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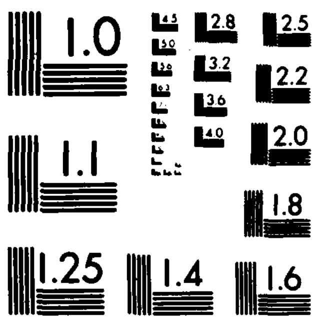


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# DEPARTMENT OF DEFENCE

## DEFENCE SCIENCE AND TECHNOLOGY ORGANISATION

## AERONAUTICAL RESEARCH LABORATORIES

MELBOURNE, VICTORIA

AERODYNAMICS REPORT 158

# CALCULATION OF THE AERODYNAMIC FLOW FIELD IN THE VICINITY OF A SEA KING HELICOPTER

by

K. R. REDDY



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AERODYNAMICS REPORT 158

# CALCULATION OF THE AERODYNAMIC FLOW FIELD IN THE VICINITY OF A SEA KING HELICOPTER

by

K. R. REDDY


*SUMMARY*

*A mathematical model of the rotor wake in the vicinity of a rotorcraft fuselage is formulated taking into account the mutual interference between the rotor and fuselage. It is assumed that the vortex sheet behind each rotor blade rolls-up into a single, blade tip vortex and that the fuselage can be represented by constant source potential flow panels. The velocity field given by the combined rotor wake and fuselage model is presented for the case of a Sea King helicopter in forward flight.*



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Box 4331, P.O., Melbourne, Victoria, 3001, Australia

## CONTENTS

## NOTATION

| | |
|---|---|
| $a_{ij}$ | core size of the vortex element $ij$ |
| $L_{ij}$ | length of the vortex element $ij$ |
| $\bar{n}$ | unit vector normal to the panel |
| $p, q$ | reference points on fuselage surface panels |
| $P_{ij}$ | vortex wake reference points |
| $r$ | distance between fuselage surface reference points |
| $R$ | outer radius of the rotor |
| $S$ | surface area of the fuselage |
| $\bar{V}$ | velocity vector |
| $V_{\sigma}$ | source induced velocity |
| $V_{\sigma n}$ | normal component of source induced velocity |
| $V_{\infty}$ | free stream velocity |
| $V_x$, $V_y$, $V_z$ | velocity components in the $xyz$ coordinate axis system |
| $W$ | weight of the rotorcraft |
| $xyz$ | cartesian axes (see figure 4) |
| $\alpha_T$ | tip-path-plane angle |
| $\Gamma_{ij}$ | vortex strength of the element $ij$ |
| $\mu$ | advance ratio |
| $\sigma$ | surface source distribution |
| $\psi$ | azimuth angle |
| $\Omega$ | rotational speed of the rotor |

# I. INTRODUCTION

The operation of helicopters is influenced significantly by the aerodynamic contributions of several components, which include the main rotor, tail rotor, fuselage, landing gear, and fairings. In the early days of development of helicopters, the engineering effort was mainly concentrated on the mechanics of flight and the development of the rotors, controls and power-plant. Little attention was paid to the aerodynamics of non-rotating parts and the effect of their interference on rotating components. Designers are now trying to reduce fuel consumption, operating costs, noise, and vibration, and improve safety, performance, and ride quality.

The importance of fuel economy and reduction in operating costs needs no elaboration. The high noise and vibration levels of the present-day helicopter reduce its usefulness and operational effectiveness. Potential gains to be made in reducing these levels include increased passenger and crew comfort, reduced detection for military helicopters, and greater acceptance by the community for civil helicopters. The importance of a knowledge of the helicopter aerodynamic environment in the analysis of vibration and noise is unquestionable.[1]

An accurate determination of the flow field in the vicinity of a helicopter is also required for the effective use of a helicopter as a weapons platform.[2] Advanced weapons and their mounts have become integral items in the design process for modern aircraft. In the type of analysis used, the capability of simulating detailed and complete geometric shapes is essential. The individual aircraft and weapon configurations are generally complex and nonslender in shape. Also, the relative sizes and locations of weapon and aircraft are such that extreme detail is required in describing the geometry and in the calculation of the velocity field in the vicinity of the weapon.[3] Knowledge of the velocity field is required in the calculation of load distribution, forces, moments, and the trajectory characteristics of a store released from the parent rotorcraft. Since a free-flight projectile such as a rocket, when fired from a helicopter, initially travels at a speed which is the same order of magnitude as the flow velocities, the flow field induced by the rotor wake system can have a significant effect of the rocket trajectory. This can necessitate some form of aiming compensation or special firing techniques.

Recent years have also seen a revival of interest in helicopter operations near the ground, known as nap-of-the-earth (NOE) operations.[4] This capability is necessary for any military aircraft required to operate in the high-threat forward battle zones, not only for attack and surveillance, but also for utility, logistics, and support for other helicopters in forward areas. Good NOE characteristics generally necessitate a more compact design. Because the helicopter components are close to one another, interactions of the separate aerodynamic effects arising from each component are likely to be of greater significance. Hence knowledge of the aerodynamic environment will help to evaluate the NOE performance of a particular rotorcraft.

The ability of the helicopter to hover, and therefore to take-off and land in confined areas, has proven to be of significant value in aerial spraying of insectisides. The helicopter rotor wake has a significant effect on spray dispersal. Even in hover, a rotor cannot generate a uniform velocity distribution (since, near the circumference of the disc the rotor's velocity is far higher than near the centre). As the speed of the rotorcraft increases, this non-uniform nature of the flow becomes more pronounced. Therefore, an understanding of the flow field in the vicinity of the helicopter could be used to determine the positioning of the spray nozzles in order to achieve a more uniform distribution of insectisides.[5,6]

Because of the large number of helicopter components influencing the flow field, its accurate determination is extremely difficult. Mutual interaction of the vortex elements generated by the separate components will deform the vortices, which, because of viscosity, will eventually dissipate. To date, mathematical models that can simulate the flow about individual or a couple of aerodynamic elements have been developed.[7-16] However, these models do not include the effect of all aerodynamic interactions.[17]

To solve this complex interactive flow problem, various simplifying assumptions are necessary. In particular, these include the number and geometry of aerodynamic elements, and the flow field generated by each element. In this report, only the two most significant aerodynamic elements, which are the main rotor and fuselage, are considered.

Reasonably simple analytical models of the flow beneath the isolated rotor, both in hover and forward flight, are now available for flight simulation studies. Similarly, simple panel methods which mathematically model the fuselage are beginning to be used on a routine basis in fixed wing aircraft and are being used in rotary wing aircraft design and analysis.[18] Combining both of these analyses into a truly representative model of the flow field is a task which would tax even the largest computer in use.[19] However, the problem can be approached progressively by gradually increasing the complexity of the model.

As an initial step towards investigating the rotor and fuselage interaction problem, an iterative approach is adopted. The effect of the rotor on the fuselage is first calculated, with the fuselage considered to be immersed in the flow field below an isolated rotor. Then the effect of the fuselage on the rotor is determined, where the components of flow generated by the fuselage exposed to the freestream (time and spatial average of rotor flow) are added to the self-induced inflow of the rotor. Mathematical details of this approach are presented in the following sections. This method is similar to the work reported in Reference 7. However, a more realistic blade tip vortex, rotor wake model is used instead of a simple vortex-tube wake model.

In developing the theory, consideration has been limited to aircraft having a single rotor. No restriction has been placed on the number of blades which the rotor may have, however. The aircraft has been assumed to be in steady forward flight. The spatial and time average of the rotor induced flow is used to calculate the fuselage on-set flow. It is assumed that the fluid is inviscid and incompressible. It is also assumed that the flow about the fuselage is not separated, so the calculation of flow field must be limited to regions where separation effects are negligible.

## 2. POTENTIAL FLOW ABOUT THE FUSELAGE USING A PANEL METHOD

The aerodynamic flow around a fuselage may be represented by models of several different levels of complexity for each of which there is an appropriate set of differential equations. These equations are solved approximately by one of a variety of numerical methods. The methods used include the various forms of the finite difference and finite element methods. The concept behind these methods may involve direct discretisation of the governing equations or indirect discretisation based on variational principles.

In the finite difference or finite element based programs, there is the need to construct a grid fitted to the computational domain and its boundaries. For the flow domains typically encountered by rotorcraft aerodynamicists, e.g. the infinite space exterior to a complete military rotorcraft equipped with a range of awkwardly-shaped missiles, guns, etc., the construction of a suitable three-dimensional grid is a major task.

The method which is very widely used in the aircraft industry is the so-called 'panel' method, sometimes known as the 'boundary integral' or 'surface singularity' method. Such methods are currently in production use by most of the fixed wing aircraft manufacturers.[20,21] These panel models are now appearing in rotary wing aircraft design and analysis.[22] Panel methods have the useful property that a grid need not be constructed for the whole flow field. Instead, it is possible to formulate the problem in such a way that the unknown quantities (source and vortex) are positioned only on the surface of the configuration (see figures 1 and 2). Once these unknown quantities have been determined, the flow solution may be readily obtained at any point. The practical advantages offered by this simplification are that the number of unknowns in the problem are reduced by an order of magnitude and that the geometric data are easily prepared. However, it is interesting to note, as evidenced by recent literature,[23] that the finite element method is proving a suitable tool for certain classes of aerodynamic problems not amenable to panel methods. In this document, the panel method is used to model the helicopter fuselage.

The problem considered is that of a steady flow of a perfect fluid about a three-dimensional body. References 20 and 21 provide a comprehensive report on this subject and contain detailed discussions of the computational aspects of the panel method.

A source type panel method for three-dimensional flow around an arbitrary non-lifting body configuration has been adopted here. The velocity $\bar{V}(x, y, z)$ of the flow is calculated by summing the free stream velocity $\bar{V}_\infty$ and the velocity $\bar{V}_\sigma(x, y, z)$ induced from the source covered surface area $S$ of the body[13] as shown in figure 1, i.e.

$$\bar{V}(x, y, z) = \bar{V}_\infty + \bar{V}_\sigma(x, y, z) \tag{1}$$

The source induced velocity $\bar{V}_\sigma$ at a reference point $p$ due to the source strength $\sigma(q)$ at the point $q$ is

$$\bar{V}_{\sigma(p)} = -\frac{1}{4\pi}\nabla\iint_S \frac{\sigma(q)}{r(p,q)}\,ds \tag{2}$$

where $r(p,q)$ is the distance from $q$ to $p$.

Using the boundary conditions

$$\bar{V}(x,y,z) = \bar{V}_\infty \text{ for } x,y,z \to \infty \tag{3}$$

and

$$V_{\sigma n} = -\bar{n}(p)\,.\,\bar{V}_\infty \text{ at } p \tag{4}$$

the well known integral equation[20]

$$2\pi\sigma(p) - \nabla\iint_S \frac{\sigma(q)}{r(p,q)}\,ds = -\bar{n}(p)\,.\,\bar{V}_\infty \tag{5}$$

is established for the source distribution $\sigma(q)$, where $\bar{n}$ is the unit vector normal to the panel, and $V_{\sigma n}$ is the normal component of the source induced velocity.

Numerical solution of equation (5) is obtained using a panel technique. The present method deals with constant source panels as basic building blocks. These are arranged in networks on the boundary surfaces as shown in figure 2. A panel is four sided and defined by specifying the spatial coordinates of its four corner points. Each panel is made planar by projecting the given corner points onto the plane passing through the mid-points of the lines connecting three pairs of adjacent corner points. Within this constraint, a panel may be any size and shape. Specification of two corner points to be coincident results in a triangular panel. One boundary condition per panel is used. All boundary conditions are imposed at discrete points called boundary or control points. Each boundary condition statement consists of the specification of the coordinates of the boundary point, a unit vector normal to the panel, and the required velocity component (zero for this case) along the unit vector.

Using the above approximations, the integral equation is decomposed into the summation equation

$$2\pi\sigma_i + \sum_{j(\neq i)}^{N} \sigma_j \bar{n}_i\,.\,\bar{V}_{ij} = -\bar{n}_i\,.\,\bar{V}_\infty \;(i = 1, 2, 3 \ldots N) \tag{6}$$

where $\bar{V}_{ij}$ is the velocity vector induced at $p_i$, the control point on the $i$th panel, by a unit source distribution on the $j$th panel; $\sigma_i$ is the unknown surface source strength of the $i$th panel; and $\bar{n}_i$ is the normal vector to the $i$th panel.

The panel source strengths $\sigma_i$ are determined by solving the above set of linear algebraic equations. Once the source strengths have been obtained, the flow direction and magnitude can be computed at the surface boundary or control points and at any arbitrary point located off the surface.

Computer time requirements vary with the number, type, and disposition of the panels. Numerical accuracy depends on the panel layout and the panel density. The source panel density should be increased in regions of high curvature and the entire panel model must be selected with a view toward minimizing the differences in source strengths between adjacent panels.

The next step in the analysis is to establish the free stream velocity $V_\infty$. The fuselage of a helicopter operates in a flow field affected by both the forward motion and rotor downwash. The relative importance of these effects varies throughout the flight regime. At hover, only rotor downwash is present, while at high speeds, there is very little downwash effect. In general the resulting flow field is highly non-uniform and time dependent. In the following section, a blade tip vortex wake model is developed to calculate this on-set flow.

## 3. ROTOR WAKE MODEL

A detailed wake model for calculating the induced velocity would represent the rotor blades by lifting surfaces, and the rotor wake by vortex sheets. The calculation of the wake geometry would involve the computation of the distortion and roll-up of these vortex sheets due to their own induced velocities and those of lifting surfaces. As the vortex sheets roll up into line vortices, viscous effects would become important in the vortex core[24]. Because the above model would require an excessive amount of computer time, further simplifications are made in the model adopted here. In the mathematical analysis,[25-29] the rotor blades are represented by lifting lines (bound vortices), and the vortex wake is idealized as a set of trailing tip vortices as shown in figure 3. As a part of the computational procedure, the tip vortex behind each blade is broken into convenient straight line segments.[30] The wake configuration at any instant is defined by the location of these vortex segments. Each vortex filament has length $L_{ij}$, core radius $a_{ij}$, and vortex strength $\Gamma_{ij}$. Induced velocities due to these vortex filaments are calculated using the well known Biot-Savart relation.

## 4. COMPUTATIONAL PROCEDURE AND RESULTS

Once the isolated fuselage source panel and vortex rotor wake computational models have been developed, the next task is to determine the source strength of each panel representing the fuselage and the geometrical distribution of the vorticity that constitutes the wake model. This is carried out by an iterative numerical process. The computations are initiated by specifying the initial wake configuration. Here a helical configuration is assumed. Using this wake geometry, the induced velocity in the rotor plane is calculated. The spatial and time average of this induced velocity is added to the rotorcraft flight velocity to obtain the fuselage on-set flow, $V_\infty$. For this on-set flow, the source strength $\sigma_i$ of each fuselage surface panel is calculated. Then the velocity at each wake reference point $P_{ij}$ (see figure 3) is calculated by summing the velocity contributions of all vortex and source elements in the flow field. The wake reference points are now moved at their computed velocity for a small time interval, thus giving a new wake geometry for calculations. The above process is repeated until convergence. The computer time and rate of convergence of the numerical procedure will depend on the total number of wake vortices taken into account, the magnitude of the time interval, the number of source panels used to represent the fuselage, and the advance ratio of the rotorcraft.

A 5-bladed Sea King rotor of diameter 18·9 m was used in the calculations. The rotor hub is fully articulated and the rotor blades have a twist of −8 deg and untapered plan form. The details of the operating conditions for which the results are calculated are given below.

| | |
|---|---|
| Advance ratio | $\mu = 0{\cdot}10$ |
| Rotor forward tilt | $\alpha_T = 0{\cdot}6583$ deg |
| Rotor speed | $\Omega = 21{\cdot}89$ rad/s |
| Rotorcraft mass | $W = 7200$ kg |

The panelling scheme used for modelling the Sea King helicopter fuselage is shown in figure 4. The thin tail end of the fuselage is not included in the calculations.

The velocity field calculated using the iterative numerical method is presented in the form of contour plots in figures 5 and 6. These contours provide the $x$, $y$, and $z$ components of the velocity. The plots include the velocity contours in two different planes, one in the plane of the rotor (figure 5) and the other 6·1 m below the rotor (figure 6). All quantities are in non-dimensional form, with lengths scaled by the rotor radius $R$ and velocities by the rotor tip speed $\Omega R$. Plotted results correspond to the rotor configuration where $\psi = 0$. The non-uniform nature of the inflow velocity at the rotor disk is evident from figure 5*c*. It is also clear that there is a substantial difference in inflow levels for the fore and aft portions of the rotor in forward flight. The characteristic upwash at the front of the rotor disc is well defined. The contribution of the fuselage to the total flow varies from zero to ten percent. Some of the results are presented in figure 7, where the total rotor flow is shown both with and without the effect of fuselage included.

## 5. CONCLUDING REMARKS

The velocity distribution about a Sea King helicopter fuselage immersed in a rotor wake has been calculated using a constant source, potential-flow panel method combined with blade tip-vortex rotor wake model. For the forward flight considered, the effect of the fuselage on the rotor downwash is about ten percent.

The mathematical model has been tested for various flight conditions. The numerical solution converges quite rapidly at high advance ratios. It takes about 70 minutes CPU time on the PDP DEC 10 computer time for the Sea King helicopter rotor wake geometry to converge when the advance ratio is 0·1. This computer time is not excessive considering the complexity of the model. However, the vortex wake convergence becomes very slow as hovering flight is approached.

Previous investigations have shown[30–32] the tip vortex representation of the rotor wake is adequate for predicting rotor induced flow with reasonable accuracy. For the attached, well behaved flow around slender shapes, a potential flow model can give adequate results. However, the helicopter fuselage with its large protrusions has a rapidly changing cross section which is conducive to flow separation. Hence for a more accurate prediction of the flow field, not only must the attached flow be represented, but the regions of separated flow have to be modelled. Hence as a further improvement to the present mathematical model it is proposed to include the effect of the fuselage flow separation and in addition the assumed uniform onset flow will be replaced with a non-uniform one more closely representative of the actual flow. A wind tunnel experiment is also planned to test the validity of the mathematical model.

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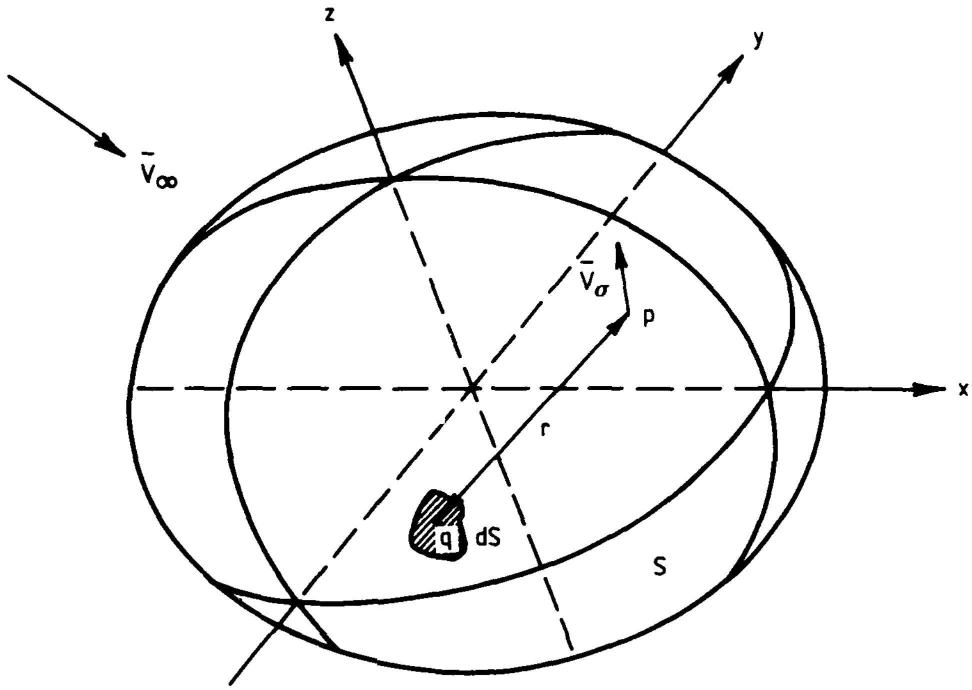



FIG. 1. ILLUSTRATION OF FLOW PAST THREE-DIMENSIONAL FUSELAGE BODY

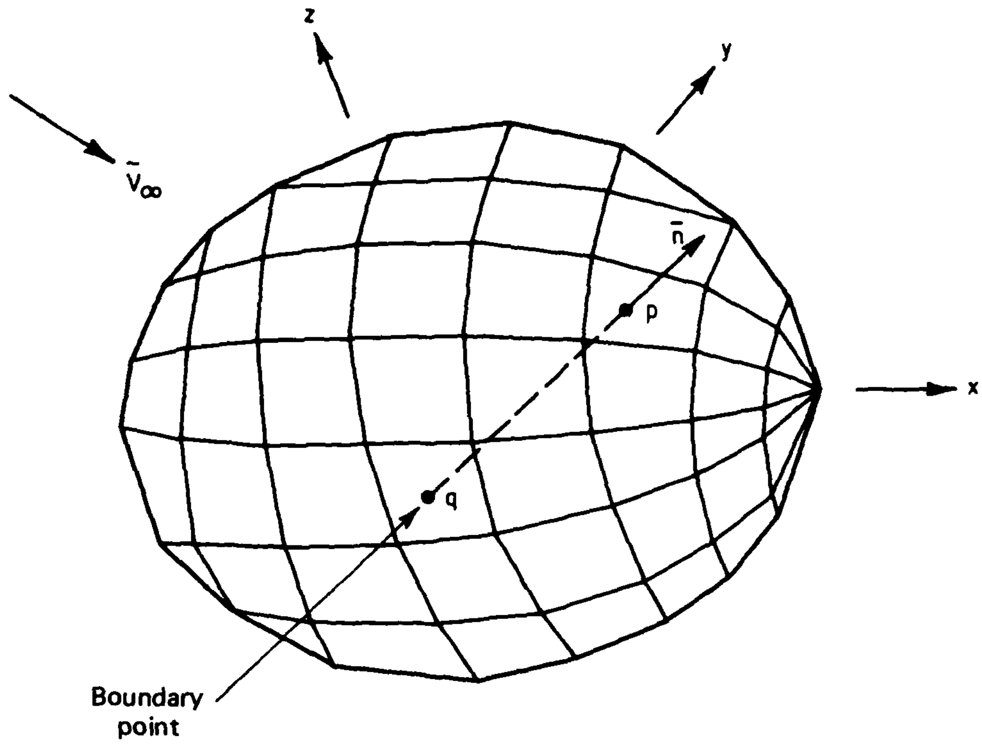



FIG. 2. APPROXIMATION OF BODY SURFACE BY SOURCE ELEMENTS

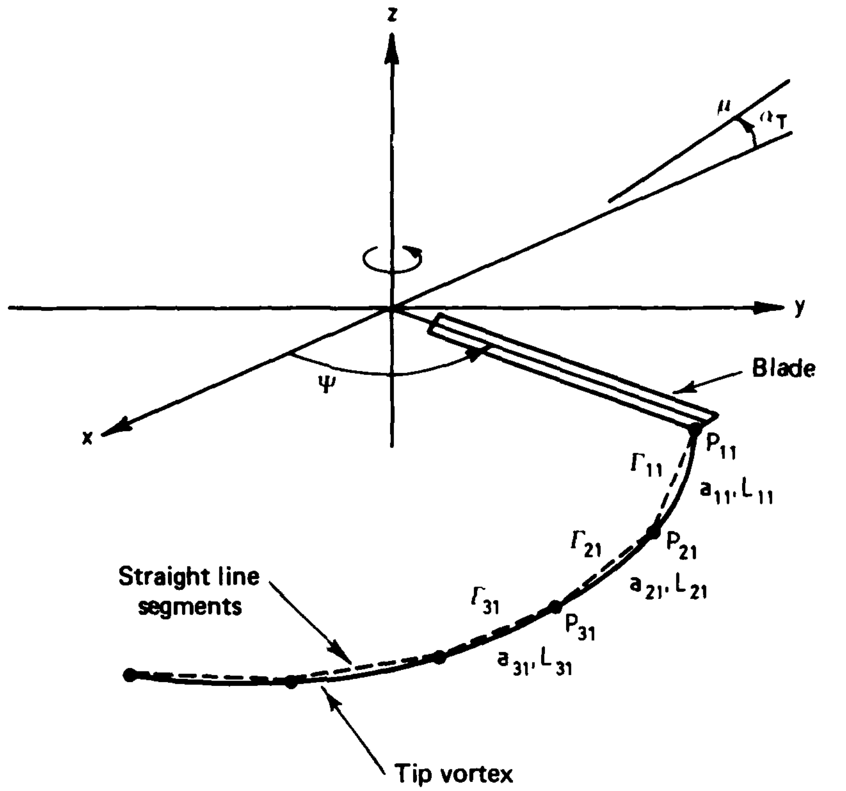



FIG. 3. A TYPICAL TIP VORTEX GENERATED BY A ROTATING BLADE

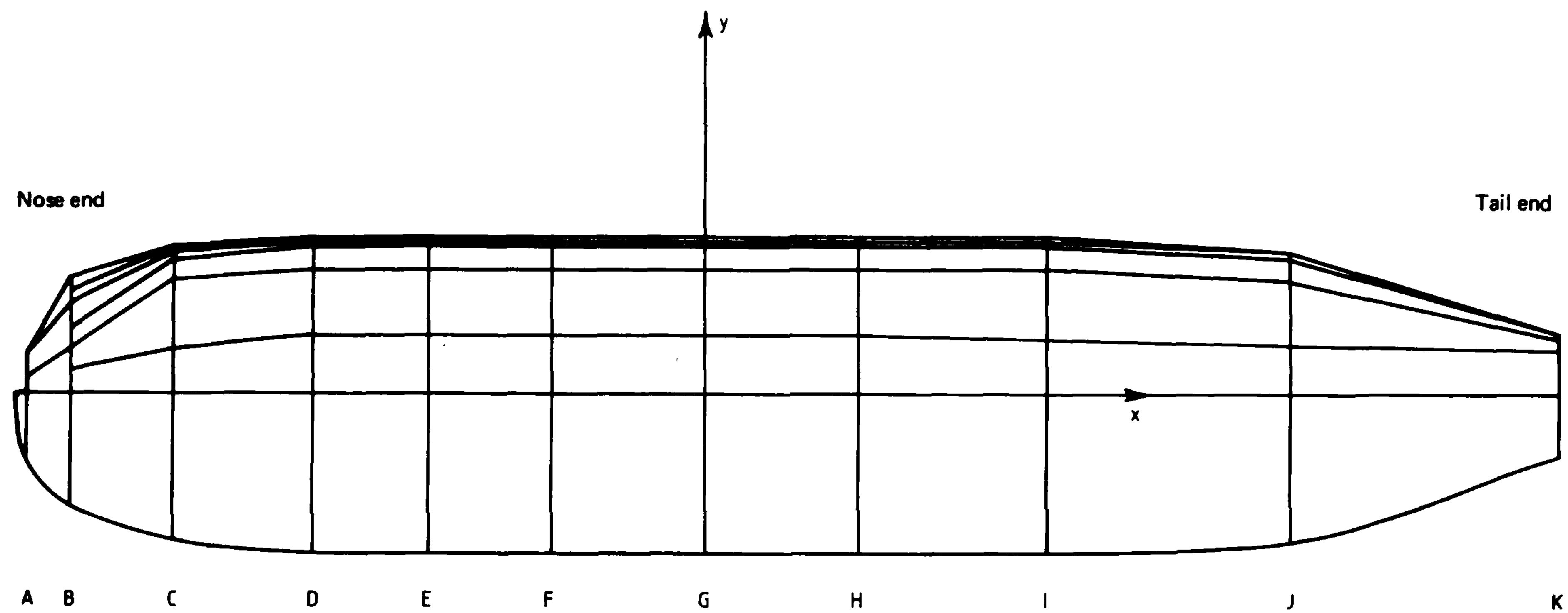



FIG. 4(a). PANEL REPRESENTATION OF A SEA KING FUSELAGE – PLAN VIEW

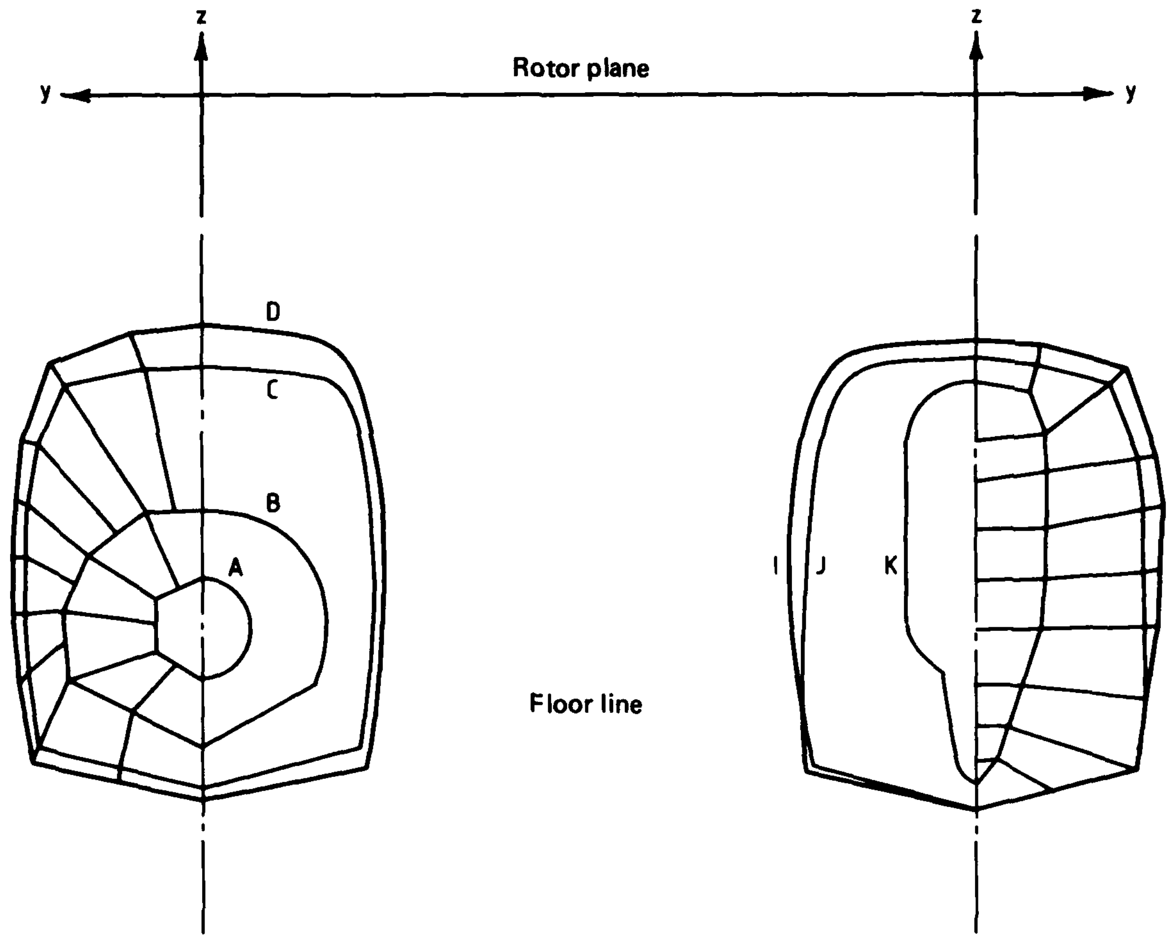



FIG. 4(b). PANEL REPRESENTATION OF A SEA KING FUSELAGE – END VIEWS

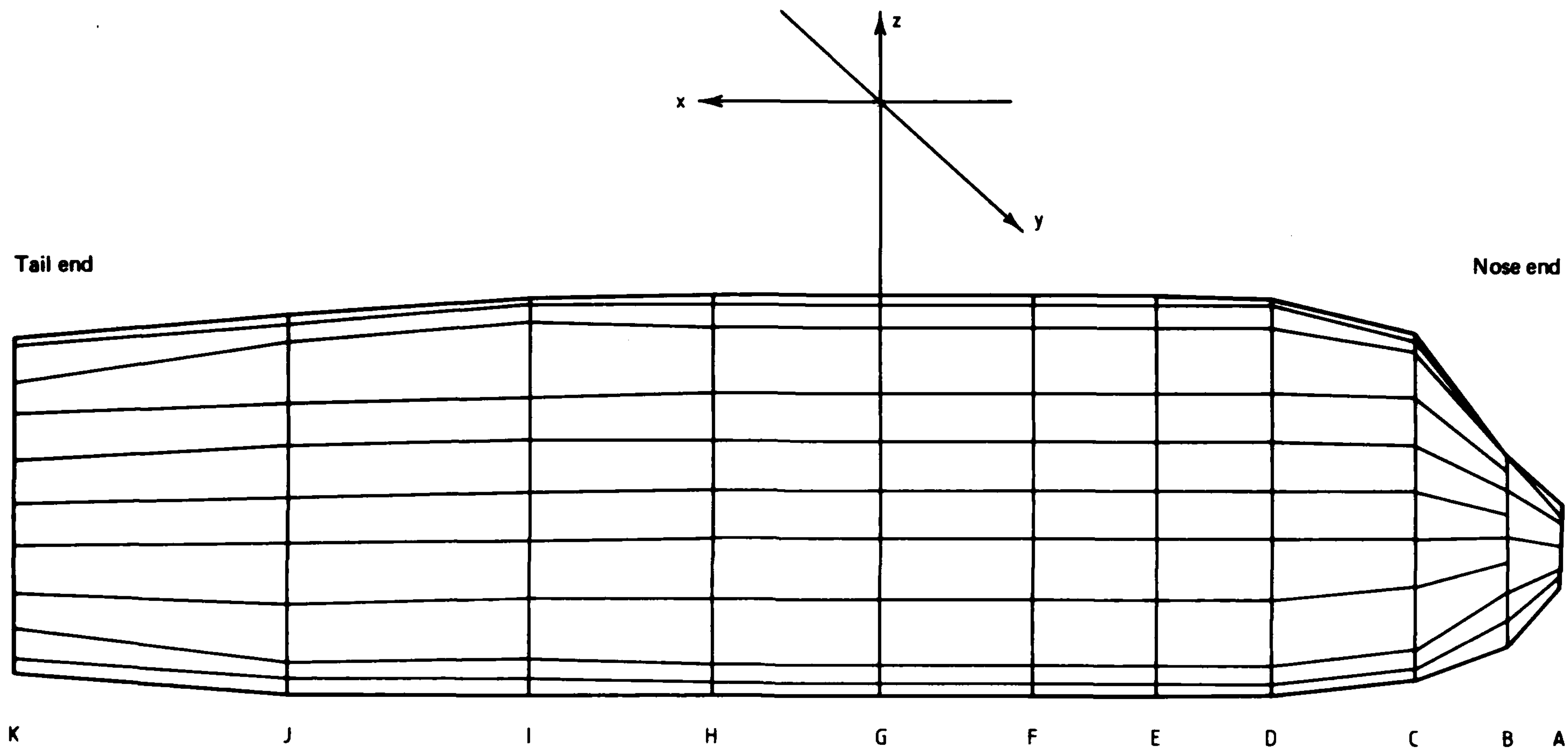
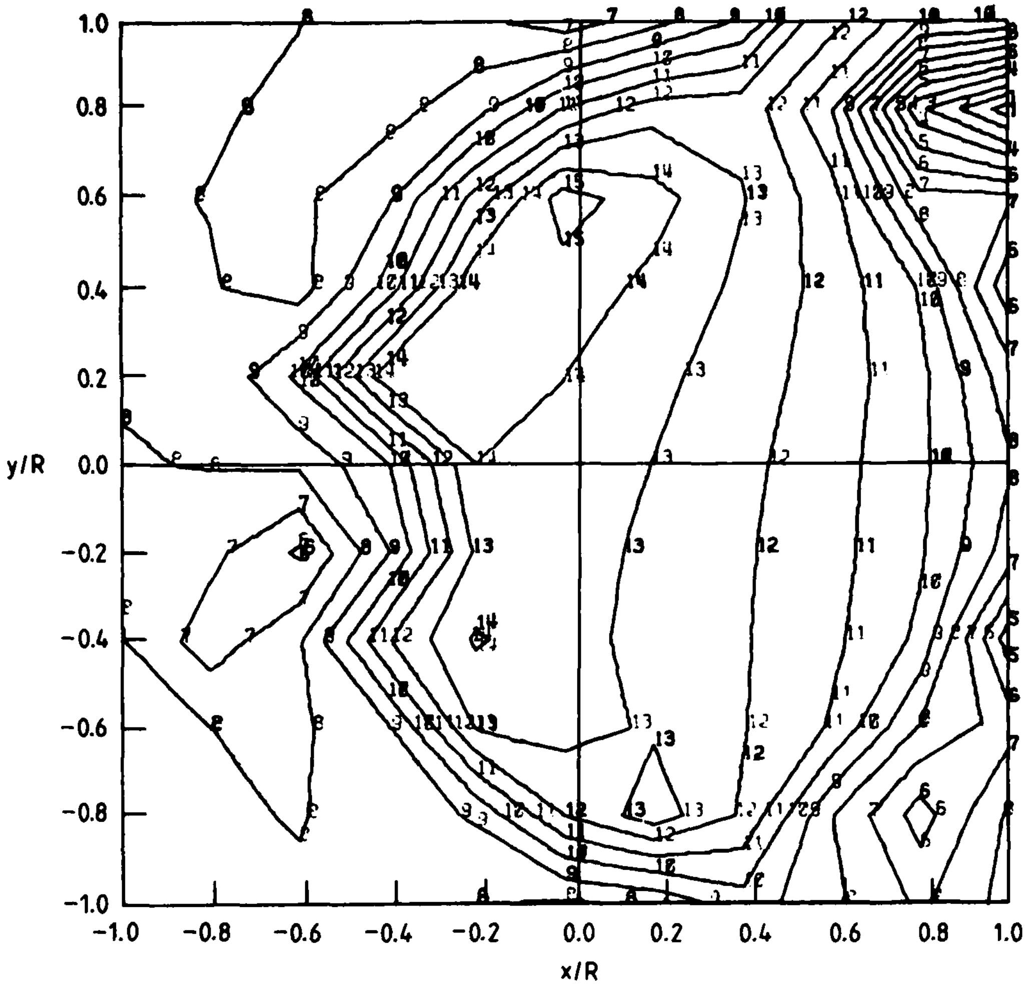
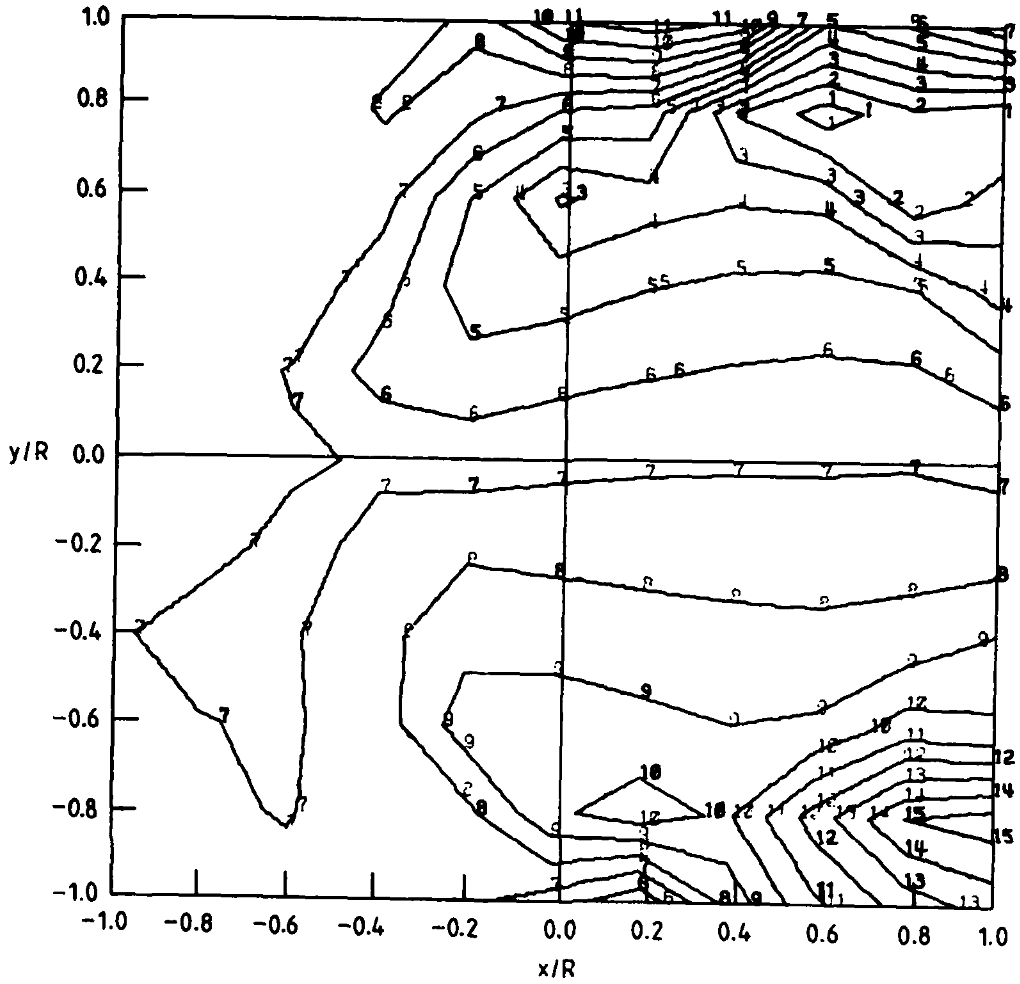



FIG. 4(c). PANEL REPRESENTATION OF A SEA KING FUSELAGE – SIDE VIEW



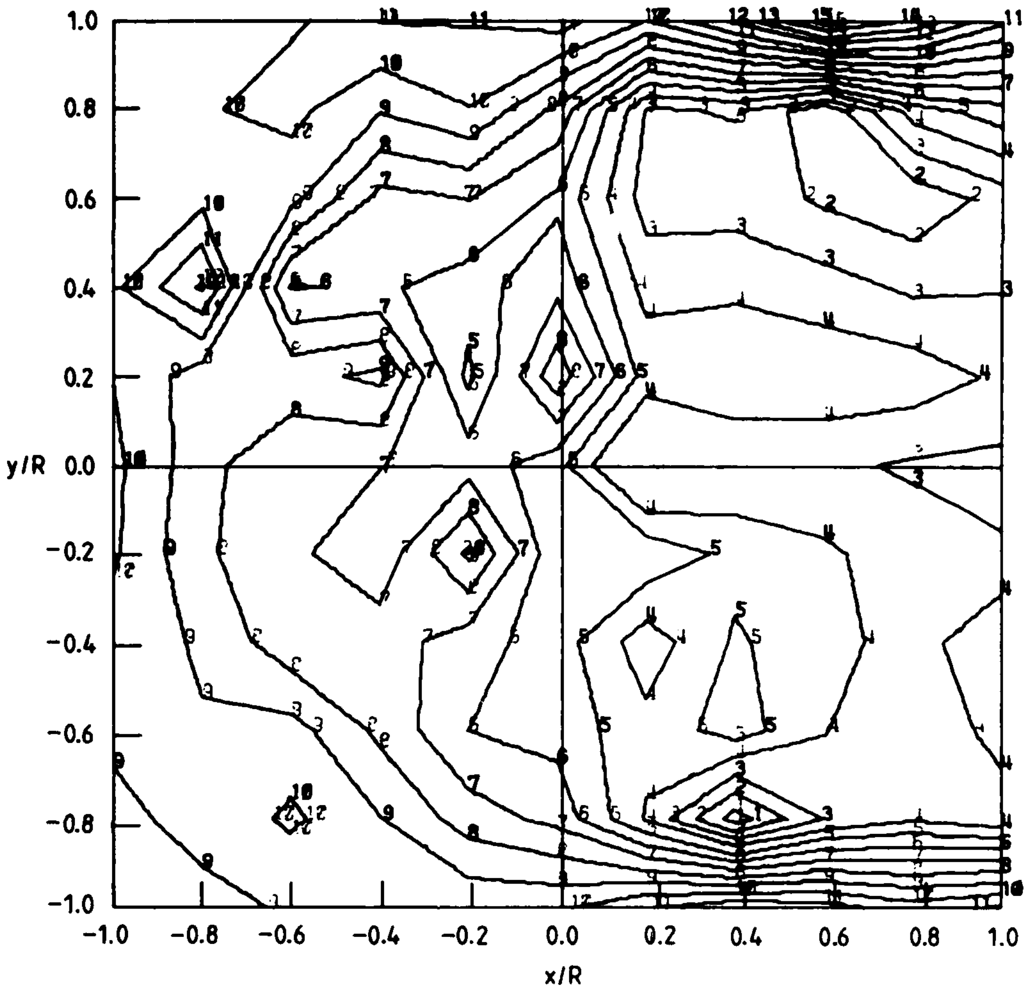

| Contour Number | Nondimensionalized velocity ($V_x/\Omega R$) |
|---|---|
| 1 | 0.06742 |
| 2 | 0.07204 |
| 3 | 0.07665 |
| 4 | 0.08127 |
| 5 | 0.08588 |
| 6 | 0.09049 |
| 7 | 0.09511 |
| 8 | 0.09972 |
| 9 | 0.10434 |
| 10 | 0.10895 |
| 11 | 0.11356 |
| 12 | 0.11818 |
| 13 | 0.12279 |
| 14 | 0.12741 |
| 15 | 0.13202 |

FIG. 5(a). CONTOURS OF x-COMPONENT OF VELOCITY IN THE PLANE OF THE ROTOR FOR FORWARD FLIGHT ($z/R = 0.0, \mu = 0.1$)



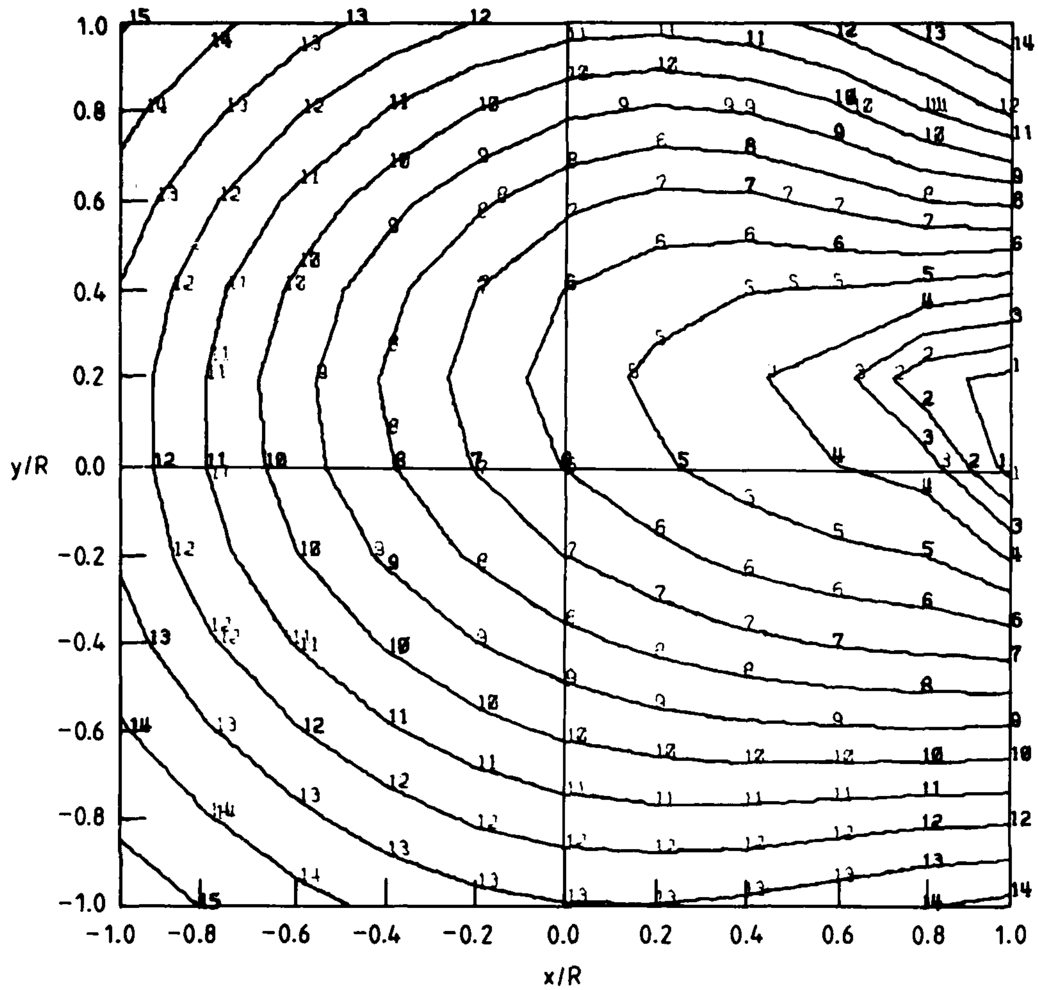
| Contour Number | Nondimensionalized velocity ($V_y/\Omega R$) |
|---|---|
| 1 | −0.03267 |
| 2 | −0.02728 |
| 3 | −0.02190 |
| 4 | −0.01651 |
| 5 | −0.01112 |
| 6 | −0.00574 |
| 7 | −0.00035 |
| 8 | 0.00504 |
| 9 | 0.01043 |
| 10 | 0.01581 |
| 11 | 0.02120 |
| 12 | 0.02659 |
| 13 | 0.03197 |
| 14 | 0.03736 |
| 15 | 0.04275 |

FIG. 5(b). CONTOURS OF y-COMPONENT OF VELOCITY IN THE PLANE OF THE ROTOR FOR FORWARD FLIGHT ($z/R = 0.0, \mu = 0.1$)



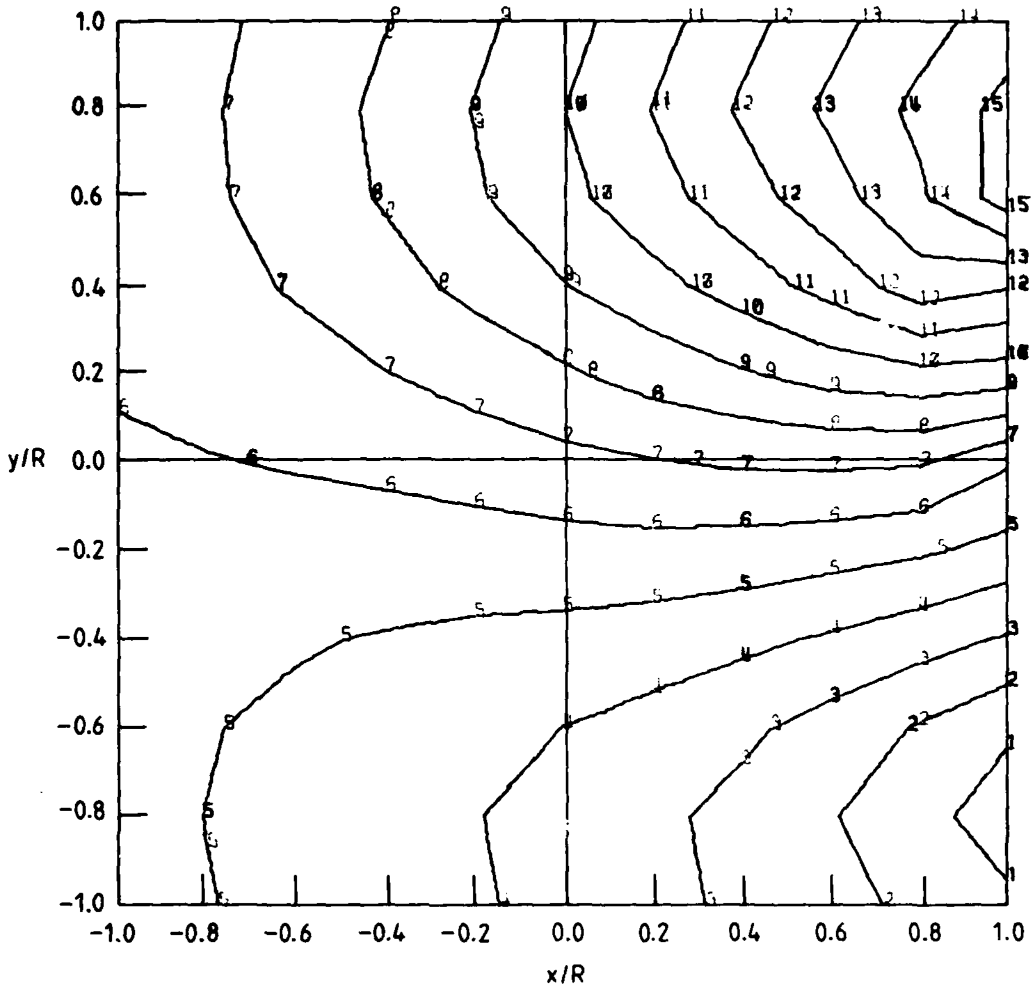
| Contour Number | Nondimensionalized velocity ($V_z/\Omega R$) |
|---|---|
| 1 | −0.09206 |
| 2 | −0.07966 |
| 3 | −0.06726 |
| 4 | −0.05486 |
| 5 | −0.04246 |
| 6 | −0.03006 |
| 7 | −0.01766 |
| 8 | −0.00526 |
| 9 | 0.00714 |
| 10 | 0.01954 |
| 11 | 0.03194 |
| 12 | 0.04435 |
| 13 | 0.05675 |
| 14 | 0.06915 |
| 15 | 0.08155 |

FIG. 5(c). CONTOURS OF z-COMPONENT OF VELOCITY IN THE PLANE OF THE ROTOR FOR FORWARD FLIGHT ($z/R = 0.0, \mu = 0.1$)



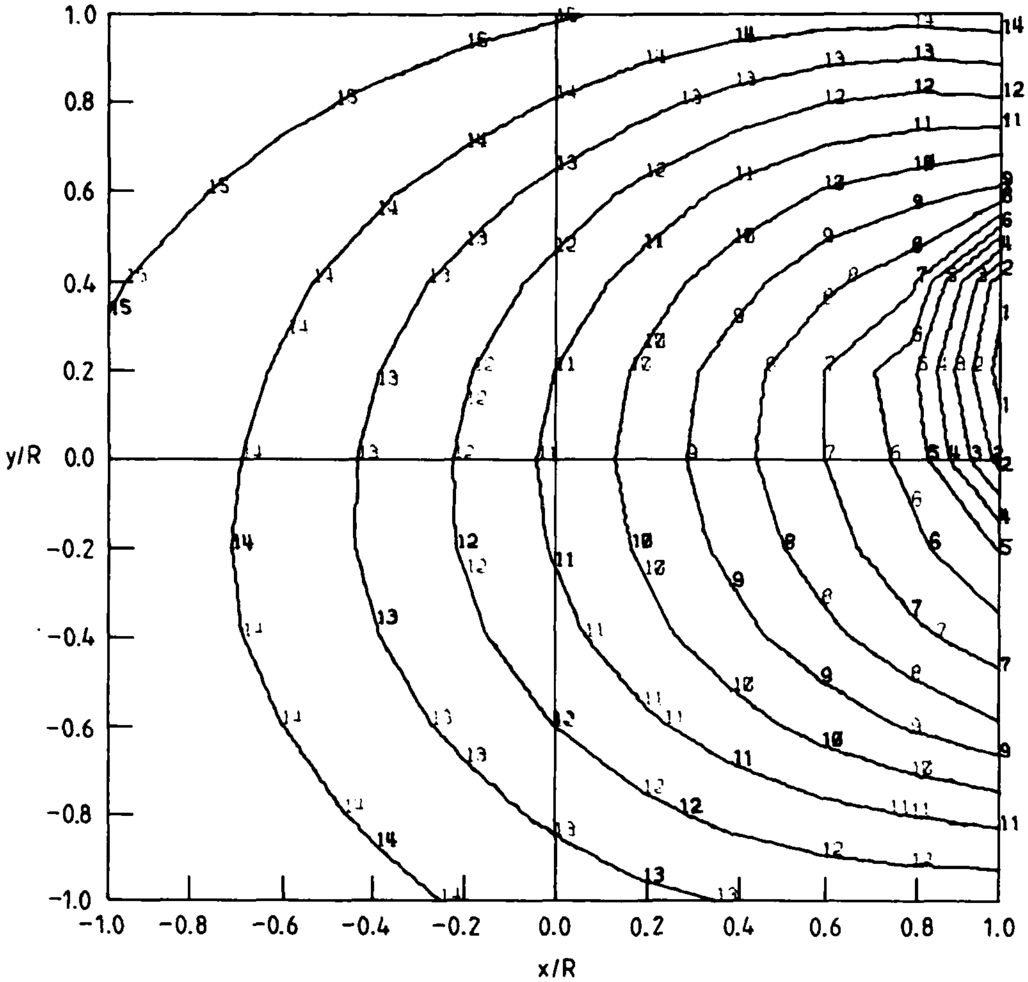

| Number | Contour Nondimensionalized velocity ($V_x/\Omega R$) |
|---|---|
| 1 | 0.07664 |
| 2 | 0.07804 |
| 3 | 0.07944 |
| 4 | 0.08084 |
| 5 | 0.08223 |
| 6 | 0.08363 |
| 7 | 0.08503 |
| 8 | 0.08643 |
| 9 | 0.08783 |
| 10 | 0.08922 |
| 11 | 0.09062 |
| 12 | 0.09202 |
| 13 | 0.09342 |
| 14 | 0.09481 |
| 15 | 0.09621 |

FIG. 6(a). CONTOURS OF x-COMPONENT OF VELOCITY IN A PLANE BELOW THE FUSELAGE FOR FORWARD FLIGHT ($z/R = -0.6452, \mu = 0.1$)




| Contour Number | Nondimensionalized velocity $(V_y/\Omega R)$ |
|---|---|
| 1 | −0.01650 |
| 2 | −0.01282 |
| 3 | −0.00914 |
| 4 | −0.00546 |
| 5 | −0.00178 |
| 6 | 0.00189 |
| 7 | 0.00557 |
| 8 | 0.00925 |
| 9 | 0.01293 |
| 10 | 0.01661 |
| 11 | 0.02028 |
| 12 | 0.02396 |
| 13 | 0.02764 |
| 14 | 0.03132 |
| 15 | 0.03500 |

FIG. 6(b). CONTOURS OF y-COMPONENT OF VELOCITY IN A PLANE BELOW THE FUSELAGE FOR FORWARD FLIGHT ($z/R = -0.6452$, $\mu = 0.1$).




| Contour Number | Nondimensionalized velocity ($V_z/\Omega R$) |
|---|---|
| 1 | −0.04880 |
| 2 | −0.04523 |
| 3 | −0.04166 |
| 4 | −0.03809 |
| 5 | −0.03452 |
| 6 | −0.03095 |
| 7 | −0.02738 |
| 8 | −0.02380 |
| 9 | −0.02023 |
| 10 | −0.01666 |
| 11 | −0.01309 |
| 12 | −0.00952 |
| 13 | −0.00595 |
| 14 | −0.00238 |
| 15 | 0.00119 |

FIG. 6(c). CONTOURS OF z-COMPONENT OF VELOCITY IN A PLANE BELOW THE FUSELAGE FOR FORWARD FLIGHT ($z/R = -0.6452$, $\mu = 0.1$)

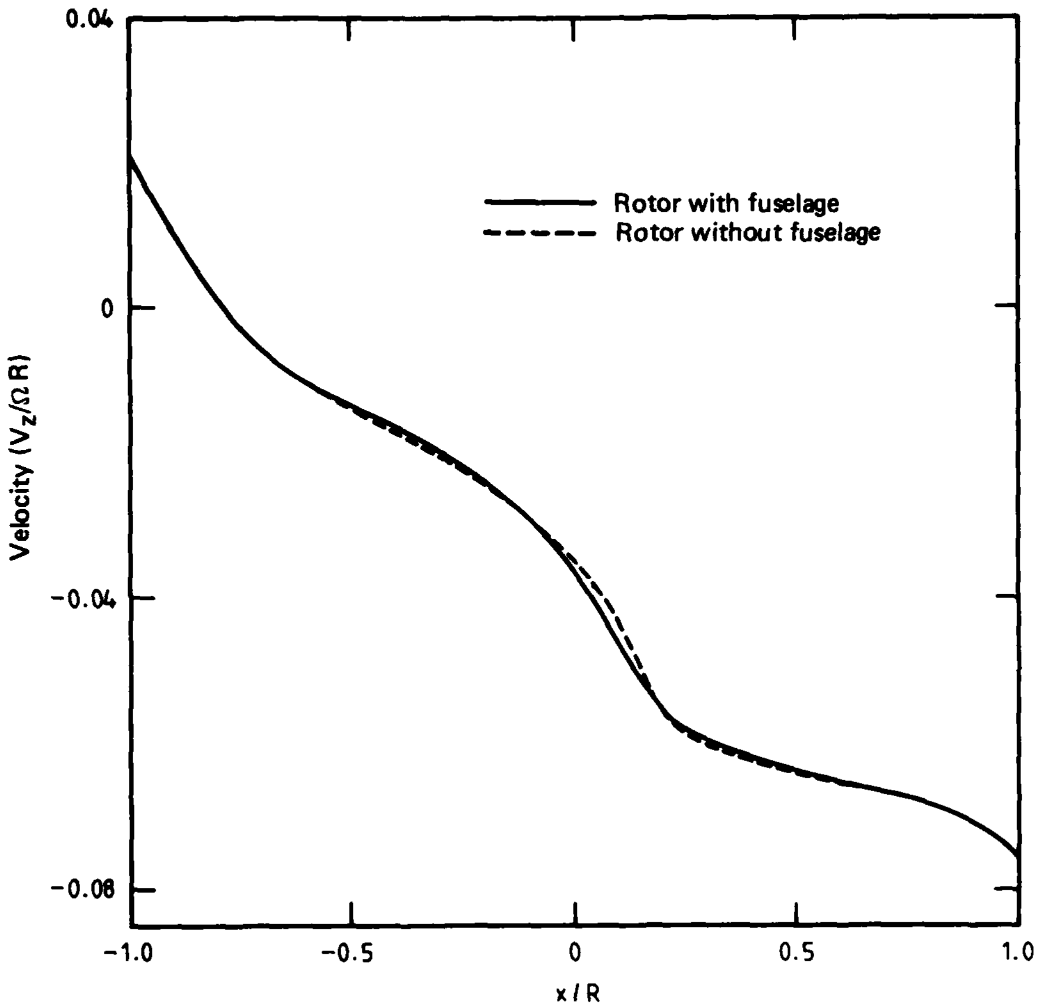



FIG. 7(a). EFFECT OF THE FUSELAGE ON DOWNWASH DISTRIBUTION IN THE PLANE OF THE ROTOR ($y/_R$ = 0.0, $z/_R$ = 0.0, $\mu$ = 0.1).

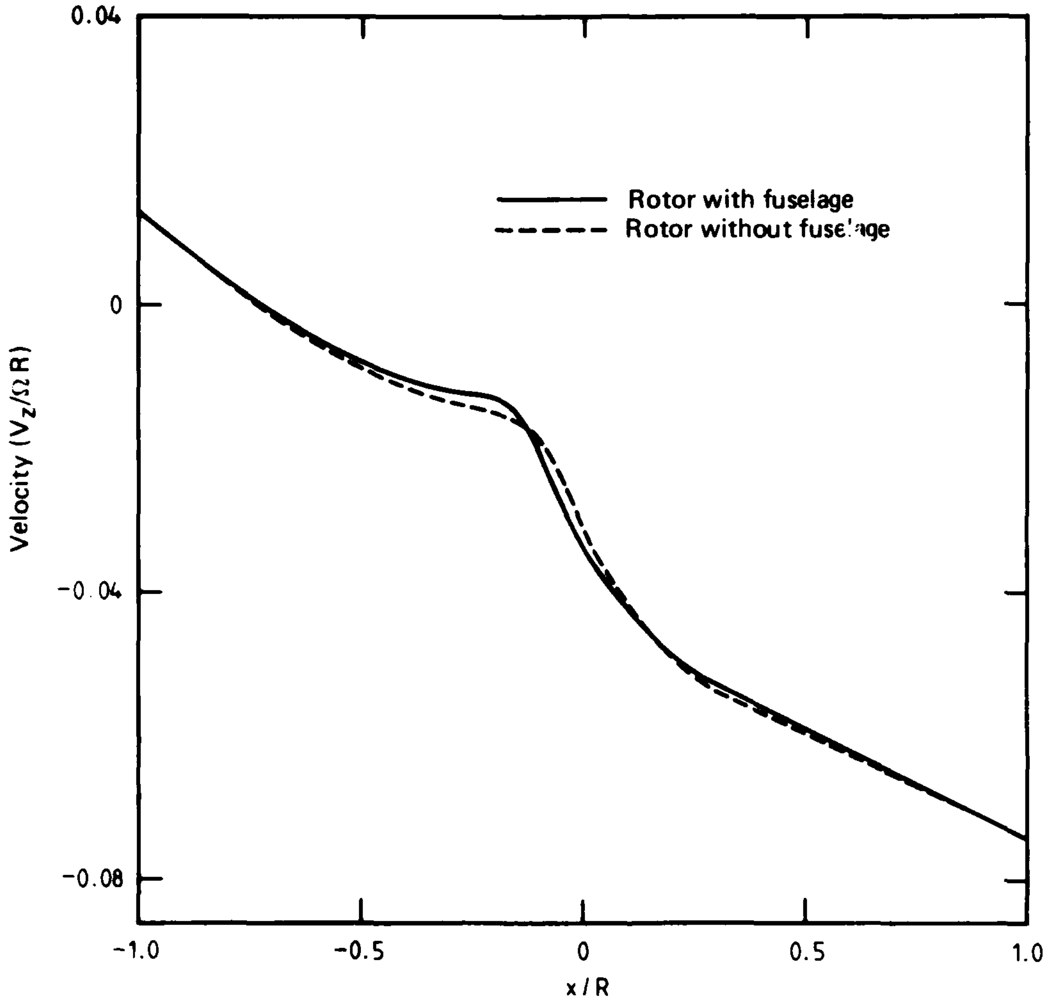



FIG. 7(b). EFFECT OF THE FUSELAGE ON DOWNWASH DISTRIBUTION IN A PLANE BELOW THE ROTOR ($y/_R$ = 0.0, $z/_R$ = −0.1, $\mu$ = 0.1)

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16. Abstract

*A mathematical model of the rotor wake in the vicinity of a rotorcraft fuselage is formulated taking into account the mutual interference between the rotor and fuselage. It is assumed that the vortex sheet behind each rotor blade rolls-up into a single, blade tip vortex and that the fuselage can be represented by constant source potential flow panels. The velocity field given by the combined rotor wake and fuselage model is presented for the case of a Sea King helicopter in forward flight.*

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