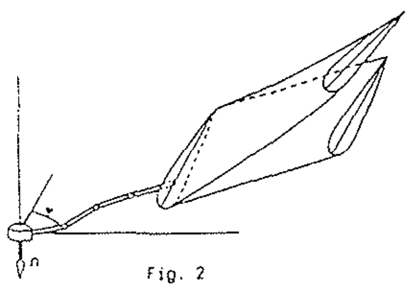


Fig. 2

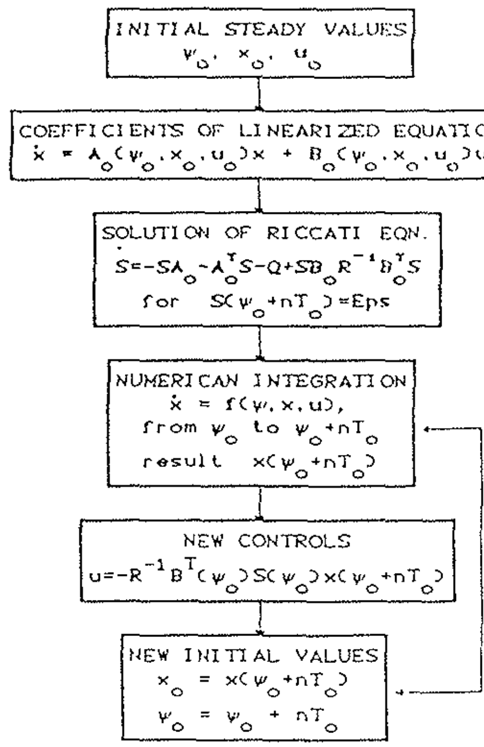


Fig. 3

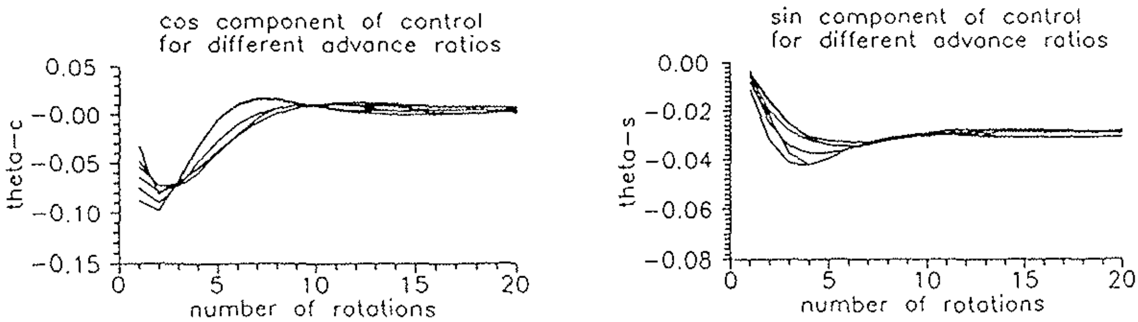


Fig. 5

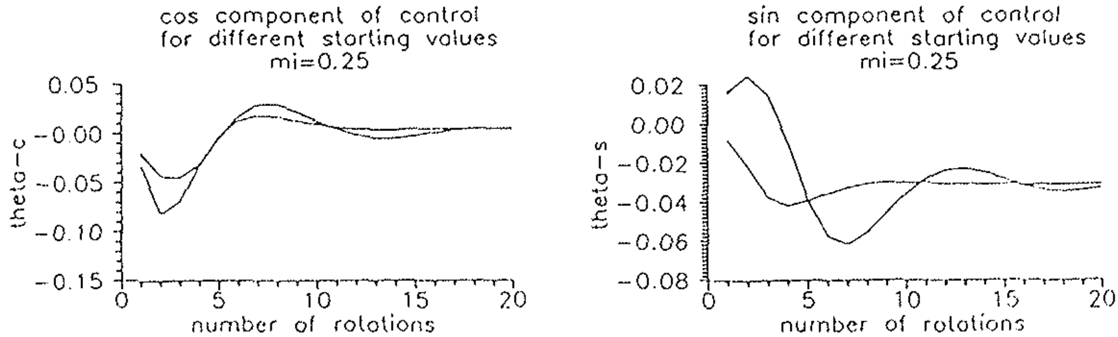
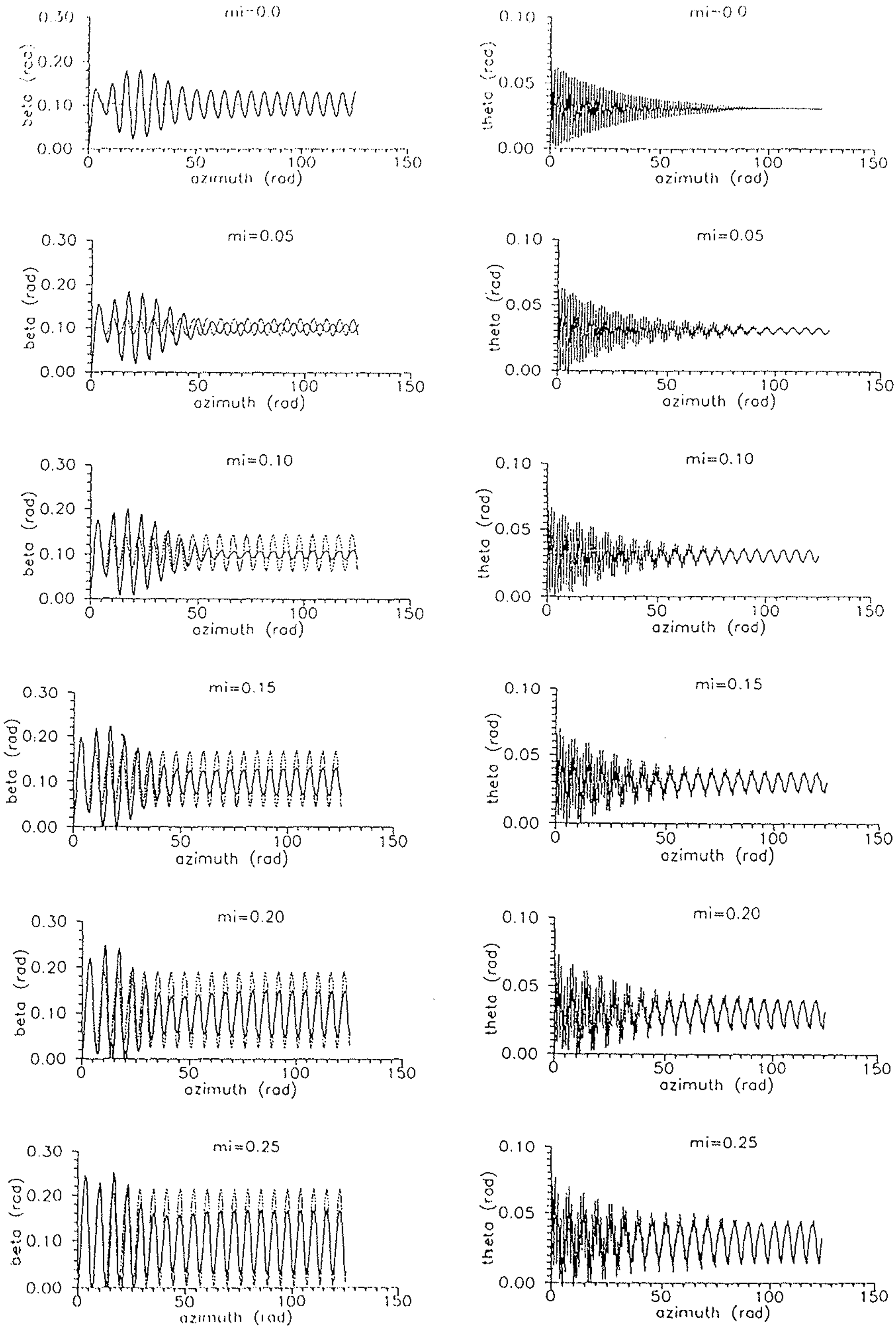


Fig. 6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azimuth (rad)
theta (rad)
mi=0.05
mi=0.10
mi=0.15
mi=0.20
mi=0.25

Fig. 4

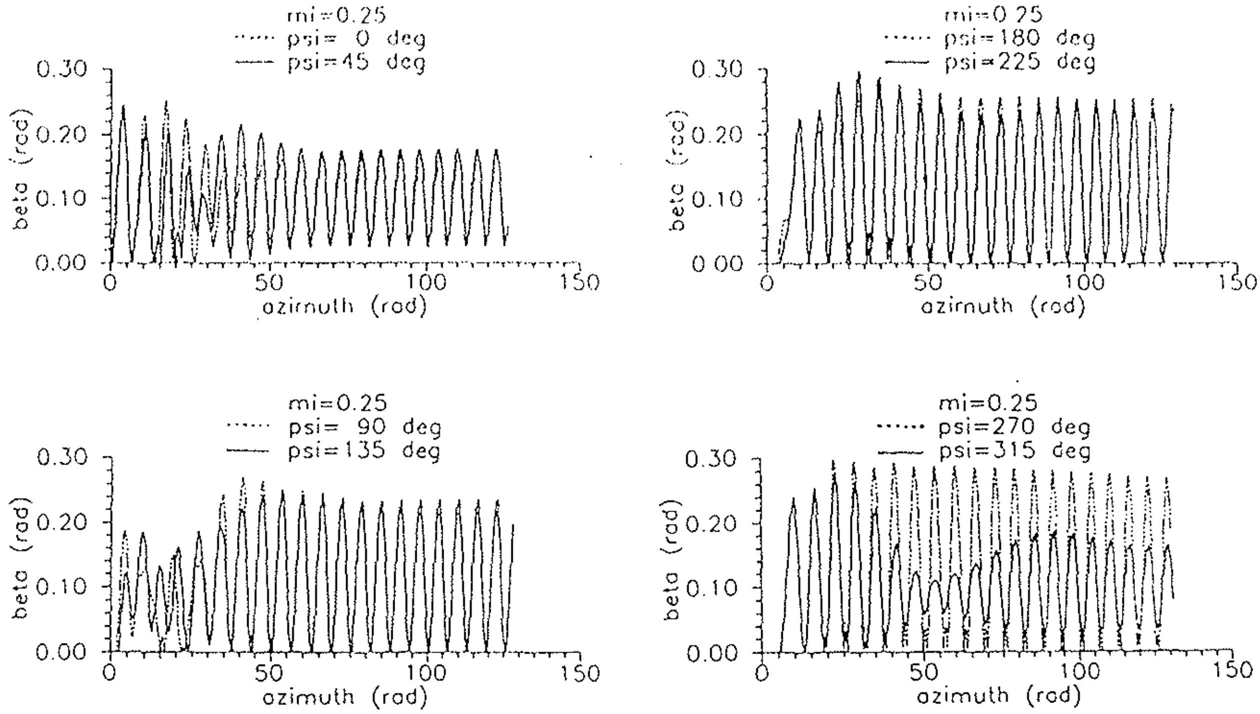

Fig 7

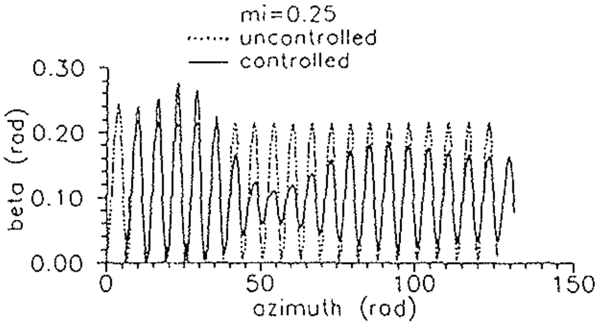

Fig.8

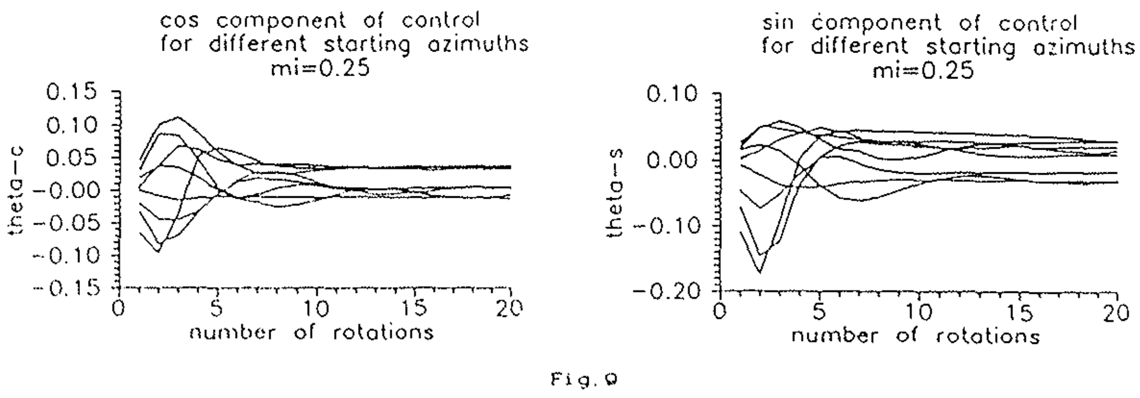

Fig.9