NINETEENTH EUROPEAN ROTORCRAFT FORUM

Paper n° G9

SMART FLAP FOR HELICOPTER ROTOR BLADE PERFORMANCE IMPROVEMENT

by

J. NARKIEWICZ, M. ROGUSZ
WARSAW UNIVERSITY OF TECHNOLOGY
WARSAW, POLAND

September 14-16, 1993
CERNOBBIO (Como)
ITALY

ASSOCIAZIONE INDUSTRIE AEROSPAZIALI
ASSOCIAZIONE ITALIANA DI AERONAUTICA ED ASTRONAUTICA

Smart Flap for Helicopter Rotor Blade Performance Improvement

Janusz Narkiewicz, Marek Rogusz
Warsaw University of Technology
Warsaw, Poland

Abstract

A feasibility study was undertaken for investigating possibility of improving helicopter rotor performance by application of a controlled flap placed at the rotor blade trailing edge.

For computing unsteady aerodynamic loads on airfoil with moving flap an ONERA type stall model has been developed. Using this model optimisation of airfoil performance by controlling trailing edge flap has been tested.

An algorithm has been developed for minimising rotor torque moment while keeping rotor thrust constant.

These methods were included into computer code for simulation of single blade motion.

Calculations were done for rigid, articulated blade with horizontal and feathering hinges and a controlled flap placed at the part of blade span.

By computer simulation the possibility of blade performance improvement by controlled flap was proved.

This research was partly sponsored by Polish State Committee for Scientific Research under grant No 0443/S1/92/03.

Notation

$a_{()}$, $r_{()}$, λ_L, σ_M - coefficients in ONERA model
$A_{()}$ - flap aerodynamic load coefficients,
b - half of chord,
C_D - drag coefficient,
C_L - lift coefficient,
C_M - moment coefficient,
$C_{()L}$ - extended linear characteristics in ONERA model,
$\Delta C_{()}$ - difference between linear extended characteristics in ONERA model and static ones,
C_{Mz} - blade moment coefficient $C_{Mz} = Mz/(0.5\pi R^3 U_T)$,
C_T - blade thrust coefficient $C_T = T/(0.5\pi R^2 U_T)$,
$F_{(X)}$ - load components in ONERA model,
F_{0S} - steady component of load,
F_{0B} - noncirculatory component of load
F_{0I} - circulatory component of load
k - reduced frequency, $k = \omega b/U$,
K - nondimensional velocity of undisturbed flow, $K=U/(b\Omega)$,
M - Mach number of undisturbed flow,
Mz - blade torque moment,
t - time,
T - blade thrust,
U(t) - free stream velocity,
U_T - blade tip velocity,
W_0 - component of airfoil velocity normal to the chord in aerodynamic centre,
W_1 - component of airfoil velocity normal to the chord, resulting from rotation about aerodynamic centre, measured in the distance b from aerodynamic centre,
$\alpha(t)$ - airfoil angle of incidence,
δ - flap deflection angle, positive "downward",
$\theta(\psi)$ - blade pitch angle,
η - blade performance index $\eta = C_{Mz}/C_T$
Γ_0 - circulation for load $()$ component in ONERA equations,
ρ - air density,
ψ - blade azimuth angle, $\psi = \Omega t$,
ω - airfoil pitch angular velocity,
Ω - rotor angular velocity.
$()'$ - differentiation with respect to azimuth angle.

Indexes
D - drag,
L - lift,
M - moment.

1. Introduction.

The crucial component which influences performance and handling qualities of a helicopter is a main rotor.

The limitations exist which influence rotor behaviour and make a challenge for technology development. The problems to be solved are:

- improvement of performance by: rotor pitch stabilisation, elimination of stall regions, proper reactions on gusts and turbulence,
- elimination of vibration by: reduction of unsteady hub loads, diminishing blade stresses, reducing fatigue loads in fuselage,
- suppression of rotor instabilities and flutter,
- avoidance of air and ground resonance,
- reduction of noise by reducing BVI effects.

Some of detrimental effects stem from physical phenomena inherent to rotor dynamics and *aerodynamics and some from the design concept of* rotor itself. To prevent these phenomena new methods of rotor control are being investigated, which makes rotor aeroservoelasticity an important part of rotary wing research and development activity [1].

Two goals of rotor control can be distinguished. First one is to perform required flight conditions, including manoeuvres. This kind of control is called "primary control" here.

The second one is to avoid some unwanted phenomena or improving rotor behaviour and it is called "additional control" in this paper.

From the helicopter first application, the concept of rotor control has not been changed. It is usually done by a swash - plate used for changing rotor blade angle of incidence. This device can provide only collective and harmonic changes of blade pitch, the same for all blades. The research for improving this concept by additional blade pitch control has been undertaken which leads to Higher Harmonic Control (HHC) and Individual Blade Control (IBC) [2] concepts.

Both of these ideas are based on utilising the existing primary control systems to undertake also additional control activity.

The HHC method (and the first trials of IBC) relays on the idea that due to periodic excitations, periodic control should be applied. So on first harmonic pitch changes needed for trimmed flight, higher harmonics are superimposed for all blades in HHC and for each blade separately in IBC.

A new approach which gives hope for joining primary and additional control into one system is smart structure technology. The idea of smart structure utilisation in rotorcraft technology is to design the integrated control system which would adjust blade shape to actual flight conditions and perform rotor primary and additional controls by the same actuating devices.

It has become applicable due to development of material, electronic and mechanical technologies. Feasibility studies of application of smart structures in rotorcraft, which have been published lately [3-5], show that this design concept is promising.

The base idea of smart rotor is changing of blade shape to obtain the required results. The two basic ways of rotor blade shape variation [6] concern blade bending and torsion.

If bending deflections are to be controlled, the required shape of blades in thrust and rotation plane can be obtained by amplifying and suppressing proper bending modes. But the loads needed for exciting bending modes seem to be too high for existing actuator materials.

Controlling of blade torsion, which leads to controlling of blade angle of incidence seems to be a more straightforward way to obtain the desired blade loads because of influencing aerodynamic environment more precisely.

In smart structure application the control of local aerodynamic loads can be achieved by changing airfoil shape [7] or using additional flap [8]. The controlled blade flap has been successfully implemented by Kaman in their products, for many years, recently in K-Max helicopter [9].

Up to now there have been done feasibility studies of controlled flap application to flight mechanics [10], vibration suppression [11] and BVI reduction [12]. The results seem to be promising.

Application of actively controlled flap to problems of flight mechanic was considered in [10,13]. It was proved there, that both trimming rotor and aeroelastic stabilisation is possible by flap control.

The objective of the research undertaken in this study was to investigate the possibility of rotor performance optimisation utilising flap control.
In [14,15] it was proved that there are possibilities of improving rotor performance by applying active control. These studies were aimed on stall and stall flutter suppression using individual blade control. In our research an attempt is undertaken to obtain the same goal by application of flap control.

As the feasibility study we have investigated:
1. unsteady aerodynamic loads modelling for airfoils with flap, stall regime included,
2. dynamic and aeroelastic influence of flap on blade behaviour,
3. control algorithms for optimal rotor performance.

The method of ONERA type stall model for calculating aerodynamic loads on airfoils with flap or with variable camber is proposed and tested. It is included into computer code for calculations of motion of single rotor blade.

The performance of blade with actively controlled flap has been evaluated by computer simulation, showing properties of open-loop system.
A control algorithm was developed and tested for different flight conditions.

2. Blade Model.

The computer model of rotor blade, developed in [16], was used in this study. The base properties of this model are reviewed here for completeness. A single blade of a helicopter rotor in steady flight is considered. An angular velocity Ω of rotor shaft is constant. Air flow velocity relative to rotor shaft can vary in time, which allows to include into analysis gusts and wind.

The rotor hub (Fig.1) can be articulated or hingeless. In the first case it is composed of three or less hinges of different type in arbitrary sequence connected by rigid elements. The length and

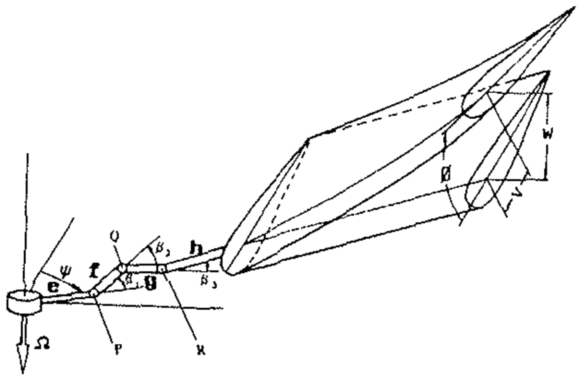

Fig.1

orientation of these elements relative to the shaft allow to account for different design angles like precone, droop or sweep.

The hinge is modelled as rotation of flap, lag or pitch type. Nonlinear damping and/or stiffness can be included as the arbitrary functions of hinge rotation angles and angular velocities. Pitch- flap coupling can also be taken into account, despite the coupling resulting from placement of hub elements .

The blade pitch control is assumed in the form:

$$\theta = \theta_0 + \theta_1\cos(\Omega t) + \theta_2\sin(\Omega t)$$

A blade can be deformable and it is attached to the last segment of the hub or directly to the rotor shaft in the case of a bearingless rotor.

The blade has straight elastic axis. The blade cross sections have symmetry of elastic properties about the chord and there is no section warping. The blade is pretwisted about the elastic axis if it is deformable or about the axis of the last stiff element, if it is rigid. Viscous structural damping of blade deformations can be included into model. The blade deflections are discretized by free vibration modes. The blade stiffness loads are obtained from linear model, valid for small deformations.

The aerodynamic loads are calculated from a two-dimensional nonlinear model described in the next section.

The vector of blade motion generalised coordinates consists of:

- elastic degrees of freedom resulting from discretization of blade deformations by natural modes,
- rigid degrees of freedom corresponding to the rotations in hinges.

Each generalised coordinate is the sum of:

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

Algebraic manipulations for obtaining coefficients in the equations of motion are performed within the computer program, where translation vectors, rotation matrices and their derivatives are arranged according to the chosen hub model.

The blade generalised masses and stiffnesses are calculated within a separate computer program that is run only once for assumed blade configuration before solving (or analysing) the equations of motion. So 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 of equations of motion .

3. Aerodynamic Loads.

Aerodynamic loads modelling is a difficult task in rotary wing problems. The requirements for method of aerodynamic load calculation stem both from flow environment and from algorithms used in analysis of aeroservoelastic problems.

From the flow modelling point of view, the method should cover:

- three components of loads: lift C_L, drag C_D, moment C_M,
- all degrees of freedom, which for 2D case, in a blade section consist of angle of attack, translations along and perpendicular to a chord line, which can be arbitrary functions of time,
- fluctuations of a flow velocity,
- three dimensional effects, which result from a complex shape of blade wake,
- incidence angle up and above stall,
- modelling of different stall types.

The method should be compatible with existing computer codes for rotorcraft stability analysis and simulation.

Some efficient methods developed in computational fluid dynamics are difficult to be adopted in algorithms for solving aeroelastic problems. For instance application of a panel method leads to a large number of states. Also efficiency of some numerical perturbation methods and differential equation solvers could be questioned when such models are utilised.

The requirements which stem from restrictions mentioned above concern:

1. expressing the flow motion in state variables,
2. describing the loads or state changes by ordinary differential equations,
3. covering the possibility of feed-back loops, which occur in control problems.

State variable formulation of aerodynamic loads allows to use existing codes for aeroelastic stability analysis. Differential equations account for arbitrary airfoil motion and model the history of motion which is important in unsteady case.

Along majority of blade span, the flow can be treated as two-dimensional, so the method applied in this study is based on 2D assumption. A method for calculating aerodynamic loads for an airfoil with flap was needed. The ONERA model was chosen for adaptation to this case.

Since its formulation [17] the ONERA model has been modified and extended [18], [19]. The base version taken to this study was developed in [20] and its modifications have been done.

All three components of aerodynamic loads are expressed in "circulatory form". Airfoil motion is described by two variables: W_0 and W_1 (Fig.2), which allows to account for arbitrary motion of airfoil and for angles of attack from $-\pi$ to π.

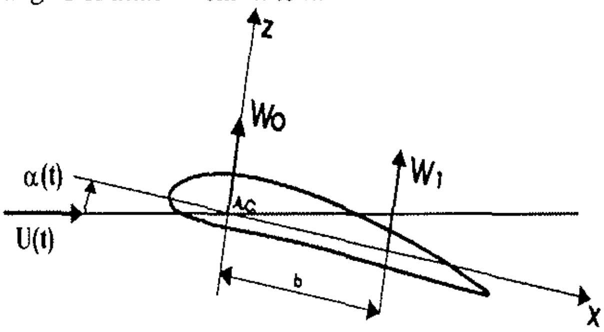

Fig.2

In a general form, the ONERA model contained expressions which depend on the derivatives of W_0 and W_1. It means that the second derivatives of airfoil *displacements were needed and airfoil acceleration* could appear in the right hand side of differential equations. It would make model difficult to be solved because of the lack of methods for integration such type of differential equations.

It was shown in [18], that the influence of time derivatives W_0 and W_1 on aerodynamic loads can be neglected. So a simplified model is used here obtained by dropping expressions with W_0' and W_1'.

Aerodynamic loads in blade sections are calculated using formulae for:

- drag and lift

$$F_{()} = \rho b U[F_{()S}+F_{()B}+F_{()1}]$$

- moment

$$F_M = 2\rho b^2 U[F_{MS}+F_{MB}+F_{M1}]$$

The components in the expressions above describe:

- steady F_{0S},
- unsteady:
  noncirculatory F_{0B},
  circulatory F_{01}.

parts of aerodynamic loads.

Formulas for calculation F_{0S}, F_{0B} ,F_{01} for simplified version of the method are given in Table I.

Table I

	LIFT	MOMENT	DRAG
$F_{()S}$	0	UC_{ML}	UC_{DL}
$F_{()B}$	0	$\sigma_M W_1$	0
$F_{()1}$	$\Gamma_{L1} + \Gamma_{L2}$	Γ_{M2}	Γ_{D2}

For all three load components "circulations" Γ_{02} are obtained as the solutions of ordinary differential equations:

$$\Gamma_{()2}'' + K\,a_{()}\Gamma_{()2}' + K^2 r_{()}\Gamma_{L2} = -K^2 r_{()}\,U\,\Delta C_{()}$$

Circulation Γ_1 in the lift equation is a solution of differential equation:

$$\Gamma_{L1}' + K\,\lambda_L \Gamma_{L1} = K\,\lambda_L(C_{LL}U + W_1)$$

Coefficients $a_{()}$, $r_{()}$, λ_L and σ_M are given in Table II. These values have been obtained during this study as the best fit to experimental data for NACA 23012 airfoil in the range of Mach number and reduced frequency appropriate for helicopter rotor blades.

Table II

	LIFT	MOMENT	DRAG
a	$0.4 + 0.8\Delta C_L^2$	$0.08 + 0.25\Delta C_L^2$	$0.25 + 0.5\Delta C_L^2$
r		$(0.2 + 0.2\Delta C_L^2)^2$	
σ		$-\pi/4.3[1.0+1.4(M^2)]$	
λ	0.68		

For instant angle of attack, static value C_{0S} *and* extended "linear" C_{0L} values should be computed. In our implementation of the model, static values are obtained from table look-up procedure for aerodynamic coefficients. The "extended linear values" are calculated as [21]:

- lift:

$$C_{LL} = \frac{\partial C_{LS}}{\partial \alpha}\sin(\alpha)\cos(\alpha) + C_{L0}$$

$$C_{L0} = 0.131\,, \frac{\partial C_{LS}}{\partial \alpha} = 5.9$$

- moment:

$$C_{ML} = \frac{\partial C_{MS}}{\partial \alpha}\sin(\alpha) + C_{M0}$$

$$C_{M0} = -\,0.008\,, \frac{\partial C_{MS}}{\partial \alpha} = -0.085$$

- drag:

$$C_{DL} = C_{D0} = 0.008$$

Constant time delay is introduced for C_L by assuming that C_L characteristic is linear up to 18° angle of attack, when pitch rate is positive. In the expressions for drag and moment, the increment ΔC_L is calculated without time delay.

This model has been extended to cover calculating of aerodynamic loads on an airfoil with flap or with variable shape (camber). It has been achieved by modification of static airfoil characteristics in the form:

$$C_{()S} = C_{()S} + DC_{()K}(e\,,\delta\,,\,\alpha,\,M)$$

In a general case the increments $\Delta C_{()K}$ are probably functions of: flap length e, its angle of deflection δ, *Mach number and airfoil angle of attack. The increase* of aerodynamic loads due to a growth of an airfoil length (resulting from adding a flap) is accounted for both by the value $\Delta C_{()K}$ and by increasing local blade chord length.

The method of calculation of aerodynamic load was tested in three phases. First aerodynamic loads on airfoil without flap were compared with experimental results. Next the influence of flap motion on an airfoil loads was investigated. Third the possibility of airfoil performance optimisation by application of a flap was checked. All calculations were done for model coefficients adjusted to NACA 23012 airfoil. The parameters of experimental data were chosen to be comparable with those of rotor blade.

In Fig.3 lift obtained from simplified model for airfoil without flap is compared with experimental data for NACA 23012 airfoil [22]. The agreement is adequate.

In Fig.4 the comparison of three aerodynamic load components is done with data for NACA 0012 airfoil [19]. In Fig.5 aerodynamic loads are compared with data for NACA 23010 airfoil [23]. These comparisons give an impression about possibility of applying proposed model to other then NACA 23012 airfoils.

For airfoil with flap, the increments $\Delta C_{()K}$ were assumed to be functions of flap deflection angle δ.

The values of $A_{()}$ based on static experimental results obtained for flap of 10% chord length were:

$$A_L = 0.018\,\delta, \qquad A_M = -0.00525\,\delta,$$
$$A_D = 0.0000115\,\delta^2$$

Sample results of loads calculations on airfoil with flap are shown in Fig.6 for flap motion $\delta = 5°\cos(\omega t)$, for three reduced frequencies k. The influence of frequency on load loops and direction of loop following with angle of attack variation agree qualitatively well with those obtained in [23] for low reduced frequencies.

The possibility of optimisation of airfoil unsteady performance by flap deflection was investigated using Powell's algorithm of minimising function with constrains.

The airfoil motion was assumed in the form:

$$\alpha(t)=\alpha_0+\Delta\alpha\sin(\omega t)$$

and the flap motion

$$\delta(t)=\Sigma_j\Delta\delta_j\sin(h\omega t+\phi_j) \qquad i=1..n$$

The optimised variables were: $\Delta\delta_i$, ϕ_i and h with i=1 or assumed i with h=1.

In Fig.7 the results of minimising function:

$$F=\left|\oint C_L d\alpha\right|+10\left|\oint C_D d\alpha\right|+100\left|\oint C_M d\alpha\right|$$

are compared for flap control with one and three harmonic.

The difficulties of controlling C_D is evident, as the C_D loops do not alter during optimisation. One harmonic control influences only C_M loop, while three harmonic control influences both C_L and C_M. But it causes also additional variation of aerodynamic loads which can be a source of airfoil (blade) excitation.

4. Static Flap Deflection.

As a plant the rigid blade with flap-pitch degrees of freedom was chosen. Blade and flap data are given in Table III and IV.

Table III

Blade Data

rotational speed	26.8	rad/s
length	7.26	m
mass	64.4	kg
inertia about flap hinge	944.0	kg*m²
chord (average)	0.44	m
flap hinge offset	0.16	m
pitch hinge offset	0.30	m
linear twist from shaft	-10	deg
pitch link stiffness	8606	N*m/rad

Table IV

Flap Data

length	0.21R
width	0.10c
distance from the shaft	0.70R

For the plant model first the possibility of influencing blade behaviour by flap deflection was investigated.

Calculations were done for hover and forward flight, different flap angles and different blade pitch. The results are shown in Fig.8 as blade mean thrust C_T versus blade performance index defined as:

$$\eta = C_{Mz}/C_T$$

The curves are functions of blade collective pitch with constant flap deflection δ.

Both blade thrust and moment depend on flap deflections and the reaction is measurable.

From these figures, the possibility of obtaining the same thrust for different pairs of δ and θ and the range of rotor performance improvement can be concluded. The possibility of improving rotor torque while keeping thrust constant appear for higher advance ratios.

These results are valid for assumed flap dimensions and placement along the blade and the blade properties.

5. Blade Performance Optimisation.

The active control is applied for improving blade performance. For constant thrust the minimum of index η is obtained by adjusting δ and θ. The algorithm starts calculations after blade steady state is obtained and is active during sequent rotations.

The algorithm consists of two parts: thrust stabilisation and minimising blade torque moment.

The calculations for algorithm validation were done for constant collective pitch $\theta o=20°$ for different advance ratios. The results are show in Fig.9. The performance index improvement was from 0% to 12% in high speed flight.

Conclusions.

The feasibility study of improving helicopter rotor blade performance by actively controlled flap is done.

For this purpose the ONERA stall model has been extended for covering calculations of aerodynamic loads on airfoils with flap.

The possibility of performance optimisation was checked.

The algorithm for control of blade flap angle and collective pitch has been developed and tested by numerical simulation for hover and forward flight showing possibility of blade performance improvement.

References.

1. Friedmann P.P., "Rotary-Wing Aeroservoelastic Problems", Proceedings of European Conference on Aeroelasticity, 1991.
2. Polychroniades M., "Generalized Higher Harmonic Control Ten Years of Aerospatiale Experience", XVI European Rotorcraft Forum, Glasgow 1990, Pap. No. III.7.2.
3. Ormiston R.A., "Can Today's Smart Structures Make Helicopter Better?", Fourth Workshop on Dynamics and Aeroelastic Stability Modeling of Rotorcraft Systems, The University of Maryland, November 19-22, 1991.
4. Strelhow H., Rapp H., "Smart Materials for Helicopter Rotor Active Control", AGARD/SMP Specialist's Meeting on Smart Structures for Aircraft and Spacecraft, 5-7 October, 1992, Lindau, Germany.
5. Straub F.K., "A Feasibility Study of Using Smart Materials for Rotor Control", 49th American Helicopter Society Forum, St.Louis, Mo, May 1993.
6. Chan W.P., Samak D.K., Chopra I., "Development of Scaled Rotor Using Smart Structures Technology", Fourth Workshop on Dynamics and Aeroelastic Stability Modeling of Rotorcraft Systems, The University of Maryland, November 19-22, 1991.
7. Zurhorst S., Ory H., "Active Twist Control of a Wind Tunnel Model Wing", Paper 91-147, Workshop on Smart Material Systems and Structures, Aachen, FRG, June 6, 1991.
8. Hanagud S., Roglin R.L., Nagesh Babbu G.L., "Smart Airfoils for Helicopter Control", Pap. No. 119 (H04), XVIII European Rotorcraft Forum, Avignon, France, September 15-18, 1992.
9. Wei F.Sh., Jones R., "Correlation and Analysis for SH-2F 101 Rotor", Journal of Aircraft, Vol.25, No.7, July 1988.
10. Yillikci Y.K., "Trimming of Rotor Blades with Periodically Deflecting Trailing Edge Flaps", Paper No.91-43, XVII European Rotorcraft Forum, Berlin, September 1991.
11. Millott Th., A., Friedmann P.P., "The Practical Implementation of an Actively Controlled Flap to Reduce Vibrations in Helicopter Rotors", 49th American Helicopter Society Forum, St.Louis, Mo, May 1993.
12. Straub F.K., Robinson L.H., "Dynamics of a Rotor with Nonharmonic Control", 49th American Helicopter Society Forum, St.Louis, Mo, May 1993.
13. Yillikci Y.K., "Aeroelastic Analysis of Rotor Blades with Flap Control", Paper No. E04, XVIII European Rotorcraft Forum, Berlin, September 1991.
14. Kretz M., "Active Elimination of Stall Conditions", Vertica, Vol.6, No.2, 1982.
15. Kretz M., "Active Expansion of Helicopter Flight Envelope", XV European Rotorcraft Forum, Amsterdam 1989, Paper No.53.
16. Narkiewicz J., Lucjanek W., "Generalized Model of Isolated Helicopter Blade for Stability Investigation", XVI European Rotorcraft Forum, Glasgow 1990, Pap. No. III.8.2.
17. Dat R., Tran T.C., Petot D., "Modele Phenomenologique de Decrochage Dynamique sur Profil de Pale d'Helicoptere", ONERA T.P. No. 1979-149.
18. Peters D.A., "Towards a Unified Lift Model for Use in Rotor Blade Stability Analysis", Journal of the American Helicopter Society, Vol.30, No.3, July 1985.
19. Petot D., "Modelization du Decrochage Dynamique par Equations Differentielles", La Recherche Aerospatiale, No.5, 1989.
20. Bartler T., Narkiewicz J., Syryczynski J., "Circulatory ONERA Model of Dynamic Stall on Airfoil", (in Polish), Reports of Aviation Institute, Warsaw, 1993.
21. Barwey D., Gaonkar G.H., Ormiston R.A., "Investigation of Dynamic Stall Effects on Isolated Rotor Flap-Lag Stability with Experimental Correlation", Journal of the American Helicopter Society, Vol.36, No.4, 1991.
22. Tyler J.C., Leishman J.G., "Analysis of Pitch and Plunge Effects on Unsteady Airfoil Behavior", Journal of the American Helicopter Society, No.3, July 1992.
23. Leishman G. J., "Unsteady Lift of an Airfoil with a Trailing- Edge Flap Based of Indicial Concepts", XVIII European Rotorcraft Forum, Avignon, France, September 15-18, 1992.

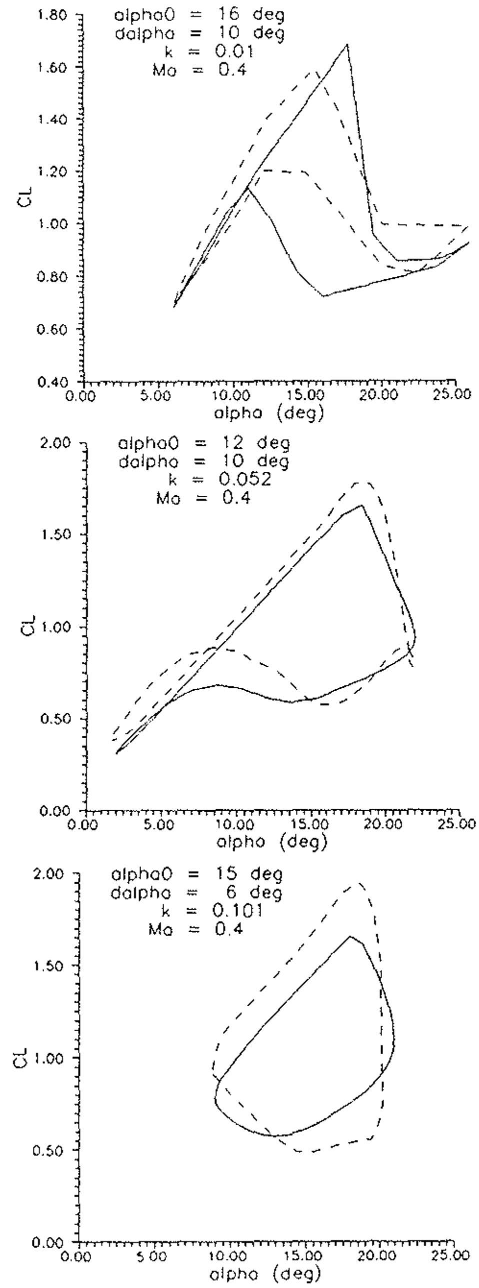

Fig.3

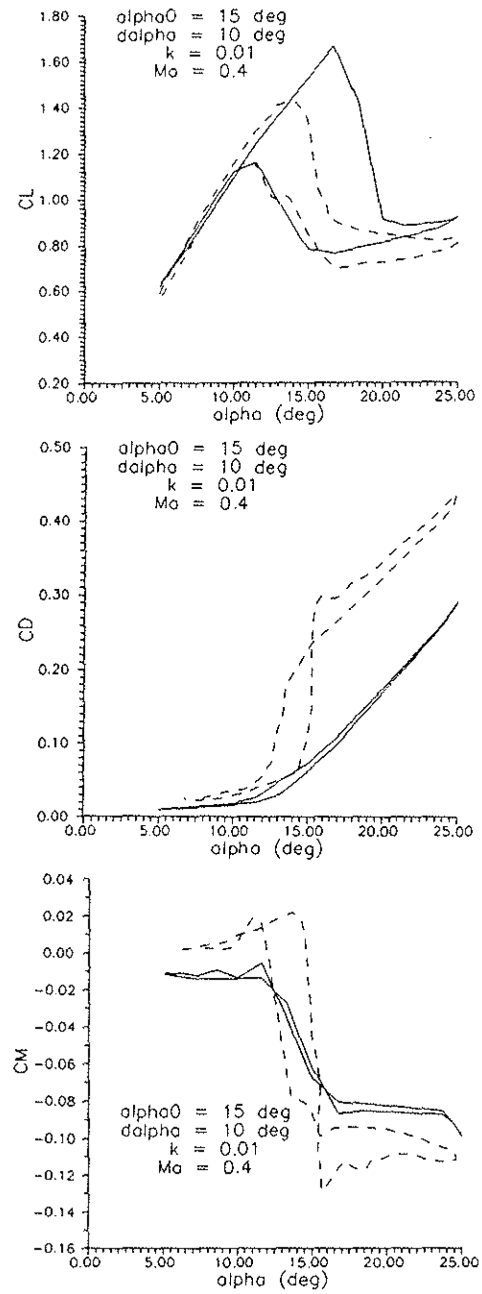

Fig.4

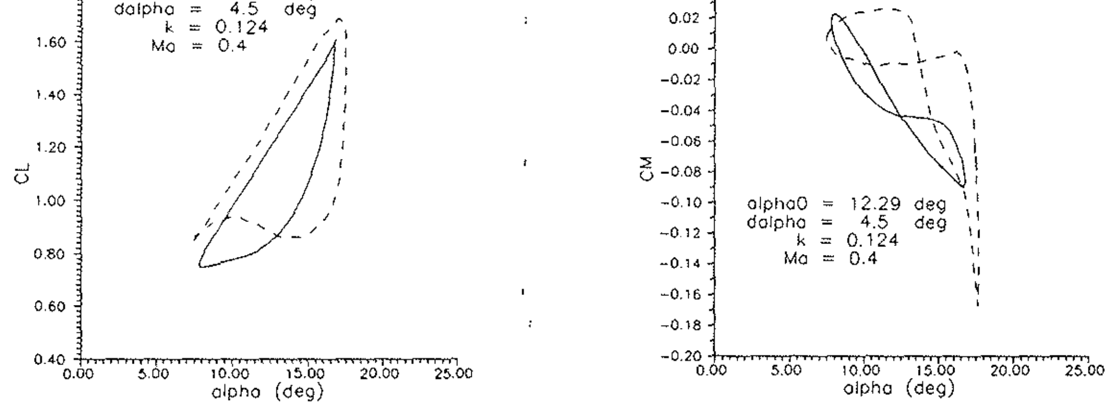

Fig.5

Fig.6 on page G9-11

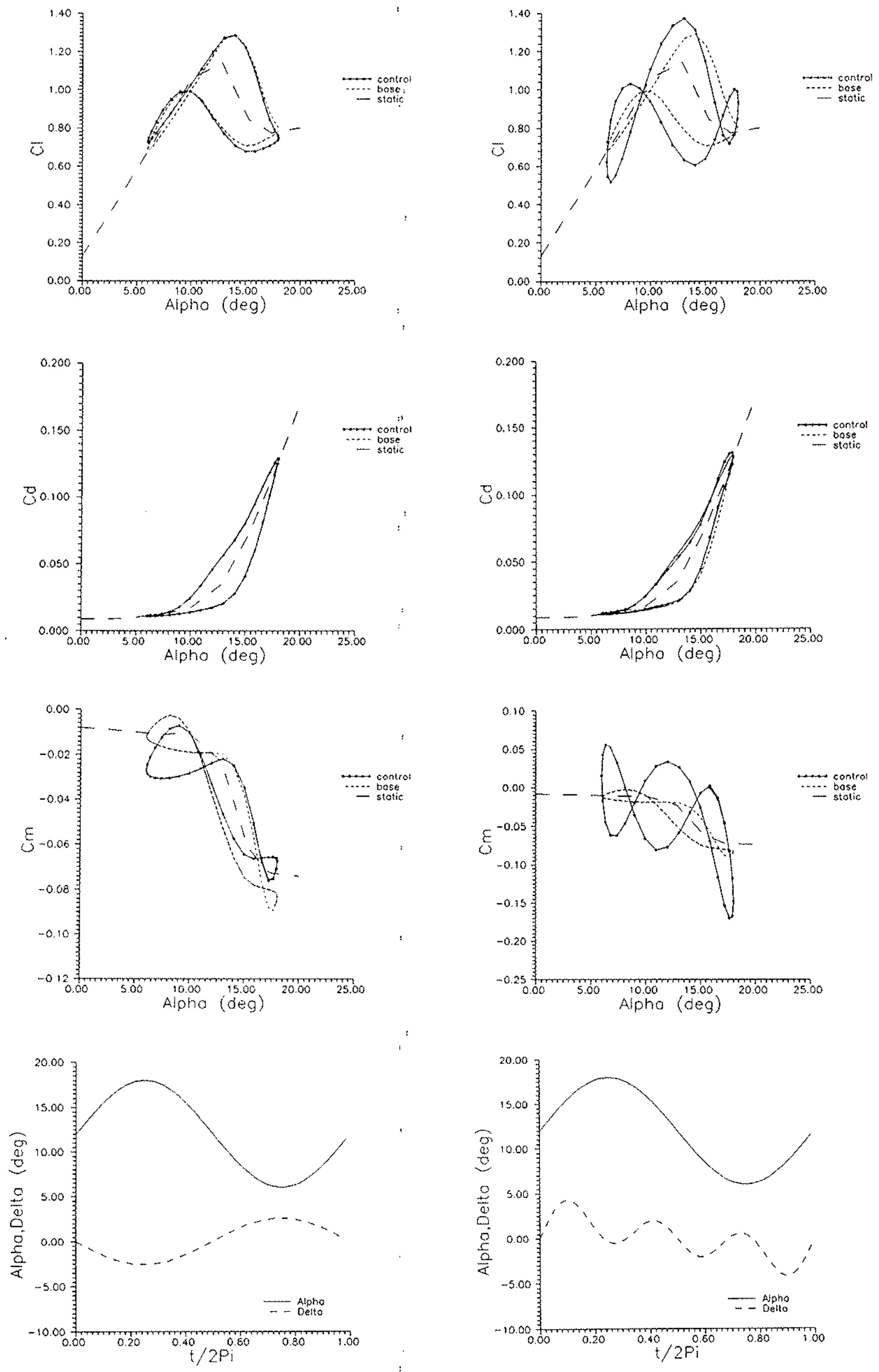

Fig.7

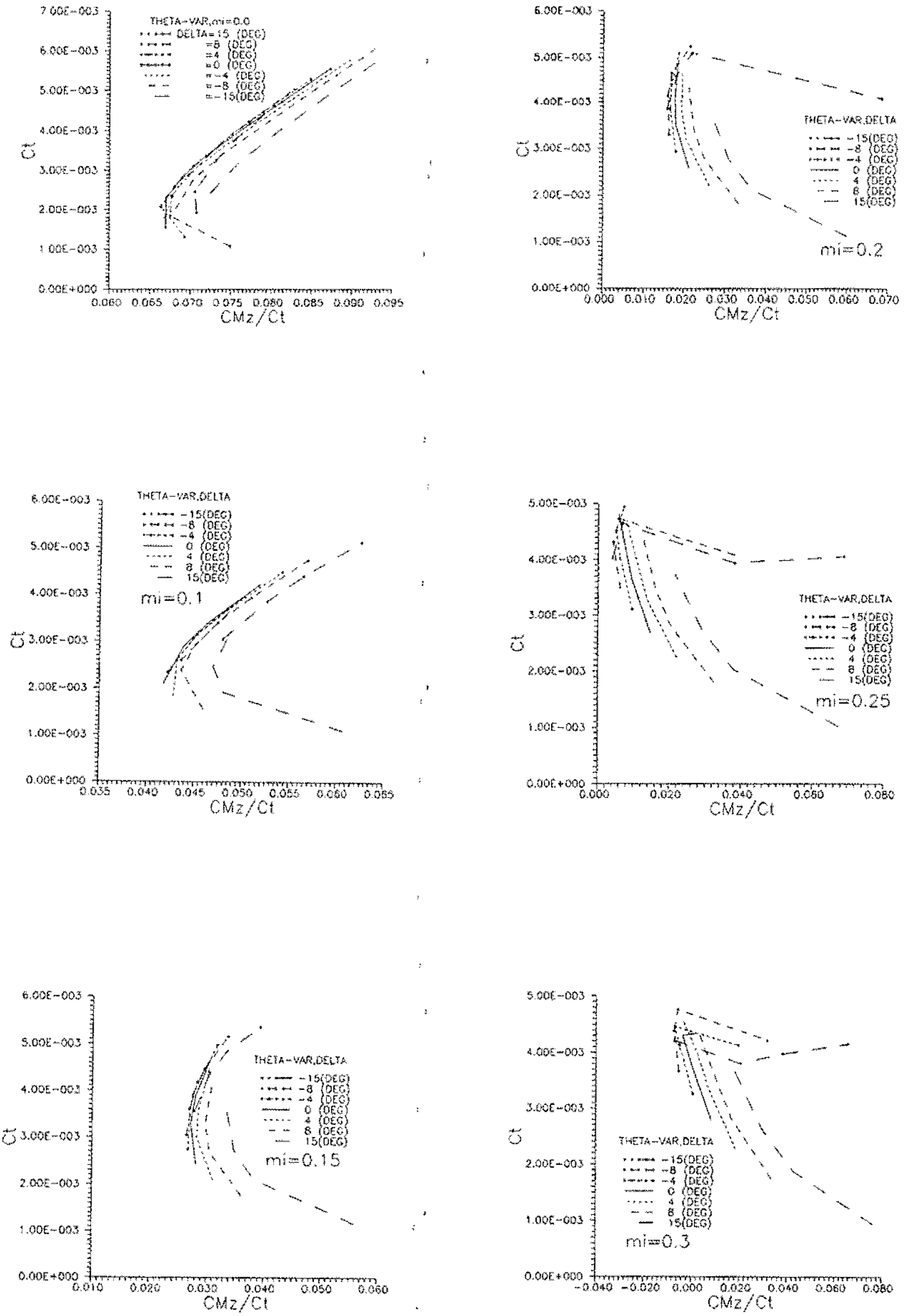
THETA-VAR,mi=0.0
DELTA=15 (DEG)
=8 (DEG)
=4 (DEG)
=0 (DEG)
=-4 (DEG)
=-8 (DEG)
=-15(DEG)
Ct
CMz/Ct
THETA-VAR,DELTA
-15(DEG)
-8 (DEG)
-4 (DEG)
0 (DEG)
4 (DEG)
8 (DEG)
15(DEG)
mi=0.1
mi=0.15
mi=0.2
mi=0.25
mi=0.3

Fig. 8

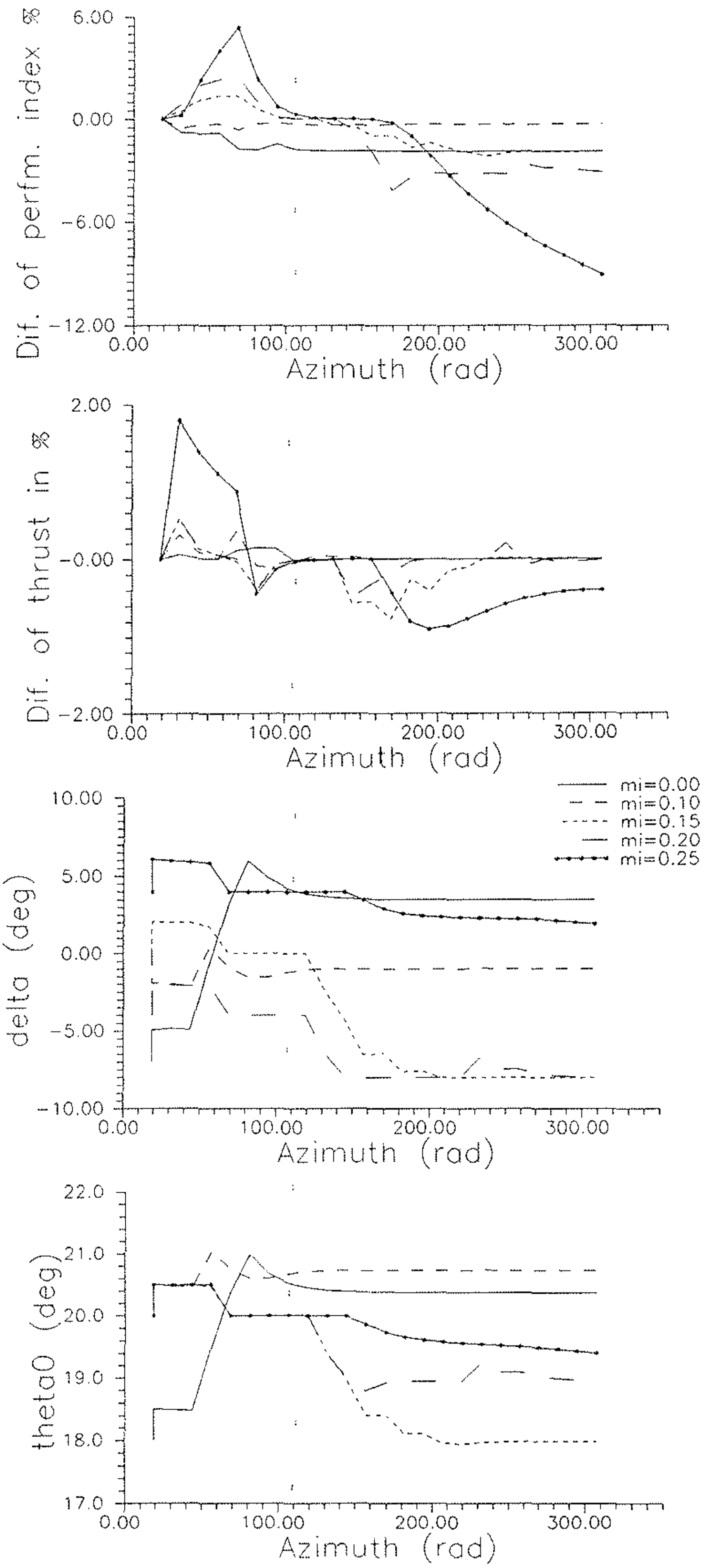
Dif. of perfm. index %
6.00
0.00
-6.00
-12.00
0.00
100.00
200.00
300.00
Azimuth (rad)
Dif. of thrust in %
2.00
-0.00
-2.00
0.00
100.00
200.00
300.00
Azimuth (rad)
mi=0.00
mi=0.10
mi=0.15
mi=0.20
mi=0.25
delta (deg)
10.00
5.00
0.00
-5.00
-10.00
0.00
100.00
200.00
300.00
Azimuth (rad)
theta0 (deg)
22.0
21.0
20.0
19.0
18.0
17.0
0.00
100.00
200.00
300.00
Azimuth (rad)

Fig.9

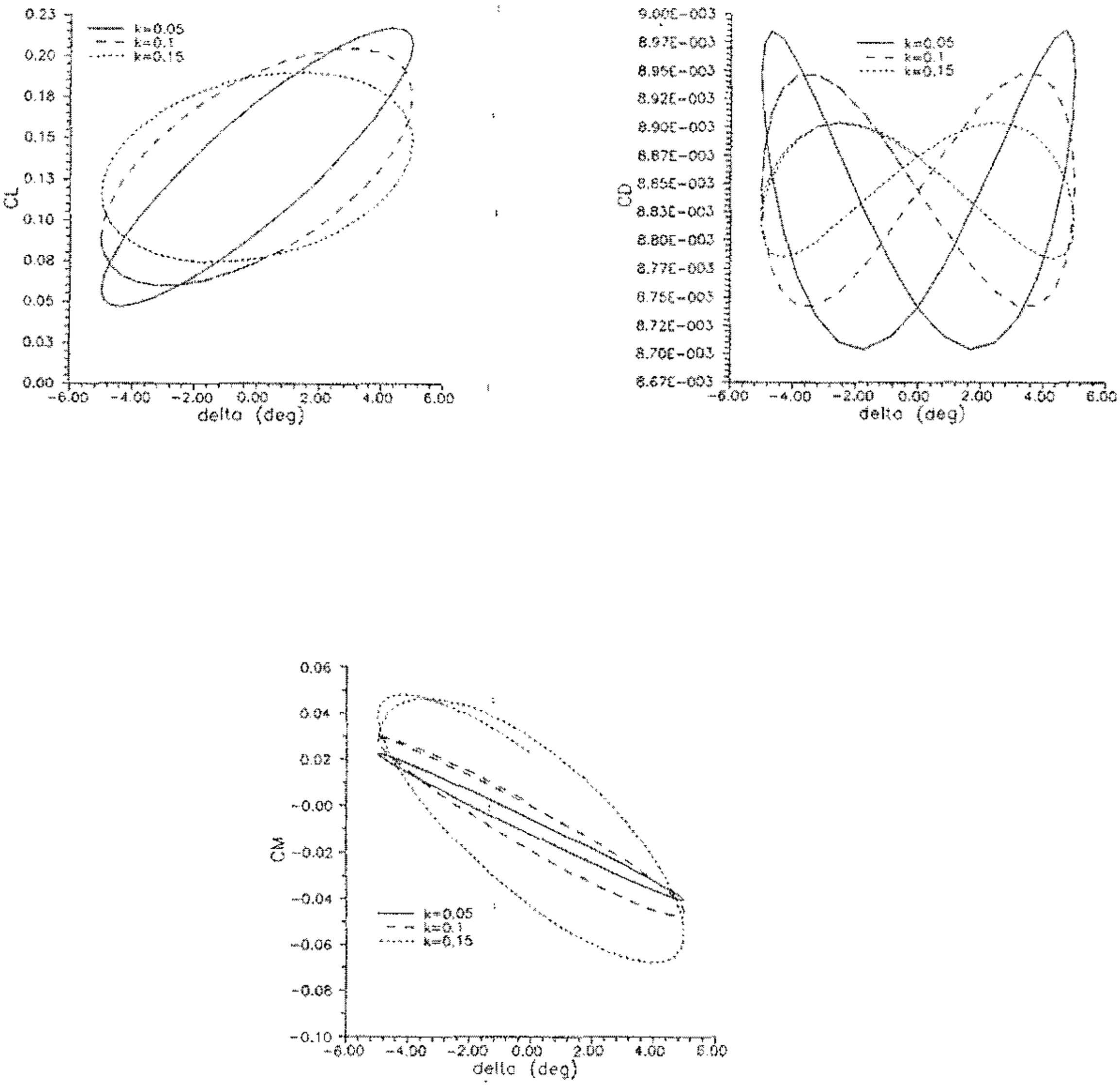

Fig.6