THE APPLICATION OF GENERAL MODEL OF MOVING OBJECT FOR TILTROTOR STABILITY ANALYSIS

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A computer model of a tilt-rotor has been developed for calculating performance, simulating flight, and investigating stability and control. The model is composed of a fuselage, wings, an empennage, engine nacelles and rotors. Tiltrotor equations of motion were obtained by summing up inertia, gravity and aerodynamic loads acting on each part of the aircraft. Aerodynamic loads at wings, empennage and rotor blades are calculated using quasisteady model. For rotor induced velocity the Glauert model is used. The influence of rotor inflow wing and empennage aerodynamic loads is calculated using actual value of induced velocity. The computer program of tilt-rotor model is developed in MatLab environment. The sub-programs for load calculation are supplemented by modules for calculation of trim states and stability and control matrices.

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NOMENCLATURE

Indexes

a	–	aerodynamic,
b	–	inertia,
c	–	related to three-dimensional body (fuselage, nacelle),
f	–	fuselage,
g, G	–	gravity,
h	–	horizontal stabilizer,
i	–	inertia,
$I=1$	–	element (wing, nacelle, rotor) on right side of aircraft,
$I=2$	–	element (wing, nacelle, rotor) on left side of aircraft,
$J=1, 2, 3$	–	rotor blade index of i-th rotor,
m	–	aerodynamic moment, moving element, mass,
n	–	nacelle,
p	–	aircraft, fuselage,
r	–	rotor.

Matrices and Vectors

$\mathbf{A}$	–	rotation matrix of moving element, $\mathbf{A} = \mathbf{A}(\phi, \varphi, \gamma)$,
$\mathbf{A}_G$	–	aircraft rotation matrix,
$\mathbf{A}_r$	–	transformation matrix of coordinate system in airfoil,
$\mathbf{A}_V$	–	velocity matrix,
$\mathbf{C}$	–	control vector, $\mathbf{C} = [\tau, \delta_w, \delta_h, \delta_v, \Theta]T$,
$\mathbf{I}_x$	–	inertia matrix of x-th element,
$k_{()}$	–	coefficients of inflow due rotors and wings,
$\mathbf{Q}_{xy}$	–	y-th loads of x-th element, $\mathbf{Q}_{xy} = [\mathbf{F}_{xy}, \mathbf{M}_{xy}]^T$,
$\mathbf{V}$	–	vector of linear speed in $O_p x_p y_p z_p$, $\mathbf{V} = [U, V, W]^T$,
$\mathbf{V}_a$	–	motion speed of section,
$\mathbf{V}_c$	–	vector of linear speed of three-dimensional body,
$\mathbf{U}$	–	induced velocity in $O_p x_p y_p z_p$,
$\mathbf{W}$	–	wind velocity in $O_P x_p y_p z_p$,
$\Delta\mathbf{W}_{()}$	–	inflow due rotors and wings in $O_P x_p y_p z_p$,
$\mathbf{X}$	–	vector of aircraft motion,

$$\mathbf{X} = \left[\mathbf{V}, ©, \mathbf{x}_g, ¦\ \right]^T =$$
$$= \left[U, V, W, P, Q, R, x_g, y_g, z_g, \Phi, \Theta, \Psi\right]^T$$

$\mathbf{Y}_x$	–	state vector of x-th element,
$\mathbf{g}$	–	vector of gravity acceleration,
$\mathbf{g}_p$	–	vector of gravity acceleration in $O_P x_p y_p z_p$,
$\mathbf{r}_{CG}$	–	position of C.G. in $O_P x_p y_p z_p$,
$\mathbf{r}_{nCG}$	–	position of C.G. in moving element coordinate system,
$\mathbf{r}_n$	–	position of moving element coordinate system in $O_P x_p y_p z_p$,
$\mathbf{x}_g$	–	translation of the aircraft relative to the ground, $\mathbf{x}_g = [x_g, y_g, z_p]^T$,
$\mathbf{\Omega}$	–	vector of angular speed in $O_P x_p y_p z_p$, $\mathbf{\Omega} = [P, Q, R]^T$,
$\mathbf{\Omega}_p$	–	velocity matrix of plane (non-moving element),

$\mathbf{\Omega}_m$ – velocity matrix of moving element,
$\mathbf{\Phi}$ – Euler's angles described in *OG* as a vector, $\mathbf{\Phi} = [\Phi, \Theta, \Psi]^T$,
$\mathbf{\Theta}$ – the control of rotor swash-plates, $\mathbf{\Theta} = [\Theta_{01}, \Theta_{02}, \Theta_{11}, \Theta_{12}, \Theta_{21}, \Theta22]^T$,
$\boldsymbol{\delta}_w$ – inclination angles of wing flaps (ailerons), $\boldsymbol{\delta}_w = [\delta_{w11}, \delta_{w12}, \delta_{w21}, \delta_{w22}]^T$,
$\boldsymbol{\delta}_h$ – inclination angles of elevators $\boldsymbol{\delta}_h = [\delta_{h1}, \delta_{h2}, \delta_{h3}]^T$,
$\boldsymbol{\delta}_v$ – inclination angles of rudders, $\boldsymbol{\delta}_v = [\delta_{v1}, \delta_{v2}]^T$,
$\boldsymbol{\omega}$ – angular speed of rotors, $\boldsymbol{\omega}_v = [\omega_1, \omega_2]^T$,
$\boldsymbol{\tau}$ – angle of nacelle tilt
A_r – characteristic area,
$C_x(\alpha, \delta)$ – drag coefficient (in two-dimensional flow),
$C_z(\alpha, \delta)$ – lift coefficient (in two-dimensional flow),
$C_{my}(\alpha, \delta)$ – aerodynamic pitch moment coefficient (in two-dimensional flow),
$C_{xc}(\alpha, \delta)$ – drag coefficient (in three-dimensional flow),
$C_{yc}(\alpha, \delta)$ – side force coefficient (in three-dimensional flow),
$C_{zc}(\alpha, \delta)$ – lift coefficient (in three-dimensional flow),
$C_{mxc}(\alpha, \delta)$ – aerodynamic bank moment coefficient (in three-dimensional flow),
$C_{myc}(\alpha, \delta)$ – aerodynamic pitch moment coefficient (in three-dimensional flow),
$C_{mzc}(\alpha, \delta)$ – aerodynamic yaw moment coefficient (in three-dimensional flow),
R_r – characteristic length,
V_a – the module of motion speed of section $V_a = |\mathbf{V}_a|$,
$c(y)$ – chord (characteristic dimension),
g – gravity acceleration,
m_x – mass of x-th element,
α – the angle of attack,
β – the angle of slope,
δ – the angle of inclination of steering element,
ρ – the thickness of air.

INTRODUCTION

In the recent years an increasing interest in tilt rotor technology is observed among rotorcraft community. The development of V-22 aircraft spurred a great research and development effort in tilt rotor aerodynamics [1], acoustics [2], flight control [3], aeroelasticity [4], etc. The next tilt rotor project, Bell-Agusta BA 609, initiated the tilt-rotor design effort in Europe. Recently several research European projects concerning tilt rotor technology are going on or have already been completed [5] and some new initiatives appeared. Despite the expanding interest in development and applications of tilt-rotor, there are not many papers published in general available literature presenting comprehensive approach to tiltrotor modelling and simulation [2, 6, 10]. In Warsaw University of Technology, a computer model of tiltrotor for flight simulation, and analysis of trim, stability and control in various flight conditions was developed.

GENERAL MODEL OF MOBILE OBJECTS

A tiltrotor computer model was assembled using the generic model of mobile objects, developed in Warsaw University of Technology for simplifying model derivation and simulation of motion for various air, water and land vehicles. This software environment in MatLab is composed of the main program and sub-programmes performing computations frequently done in dynamics of airplanes, helicopters, sea vessels etc.

The possibility of the general approach stems from the fact that inertia, gravity and aerodynamic/hydrodynamic loads that act on the vehicles considered are described by the same formulae in reasonably chosen coordinate systems. As a consequence, using the generic model software, computer models of various vehicles may be prepared in a fast and efficient way by proper selecting of systems of coordinates and application of supporting subroutines.

Within this software a vehicle is modelled as the base part („fuselage") to which the other parts/elements of it are joined. These parts may be fixed to the fuselage or may rotate and/or translate relative to it.

Equations of motion are derived in the main system of coordinates fixed to the fuselage by summing loads acting on all elements of a vehicle. These loads are calculated in local coordinate systems fixed to elements and then transformed to the coordinate system of the fuselage. It is easy to change the number of elements of a model by adding new or deleting existing elements and enriching the methods of calculating loads by application of various methods. The computer program for modelling a vehicle dynamics is constructed in the same, modular way, by applying subroutines for calculating each type of loads and their transformation to the main, „fuselage" system of coordinates.

Various subprograms were developed for frequently performed operations like calculating inertia matrices, flow velocities in different coordinate systems, angles of attack, slip angles, table-look procedures for aerodynamic coefficients etc.

SIMULATION SOFTWARE FOR THE GENERAL MODEL OF MOBILE OBJECTS

Tiltrotor simulations were carried out in MatLab software environment.

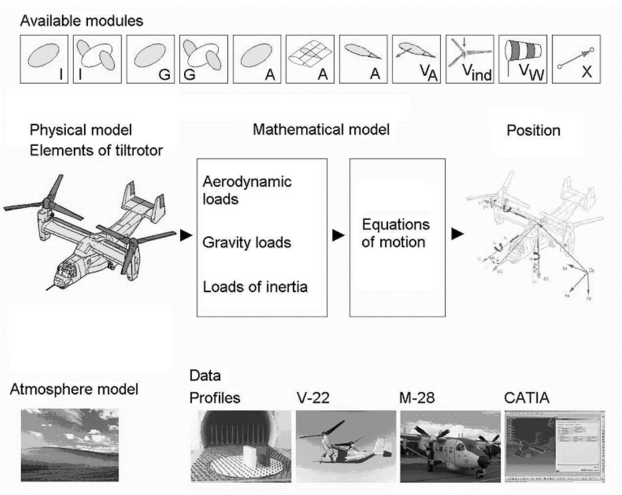

Fig. 1. The schema of simulation software

As it is illustrated in Fig. 1 the number of primary subroutines is limited to eleven (upper line icons, from left to right).

- The module **Ip_matrix** is used for calculation of inertia loads of non-movable elements (fuselage with wings and stabilisers). The input data are mass, moments of inertia in an element coordinate system, an attitude of the element in the aircraft coordinate system. The results of calculations in this module are moments of inertia in the aircraft coordinate system.
- The module **Ir_matrix** is used for the calculation of inertia loads of movable (rotating) elements (nacelles, rotor blades). The input data are mass, moments of inertia in the coordinate system of an element, its attitude and movement (rotations) in the aircraft coordinate system. Moments of inertia in the aircraft coordinate system are results of calculations in this module.
- The module **Fpg_Mpg** is used for the calculation of gravity loads of non-movable elements (fuselage with wings and stabilisers). The input data are gravity acceleration, mass of an element, its attitude in the aircraft coordinate system. Gravity loads in aircraft and element coordinate systems are results of calculations in the module.
- The module **Fng_Mng** is used for calculation of gravitaty loads of movable (rotating) elements (nacelles, rotor blades). The gravity acceleration, mass of the element, its attitude and movement (rotations) in the aircraft coordinate system are the input data. Gravity loads in aircraft and element coordinate systems are results of calculations in the module.
- The module **Fac_Mac** is used for calculation of aerodynamic loads of elements with three-dimensional airflow (fuselage, nacelles). The attitude of an element (its aerodynamic centre) in the aircraft coordinate system, the inflow on an element, the database of aerodynamic coefficients (for instance obtained from wind tunnel tests) are the input data. Aerodynamic loads in aircraft and element coordinate systems are results of calculations in the module.
- The module **Fap_Map** is used for calculation of aerodynamic loads of elements with two-dimensional airflow (wings, stabilisers, rotor blades). The attitude, movement (rotations) and the geometry of an element (dimensions, positions of wing, stabiliser, blade cross sections, aerodynamic centres in airfoils) in the aircraft and/or element coordinate system, the inflow on an element, aerodynamic coefficients are the input data. The Fap_Map module integrates aerodynamic loads calculated by the dFac_dMac module along the span of an element. Aerodynamic loads in an aircraft and an element coordinate systems are results of calculations in this module.
- The module **dFac_dMac** is used for calculation of aerodynamic loads of elements with two-dimensional airflow (wings, stabilisers, rotor blades) in their cross sections. It is executed by the Fap_Map module. Input data are shared with the Fap_Map module. Aerodynamic loads in local airfoil coordinate system are results of calculations in this module.
- The module **V_a** is used for the calculation of airflow velocity in an element cross sections with two-dimensional airflow (wings, stabilisers, rotor blades). It is executed by modules used for aerodynamic loads calculations and it shares input data with them.
- The module **ind_vel** is used for the calculation of induced velocity (Glauert method or dynamic inflow). It is executed by modules used for calculations of rotor aerodynamic loads (Fap_Map and dFac_dMac modules). It shares input data with them. Output data are transformed both to the rotor hub coordinate systems and the aircraft coordinate system.
- The module **delta_Wi** is used for the calculation of the air velocity. It is executed by modules used for aerodynamic loads calculations and it shares input data with them. Results of calculations taken place in this module are transformed to coordinate system of an airfoil local inflow.
- The module **transfer_F_M** for the calculation of the position of in main coordinate system and the transfer of data from local to main system of coordinates.

There are also some additional short modules used for often repeated calculations, e.g. vector multiplications, transformation matrices of coordinate systems.

The derivation of equations of motion takes place in the main system of coordinates fixed to the fuselage by summing loads acting on all elements of a vehicle.

Outputs of first six modules calculating inertia, gravity and aerodynamic/hydrodynamic loads acting on whole elements given in the main, „fuselage" system of coordinates are used directly in the derivation of equations of motion. Next five modules act as additional functions used by first six modules or independently as modules calculating atmosphere conditions and the attitude of tiltrotor and all its elements during the derivation of initial and actual step conditions.

The computer program allows on „modular modelling" of a vehicle dynamics. When the number of elements is modified by adding or removing some of them or by modifying methods of calculation loads it is done by adding or removing elements and transformation of their loads to the main, „fuselage" system of coordinates.

TILTROTOR MODEL

The tiltrotor model investigated in this study was based on V-22 Osprey aircraft. Due to the modular structure of the software, it is possible to adjust the model to a tilt-wing, a partly tilting wing or the other rotorcraft with different empennage, wings, nacelles, etc.

The model of tiltrotor (Fig. 2) is composed of fuselage, two wings with two trailing edge flaps at each of the wing, nacelles mounted at the wings tips, rotors mounted in front of nacelles and horizontal stabilizer with three flaps and two vertical fins with one rudder mounted on each fin. All components of the aircraft are rigid. The rotors have three blades fixed to the shaft by pitch bearing. The pitch angles of rotor blades are controlled by swash-plates for constant (collective) and harmonic (cyclic) components.

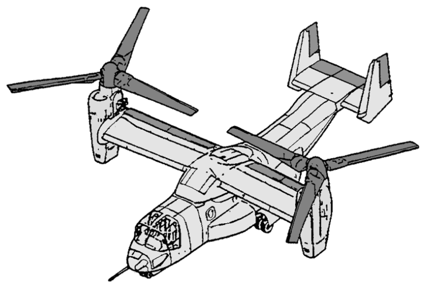

Fig. 2. Configuration of the tiltrotor

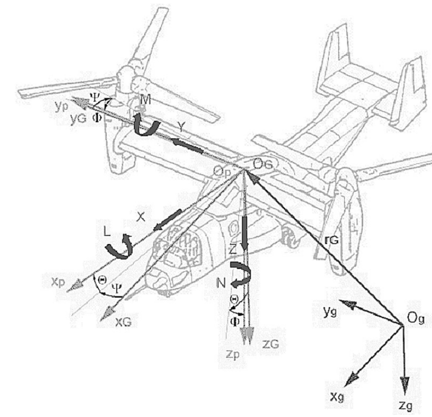

Fig. 3. Aircraft systems of coordinates

The tiltrotor equations of motion are obtained in the aircraft coordinate system $O_p x_p y_p z_p$, fixed to the fuselage (Fig. 3). The centre O_p of the aircraft system is placed in the point, where rotor shafts intersect with the fuselage plane of symmetry and this may be not a centre of gravity of the airplane. The $O_p x_p$ axis lies in the longitudinal symmetry plane of the fuselage, parallel to the horizontal reference plane of the aircraft and is directed „to the cockpit of aircraft". The $O_p z_p$ axis lies in the longitudinal plane of symmetry of fuselage and is directed, down to the undercarriage. The $O_p y_p$ axis is pointing right, while looking along the $O_p x_p$ axis.

The two other systems of coordinates important for the simulation of aircraft motion (Fig. 3) are ground coordinate system $O_g x_g y_g z_g$ fixed to the ground and gravity coordinate system $O_G x_G y_G z_G$ related to the gravity acceleration.

The centre of ground coordinate system O_g may be placed in any selected point in space (for instance: a selected point on the airfield). The aircraft motion is simulated from this point. The $O_g z_g$ axis is vertical, it coincidences with the direction of the Earth gravity acceleration and it is directed along its positive value. The axis $O_g x_g$ and $O_g y_g$ lie in the horizontal plane, i.e. the plane perpendicular to the direction of gravity acceleration. The $O_g x_g$ lies along local geographical meridian pointing north and the $O_g y_g$ axis points east.

The gravity system of coordinates is used for describing the position and the attitude of the aircraft. The axis of the gravity system are parallel to the axis of ground coordinate system. The centre O_G of gravity system of coordinates coincides with the centre of fuselage O_p. The gravity system is translated from the inertia system by the vector $\mathbf{x}_G(t) = [x_G(t), y_G(t), z_G(t)]^T$, which is the function of time.

The relation between the gravity and the aircraft system of coordinates is described by Euler angles of rotation and the relation between the coordinates in both systems is given by equation:

$$\mathbf{x}_p = \mathbf{A}_G\,(\Psi, \Theta, \Phi)\,\mathbf{x}_G \tag{1}$$

where the rotation matrix $\mathbf{A}_G$ has the form:

$$A_G = \begin{bmatrix} \cos\Psi\cos\Theta & \sin\Psi\cos\Theta & -\sin\Theta \\ \cos\Psi\sin\Theta\sin\Phi - \sin\Psi\cos\Phi & \sin\Psi\sin\Theta\sin\Phi + \cos\Psi\cos\Phi & \cos\Theta\sin\Phi \\ \cos\Psi\sin\Theta\cos\Phi + \sin\Psi\sin\Phi & \sin\Psi\sin\Theta\cos\Phi - \cos\Psi\sin\Phi & \cos\Theta\cos\Phi \end{bmatrix} \tag{2}$$

The aircraft motion is described by the vector:

$$\begin{aligned}\mathbf{X} &= \left[\mathbf{V}, \Omega, \mathbf{x}_g, \Phi\right]^T = \\ &= \left[U, V, W, P, Q, R, x_g, y_g, z_g, \Phi, \Theta, \Psi\right]^T\end{aligned} \tag{3}$$

as the composition of four vectors describing:

translation velocity $\mathbf{V} = [U, V, W]^T$,
rotation (rates) $\mathbf{\Omega} = [P, Q, R]^T$,
Euler angles written in vector form $\mathbf{\Phi} = (\Phi, \Theta, \Psi)^T$,
translation of the aircraft relative to the ground
$\mathbf{x}_g = [x_g, y_g, z_p]^T$.

EQUATION OF MOTION

The aircraft equations of motion are derived using d'Alambert principle, summing up at the point O_p all loads (forces and moments) acting on the fuselage, wings, control surfaces, nacelles, and rotors. The system of six equations of motion is obtained, which may be grouped as two subsystems for forces and moments acting on fuselage and wings (index p), two nacelles (indices $n1$, $n2$) and two rotors (indices $r1$, $r2$):

$$\mathbf{F}_p + \mathbf{F}_{n1} + \mathbf{F}_{n2} + \mathbf{F}_{r1} + \mathbf{F}_{r2} = 0 \tag{4}$$

$$\mathbf{M}_p + \mathbf{M}_{n1} + \mathbf{M}_{n2} + \mathbf{M}_{r1} + \mathbf{M}_{r2} = 0 \tag{5}$$

Each element of above equations consists of inertia, aerodynamic and gravity parts

$$\mathbf{Q}_{(\,)} = [\mathbf{F}_{(\,)}, \mathbf{M}_{(\,)}]^T = \mathbf{Q}_{(\,)i} + \mathbf{Q}_{(\,)a} + \mathbf{Q}_{(\,)g} \tag{5a}$$

INERTIA LOADS

The expression for inertia loads obtained from the conservation of momentum, which after performing some mathematical manipulations, may be written in the matrix form:

$$\mathbf{Q}_{pb} = \mathbf{I}_p \dot{\mathbf{Y}}_p + \Omega_p \mathbf{I}_p \mathbf{Y}_p \tag{6}$$

where: tilt-rotor state vector $\mathbf{Y}_p = [U, V, W, P, Q, R]^T$ is composed of components of aircraft translation velocity and rates.

Inertia matrix has the form:

$$\mathbf{I}_p = \begin{bmatrix} m_p & 0 & 0 & 0 & S_{zp} & -S_{yp} \\ 0 & m_p & 0 & -S_{zp} & 0 & S_{xp} \\ 0 & 0 & m_p & S_{yp} & -S_{xp} & 0 \\ 0 & -S_{zp} & S_{yp} & I_{xp} & -I_{xyp} & -I_{xzp} \\ S_{zp} & 0 & -S_{xp} & -I_{xyp} & I_{yp} & -I_{yzp} \\ -S_{yp} & S_{xp} & 0 & -I_{xzp} & -I_{yzp} & I_{zp} \end{bmatrix} \tag{6a}$$

and the velocity matrix is:

$$\Omega_p = \begin{bmatrix} 0 & -R & Q & 0 & 0 & 0 \\ R & 0 & P & 0 & 0 & 0 \\ -Q & P & 0 & 0 & 0 & 0 \\ 0 & -W & V & 0 & -R & Q \\ W & 0 & -U & R & 0 & P \\ -V & U & 0 & -Q & P & 0 \end{bmatrix} \tag{6b}$$

The expression (6) describes inertia loads acting on the fuselage and on the parts of rotorcraft fixed to it, i.e. wings and control surfaces.

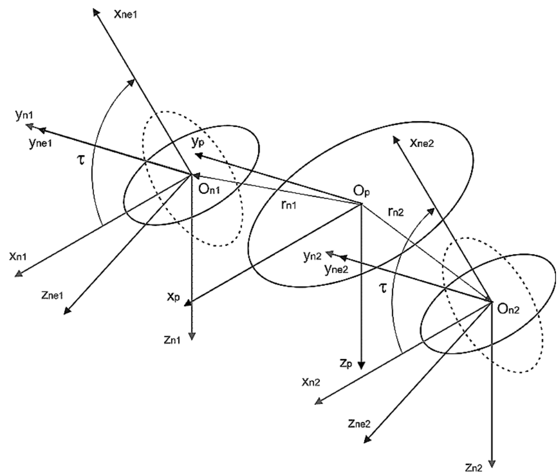

Fig. 4. Nacelle coordinate systems

For the parts of aircraft rotating relative to the fuselage, i.e. nacelles (Fig. 4) and rotors, the inertia loads are calculated as:

$$\mathbf{Q}_{mb} = \mathbf{I}_m \left(\dot{\mathbf{Y}}_p + \dot{\mathbf{Y}}_m\right) + \left(\Omega_p + \Omega_m\right)\mathbf{I}_m\left(\mathbf{Y}_p + \mathbf{Y}_m\right) \tag{7}$$

In (7) the inertia matrices of rotating elements are calculated in the fuselage system of coordinates.

Additional velocity matrices $\mathbf{\Omega}_m$ are added, containing in general case rates and velocities of these elements relative to the fuselage. In the tiltrotor case it has the form:

$$\mathbf{I}_m = \begin{bmatrix} m_m & 0 & 0 & 0 & S_{zm} & -S_{ym} \\ 0 & m_m & 0 & -S_{zm} & 0 & S_{xm} \\ 0 & 0 & m_m & S_{ym} & -S_{xm} & 0 \\ 0 & -S_{zm} & S_{ym} & I_{xm} & -I_{xym} & -I_{xzm} \\ S_{zm} & 0 & -S_{xm} & -I_{xym} & I_{ym} & -I_{yzm} \\ -S_{ym} & S_{xm} & 0 & -I_{xzm} & -I_{yzm} & I_{zm} \end{bmatrix} \tag{7a}$$

$$\Omega_m = \begin{bmatrix} 0 & -R_m & Q_m & 0 & 0 & 0 \\ R_m & P_m & -P_m & 0 & 0 & 0 \\ -Q_m & 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & -R_m & Q_m \\ 0 & 0 & 0 & R_m & 0 & -P_m \\ 0 & 0 & 0 & -Q_m & P_m & 0 \end{bmatrix} \tag{7b}$$

The nonlinear parts of equation (7) contain all accelerations acting on the rotating elements including gyroscopic effects.

GRAVITY LOADS

The gravity forces and moments are calculated in the centres of gravity of: fuselage and aircraft elements and transformed to the centre O_p of the fuselage system of coordinates. The vector of gravity acceleration in the gravity system of coordinates has the components:

$$\mathbf{g} = [0, 0, g]^T \tag{8}$$

It is rotated to the fuselage system of coordinates using transformation:

$$\mathbf{g}_p = \mathrm{A}_G\,(\Psi, \Theta, \Phi)\,\mathbf{g} \tag{9}$$

The masses of fuselage, wings and empennage are accounted for together and the gravity loads of these parts of the aircraft are calculated as fuselage gravity loads:

$$\mathbf{F}_{pg} = m_p\,\mathbf{g}_g = m_p\,\mathrm{A}_G\mathrm{g} \tag{10a}$$

$$\mathbf{M}_{pg} = \mathbf{r}_{CG} \times \mathbf{F}_{pg} = \mathbf{r}_{CG} \times m_p\,\mathbf{A}_G\mathrm{g} \tag{10b}$$

where vector $\mathbf{r}_{CG}$ describes the position of C.G. of fuselage/wings/empennage relative to O_p. The positions of C.G. of other parts of airplane in the local systems of coordinates are calculated as:

$$\mathbf{r}_{CG} = \mathbf{r}_n + \mathbf{A}(\phi, \theta, \psi)\,\mathbf{r}_{nCG} \tag{11}$$

where:

$\mathbf{r}_n$ – vector of the element C.G. relative to the fuselage centre,

$\mathbf{A}(\phi, \theta, \psi)$ – general description of the matrix of rotation of the local system of coordinates (fixed to the element) relative to the fuselage system of coordinates, it should be defined separately for each rotating element of the tilt-rotor.

$\mathbf{r}_{nCG}$ – vector of the position of the CG of the element in the local system of coordinates.

The gravity loads acting on the other aircraft elements are transferred to the fuselage system of coordinates using formulae:

$$\mathbf{F}_{ng} = m_n\,\mathbf{g}_p = m_n\,\mathrm{A}_G\mathbf{g} \tag{12a}$$

$$\begin{aligned} \mathbf{M}_{ng} &= \left[\mathbf{r}_n + \mathbf{A}(\phi,\theta,\psi)\mathbf{r}_{nCG}\right]\times\mathbf{F}_{ng} = \\ &= \left[\mathbf{r}_n + \mathbf{A}(\phi,\theta,\psi)\mathbf{r}_{nCG}\right]\times m_n\mathbf{A}_G\mathbf{g} \end{aligned} \tag{12b}$$

AERODYNAMICS LOADS

In the generic model, two methods of calculation of quasi-steady aerodynamic loads are used: 2D model for elongated elements composed of airfoils and 3D model for solid bodies.

Two dimensional flow (Fig. 5) is used for wings, rotor blades and horizontal and vertical stabilizers. For better description some coordinate systems are introduced:

- coordinate system of the mounting point of the element $O_N x_N y_N z_N$,
- local airfoil coordinate systems $O_a x_a y_a z_a$,
- coordinate system fixed to the airfoil aerodynamic centre $O_{ac} x_{ac} y_{ac} z_{ac}$,
- coordinate system connected with geometrically twisted airfoil $O_{ag} x_{ag} y_{ag} z_{ag}$,
- coordinate system defined by the vector of motion speed of local airfoil $O_{av} x_{av} y_{av} z_{av}$,
- coordinate system of local inflow of an airfoil $O_V x_V y_V z_V$.

A transformation of mentioned above coordinate systems can be described as

$$\mathbf{X}_N = \mathbf{r}_y + \mathbf{r}_{ac} + \mathbf{A}_{ag}\,\mathbf{A}_{av}\,\mathbf{A}_{aV}\,\mathbf{x}_V = \mathbf{r}_y + \mathbf{r}_{ac} + \mathbf{A}_r\,\mathbf{x}_V$$

where $\mathbf{A}_r = \mathbf{A}_{ag}\,\mathbf{A}_{av}\,\mathbf{A}_{aV}$

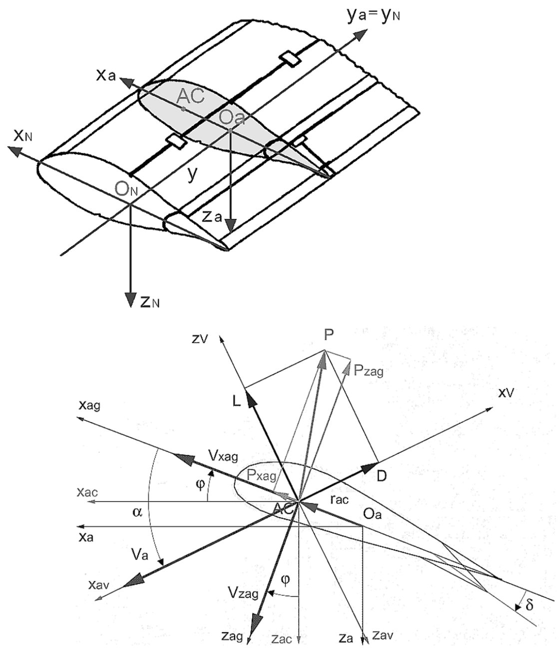

Fig. 5. Two-dimensional flow model

The elongated element is divided along the span for the cross sections in which two dimensional flow is assumed. In each cross section the instant, total flow velocity (Fig. 4) is calculated as:

$$\mathbf{V}_{(\,)i} = \mathbf{A}_{r(\,)}^{-1}\mathbf{A}^{-1}\left(\mathbf{V} + \Omega\times\mathbf{r}_{(\,)} - \mathbf{A}_G\mathbf{W} + \mathbf{U}_{(\,)} + \Delta\mathbf{W}_{(\,)}\right) \quad (13)$$

taking into account velocities of:

a) motion of the fuselage and the aircraft elements in inertia coordinate system,
b) motion of the air **W** due to wind, gusts, etc,
c) inflow due rotors and wings.

$\mathbf{A}_{r(\,)}$ matrix is a substitute matrix describing rotations of coordinate systems in airfoil (Fig. 5) from the system of the mounting point of the element to the system of local velocity inflow ($\mathbf{A}_r = \mathbf{A}_{ag}\,\mathbf{A}_{av}\,\mathbf{A}_{aV}$). Inflow due rotors and wings $\Delta\mathbf{W}_{(\,)}$ is calculated as:

$$\Delta\mathbf{W}_{()} = k_{()}\left(\mathbf{V} + \Omega\times\mathbf{r}_{()} - \mathbf{A}_G\mathbf{W} + \mathbf{U}_{()}\right) \quad (14)$$

where $k_{(\,)}$ - coefficient assumed for given element.

On rotor blades, rotor induced velocity $\mathbf{U}_{(\,)}$ obtained from Glauert model is also taken into account calculating flow velocity. Aerodynamic loads, assumed acting in the „aerodynamic centre" (AC) of the cross sections are calculated using aerodynamic coefficients for the airfoil section obtained for the actual airfoil angle of attack α and deflection δ of flaps (if they exist) by a table look procedure. In the cross sections, vectors of the aerodynamic forces and moments are calculated in the flow coordinate system as:

$$d\mathbf{P} = [dD, 0, dL]^T, \quad d\mathbf{M} = [0, dM, 0]^T \quad (15a)$$

where

$$\text{drag} \qquad dD = \frac{1}{2}\rho c(y) V_a^2 C_x(\alpha,\delta)\,dy \quad (15b)$$

$$\text{lift} \qquad dL = \frac{1}{2}\rho c(y) V_a^2 C_z(\alpha,\delta)\,dy \quad (15c)$$

$$\text{moment} \qquad dM = \frac{1}{2}\rho c^2(y) V_a^2 C_{my}(\alpha,\delta)\,dy \quad (15d)$$

The loads are transferred to the element local system of coordinates, integrated along the span and transferred to the fuselage system of coordinates.

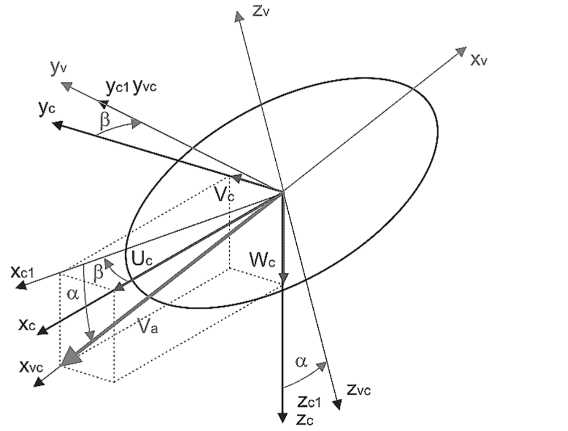

Fig. 6. Three-dimensional airflow

Fuselage and nacelles (Fig. 6) are treated as three-dimensional bodies. Aerodynamic loads, as in the 2D case, are calculated in the centre of local, flow coordinate system:

$$P_{()a} = \frac{1}{2}\rho A_r V^2 C_{()c}(\alpha,\beta) \quad (16a)$$

$$M_{()a} = \frac{1}{2}\rho A_r R_r V^2 C_{m()c}(\alpha,\beta) \quad (16b)$$

using local instant velocity $\mathbf{V}_c = [U_c, V_c, W_c]^T$ in the body aerodynamic centre. The angle of incidence α and angle of slip β are calculated as:

$$\alpha = a\sin\left[\frac{W_c}{\sqrt{U_c^2+V_c^2+W_c^2}}\right], \qquad \beta = a\sin\left[\frac{V_c}{\sqrt{U_c^2+V_c^2}}\right]. \quad (17)$$

and table look procedure is used for obtaining aerodynamic coefficients. The loads are from the flow system of coordinates are transformed to the element system of coordinates using rotation matrix

$$\mathbf{A}_V = \begin{bmatrix} -\cos\beta\cos\alpha & -\sin\beta & \cos\beta\sin\alpha \\ -\sin\beta\cos\alpha & \cos\beta & \sin\beta\sin\alpha \\ -\sin\alpha & 0 & -\cos\alpha \end{bmatrix} \quad (18)$$

Finally, as in 2D case, the loads are transformed from element local system of coordinates to fuselage system of coordinates.

DETAILS OF MODELLING TILTROTOR PARTS

Fuselage

The inertia loads acting on fuselage are calculated from (6), gravity loads from (10) and the aerodynamic loads from (16). The inertia matrix covers fuselage, wings and empennage – elements fixed to the fuselage.

Wings

The tiltrotor wings have prescribed planform, twist and airfoil distribution along the span. At each wing there are two flaps (ailerons) controlled individually. Flap deflection angles $\boldsymbol{\delta}_w = [\delta_{w11}, \delta_{w12}, \delta_{w21}, \delta_{w22}]^T$ form part of control variables in the simulation of aircraft motion. The aerodynamic loads are calculated using 2D model. Induced velocities of the rotors are included into velocity of wing airflow calculated in the sections along the span.

Horizontal stabilizer

The horizontal stabilizer forms part of the empennage and is mounted at the fuselage tail. There are three flaps (elevators) at the trailing edge of the stabiliser, controlled individually. The horizontal stabilizer may have arbitrary airfoil shape and twist angle distribution along the span. The aerodynamic loads are calculated using 2D model. Influence of the induced velocity of the rotors is taken into account in the stabilizer sections along the stabilizer span with the time delay due to the distance travelled from rotor to empennage. The elevator deflection angles, written as a vector $\boldsymbol{\delta}_h = [\delta_{h1}, \delta_{h2}, \delta_{h3}]^T$ are part of the vector of control variables.

Vertical stabilizers

The vertical stabilizers are mounted at the tips of horizontal stabilizer. There is one flap (rudder) at the trailing edge of each of the two stabilizers, controlled individually. The vertical stabilizers may have arbitrary airfoil shape and twist angle distributions along the span. The aerodynamic loads are calculated using 2D model. Influence of the induced velocity of the rotors is taken into account in the proper sections along the stabilizer span with the time delay due to the distance travelled from rotor to empennage. The rudder deflection angles $\boldsymbol{\delta}_v = [\delta_{v1}, \delta_{v2}]^T$ are part of the vector of control variables.

Engine nacelles

The engine nacelles are placed at the tip of each wing. They may rotate about axis perpendicular to the fuselage plane of symmetry. The nacelles inertia loads are calculated using expression (7) and gravity loads using expression (12). Aerodynamic loads are calculated using 3D model. The velocity of the airflow around the nacelles contains inflow from rotors and component from rate due to rotation of nacelles about fuselage axis. The angle of nacelle rotation τ is included into the set of control variables.

Rotors

The axis of rotors rotation are perpendicular to the axis of nacelle rotation relative to the fuselage. When the rotor axis are in the horizontal position, the right rotor rotates clockwise, and the left counter clockwise looking from the rear of fuselage. For inertia loads calculations, rotors are treated as rotating discs and in hub system of coordinates the inertia matrices have the form:

$$\mathbf{I}_{ri} = \begin{bmatrix} m_{ri} & 0 & 0 & 0 & 0 & 0 \\ 0 & m_{ri} & 0 & 0 & 0 & 0 \\ 0 & 0 & m_{ri} & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 & I_{zri} \end{bmatrix} \tag{19}$$

Final form of inertia matrices of rotors (7a) results from nacelle and rotors rotation about fuselage axis.

Each rotor has three blades mounted to the shaft by pitch bearings. The pitch of the rotor blades is controlled by the swash-plate, resulting in collective and periodic control in the form:

$$\theta_{ij} = \theta_{0i} + \theta_{1i} \cos\left(\Omega_i t + \frac{2\pi}{3} j\right) + \theta_{2i} \sin\left(\Omega_i t + \frac{2\pi}{3} j\right) \tag{20}$$

The blades may have arbitrary planform, twist and airfoil distribution along the span. The aerodynamic loads are calculated using strip theory with quasisteady aerodynamic loads using table-look procedure for airfoils aerodynamic coefficient calculations (2D model described above). The induced flow is calculated using Glauert model. The control of rotor swash-plates may be written in the vector form $\boldsymbol{\Theta} = [\Theta_{01}, \Theta_{02}, \Theta_{11}, \Theta_{12}, \Theta_{21}, \Theta_{22}]^T$ containing the sub-set of tiltrotor control variables.

The control vector of tiltrotor:

$$\mathbf{C} = \left[\tau, \delta_w, \delta_h, \delta_v, \Theta\right]^T \tag{21}$$

consists of (respectively) tilt angle of nacelles, angles of deflection of ailerons, elevators and rudders, pitch control of rotors, which is 16 control states. Rotor angular velocity is assumed constant.

CALCULATION OF TRIM IN STEADY FLIGHT

Including inertia loads into general form of equations of motion, the tiltrotor equations of motion have the form:

$$\begin{aligned} &\mathbf{I}_p \dot{\mathbf{Y}}_p + \Omega_p \mathbf{I}_p \mathbf{Y}_p + \sum_{i=1}^{2} \left(\mathbf{I}_{ni} \dot{\mathbf{Y}}_{ni} + \Omega_{ni} \mathbf{I}_{ni} \mathbf{Y}_{ni}\right) + \\ &\sum_{i=1}^{2} \left(\mathbf{I}_{ri} \dot{\mathbf{Y}}_{ri} + \Omega_{ri} \mathbf{I}_{ri} \mathbf{Y}_{ri}\right) = \\ &= \mathbf{Q}_{pg} + \sum_{i=1}^{2} \mathbf{Q}_{nig} + \sum_{i=1}^{2} \mathbf{Q}_{rig} + \mathbf{Q}_{pa} + \sum_{i=1}^{2} \mathbf{Q}_{nia} + \sum_{i=1}^{2} \mathbf{Q}_{ria} \end{aligned} \tag{22}$$

In the steady flight the accelerations are zero, and the equations of motion are reduced to system of 6 algebraic equations:

$$\begin{aligned} \mathbf{Q} = &-\Omega_p \mathbf{I}_p \mathbf{Y}_p - \sum_{i=1}^{2} \left(\mathbf{I}_{ni} \dot{\mathbf{Y}}_{ni} + \Omega_{ni} \mathbf{I}_{ni} \mathbf{Y}_{ni}\right) - \\ &- \sum_{i=1}^{2} \left(\mathbf{I}_{ri} \dot{\mathbf{Y}}_{ri} + \Omega_{ri} \mathbf{I}_{ri} \mathbf{Y}_{ri}\right) + \mathbf{Q}_{pg} + \sum_{i=1}^{2} \mathbf{Q}_{nig} + \\ &+ \sum_{i=1}^{2} \mathbf{Q}_{rig} + \mathbf{Q}_{pa} + \sum_{i=1}^{2} \mathbf{Q}_{nia} + \sum_{i=1}^{2} \mathbf{Q}_{ria} = \mathbf{0} \end{aligned} \tag{23}$$

The equations of motion have the general form:

$$\mathbf{Q}\left(\mathbf{V}_p, \Omega_p, \Phi, \tau, \delta_w, \delta_h, \delta_v, \Theta\right) = 0 \tag{24}$$

The values of trim controls in steady flight are calculated using Levenberg-Marquardt method to minimise the total loads (24) acting on tiltrotor. This approach [8] allows to obtain the trim states for the cases when the number of calculated trim parameters is arbitrary, i.e. less, equal or greater than the number of equations of motion.

Minimizing the nonlinear functions numerically leads to computing local minima, which may not be the real solution of the trim from physical point of view. To avoid such cases, the total loads acting on tiltrotor in steady flight were monitored. The simulations of tiltrotor flight were done using controls calculated in trim procedure to prove that parameters obtained by minimisation method were correct.

DATA FOR SIMULATION

The aim of simulation performed in this study was to check the validity of the model and to get insight into tiltrotor behaviour in the trimmed state. Data of V-22 were used in simulation. The part of the design data was taken from open literature (eg. [7, 12]). For parameters with no available data, the values were assumed as for the parts of similar aircraft [8] (Fig. 1). The base dimensions of simulated tiltrotor are shown in Fig. 7 [7] and given in Table 1.

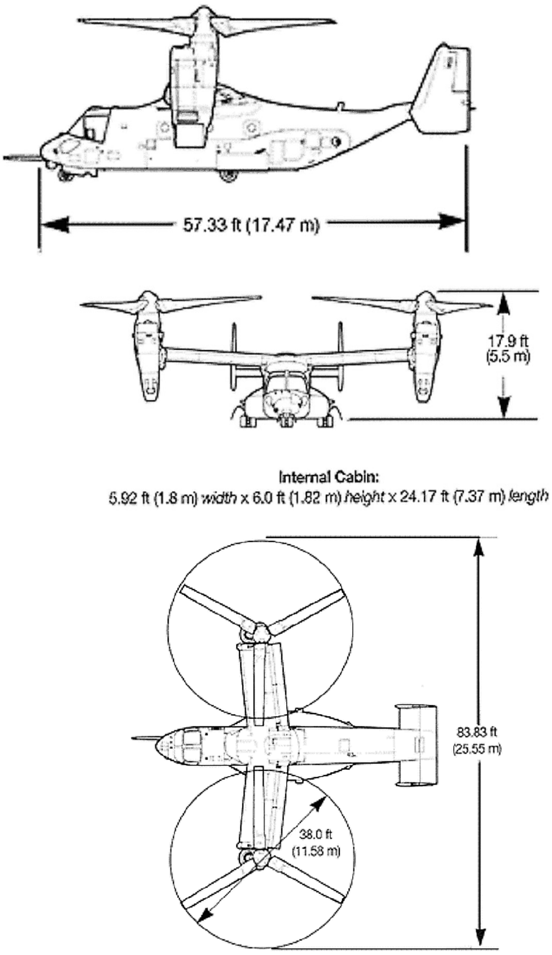

Fig. 7. Dimensions of V-22

Tab. 1. Tiltrotor data

Rotor System		
Number of blades		3
Blades tip speed	m/s (fps)	201,75 (661,90)
Diameter	m (ft)	11,58 (38,00)
Disc area	m^2, (ft^2)	210,70 (2 268,00)
Weights		
Take off	kg (lbs)	15 032 (33 140)
Dimensions		
Length, fuselage	m (ft)	1 748 (57,33)
Width, rotors turning	m (ft)	25,55 (83,33)
Width, horizontal stabilizer	m (ft)	5,61 (18,42)
Height, nacelles fully vertical	m (ft)	6,63 (21,76)
Height, vertical stabilizer	m (ft)	5,38 (17,65)

RESULTS OF SIMULATIONS

Before the simulation of the complete tiltrotor motion, separate modules and complete code were debugged Next for the tiltrotor data the results of simplified cases were checked for consistency with the proper reactions on the input. For instance, analysis of the influence of each control surfaces [8, 9] on tiltrotor flight was done for the selected flight phases [11]. The tiltrotor flight simulations were done for three tiltrotor flight modes: helicopter, airplane and conversion.

The longitudinal, symmetrical cases of flight are presented in this paper. The side V velocity, angular velocities P, Q, R roll Φ and yaw Ψ angles and the rudder deflection angles $\boldsymbol{\delta}_v = \delta_{v1} = \delta_{v2}$ were assumed zero. The same values were assumed for the deflection of wing flaps $\boldsymbol{\delta}_w = \delta_{w11}, \delta_{w12} = \delta_{w21} = \delta_{w22}$, angles of elevators $\boldsymbol{\delta}_h = \delta_{h1} = \delta_{h2} = \delta_{h3}$ and collective control of rotor pitch $\boldsymbol{\Theta}_0 = \Theta_{01} = \Theta_{02}$ for flaps on wings and elevator. The longitudinal cyclic control of rotor was assumed symmetrical and $\boldsymbol{\Theta}_1 = \Theta_{11} = \Theta_{12}$ and the lateral cyclic control unsymmetrical $\boldsymbol{\Theta}_2 = \Theta_{21} = -\Theta$.

During trim calculations for assumed values of forward U and vertical W flight velocities, the tiltrotor pitch angle Θ, deflection of flaps on wing $\boldsymbol{\delta}_w$ and elevators $\boldsymbol{\delta}_h$, collective $\boldsymbol{\Theta}_0$ and cyclic $\boldsymbol{\Theta}_1$ pitch of rotor blades and the nacelle tilt angle τ were computed.

The airplane, conversion and helicopter modes of tiltrotor flight were considered. The results of steady flight in airplane mode (horizontal with vertical climb) are given in Fig. 8÷11. The forward flight velocity was changed within the range $U = 10÷180$ m/s, and the vertical velocity was changed within the range $W = -20÷20$ m/s.

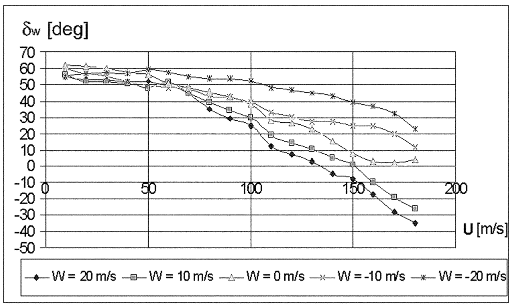

Fig. 8. Deflection of flaps

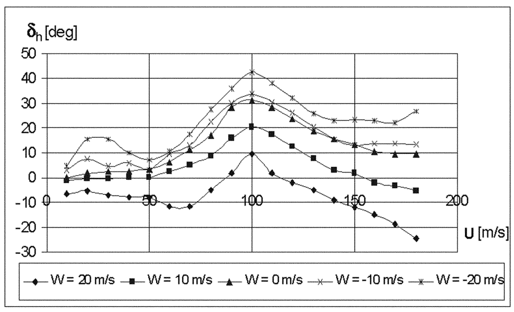

Fig. 9. Deflection of elevators

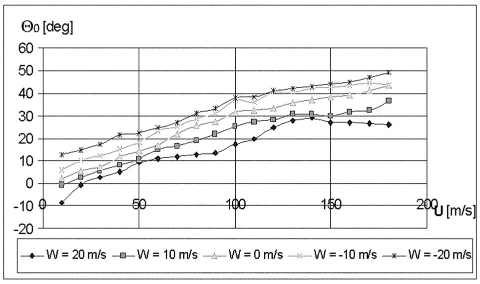

Fig. 10. Collective pitch

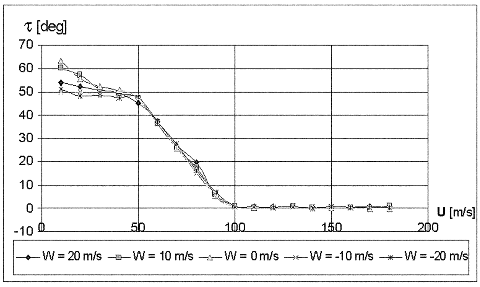

Fig. 11. Tilt angle of nacelles

The tiltrotor pitch angle obtained from simulations was equal 0 for assumed flight conditions, so it is not presented in graphs. Calculated deflection angles of wing flaps (Fig. 8) and elevators (Fig. 9) do not exceed the values of available control surface deflections of V-22. Collective control of rotor (Fig. 10) is almost proportional to forward speed. When forward speed is low (below 60 m/s) minimal tilt angle of nacelles (Fig. 11) assuring proper values of lift is about 70°. The deflection of wing flaps is maximum for low forward speed, to balance the inclination of lift from rotors. The deflection of elevators is maximal, when aircraft is in a conversion mode, because of the necessity to provide proper tiltrotor pitch moment. It has the maximum value when the conversion stops and the rotor axis are in the horizontal position.

Steady forward flight with vertical climb in conversion mode

In conversion mode, steady forward flight with vertical climb velocity with possibility of nacelle tilt angle deflection was simulated. The forward flight velocity was changed within the range $U = 20 \div 180$ m/s.

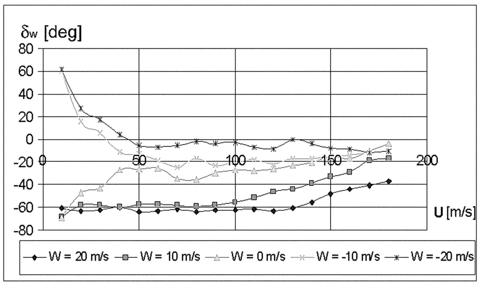

Fig. 12. Deflection of flaps

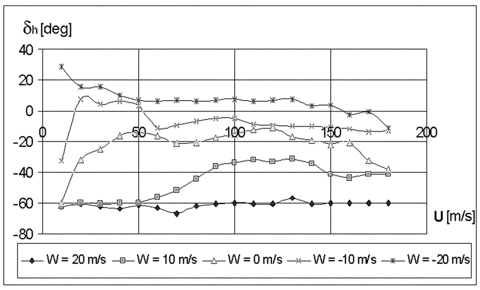

Fig. 13. Deflection of elevators

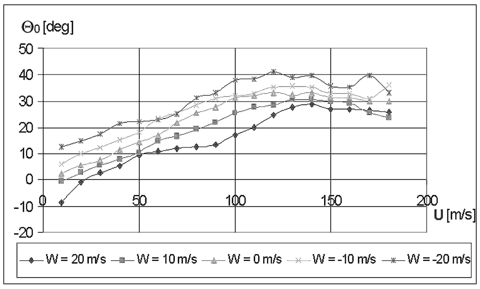

Fig. 14. Collective pitch

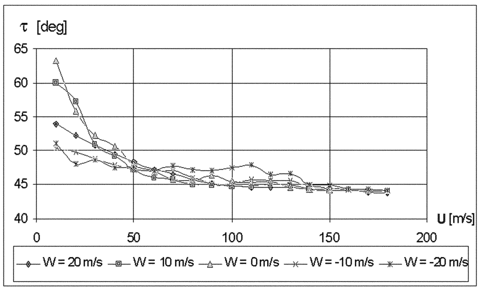

Fig. 15. Tilt angle of nacelles

As in the previous case, the values of calculated deflection angles of wing flaps (Fig. 12) and elevators (Fig. 13) do not exceed the values of available control for V-22. Collective pitch of rotor is approximately proportional to forward speed. The deflection of wing flaps (Fig. 12) is maximum for low forward speed. When forward speed is low minimum tilt angle of nacelles (Fig. 15) is about 55° assuring the proper aerodynamic lift. It becomes smaller (about 43°) when forward speed increases.

Steady forward flight with vertical climb in helicopter mode

In helicopter mode, steady forward flight with vertical climb with nacelles tilt angle 90° was simulated. The forward flight velocity was changed within the range $U = -70 \div 70$ m/s. The vertical velocity was changed within the range $W = -20 \div 20$ m/s.

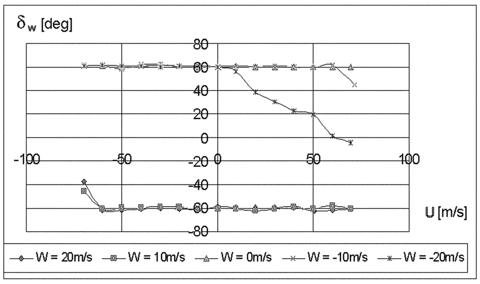

Fig. 16. Deflection of flaps

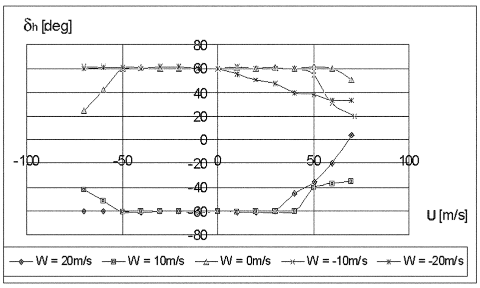

Fig. 17. Deflection of elevators

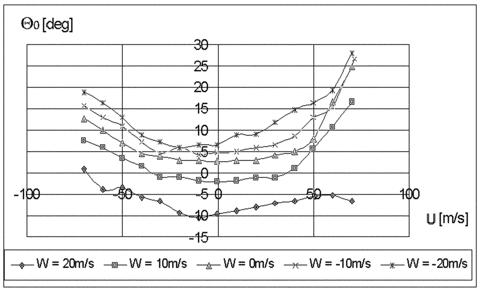

Fig. 18. Collective pitch

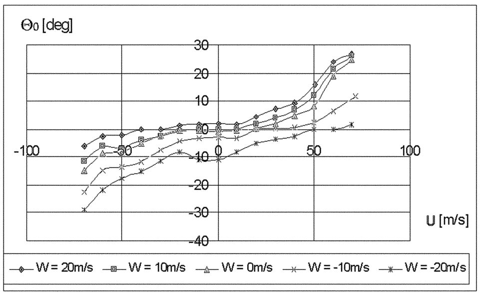

Fig. 19. Cyclic control (longitudinal)

For assumed steady flight conditions the negligible small value of pitch angle of tiltrotor was obtained. Collective control of rotor swash-plates (Fig. 18) increases with the increase of vertical speed. Cyclic control of rotor pitch (Fig. 19) varies in the opposite way: when vertical speed increases the pitch angle of tiltrotor is also stabilized. Values of inclination angles of wing flaps (Fig. 16) and elevators (Fig. 17) obtained in calculations do not exceed available values for V-22. In the range of negative forward speed the influence of flap deflection on tiltrotor motion is not substantial, but it becomes noticeable at positive forward speed greater than 10÷20 m/s. The sign of deflection of control surfaces depends on the direction of vertical speed.

SIDE FLIGHT IN HELICOPTER MODE

In the side flight in helicopter mode the forward flight velocity was changed within the range U = -70÷70 m/s, and the side flight velocity was changed within the range V = = -20÷20 m/s. The tiltrotor vertical velocity W = 0. Pitch lateral cyclic control was symmetrical $\mathbf{\Theta}_1 = \Theta_{11} = \Theta_{12}$ and longitudinal cyclic control - unsymmetrical $\mathbf{\Theta}_2 = \Theta_{21} = -\Theta_{22}$. Aircraft pitch and yaw angles, pitch and yaw rates and both rudder deflections were assumed zero.

From equilibrium conditions roll angle of tiltrotor Φ, inclination angles of wing flaps $\boldsymbol{\delta}_w = \delta_{w11}$, $\delta_{w12} = \delta_{w21} = \delta_{w22}$, inclination angles of elevators $\boldsymbol{\delta}_h = \delta_{h1} = \delta_{h2} = \delta_{h3}$, collective control of rotor swash-plate $\mathbf{\Theta}_0 = \Theta_{01} = \Theta_{02}$ were calculated. The results are shown in Fig. 20÷23.

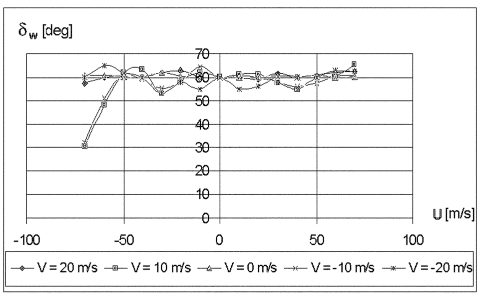

Fig. 20. Deflection of flaps

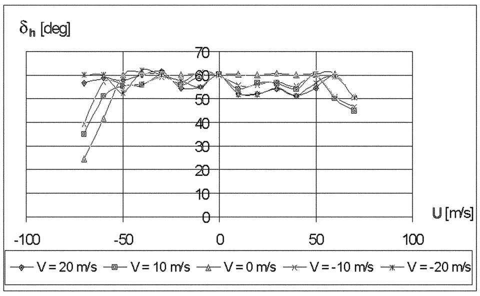

Fig. 21. Deflection of elevators

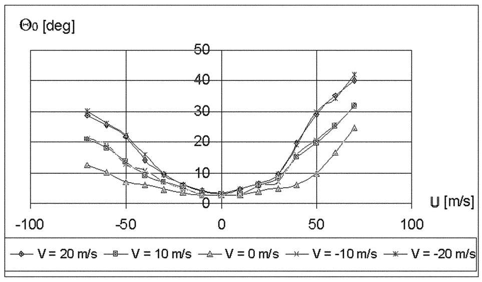

Fig. 22. Collective pitch

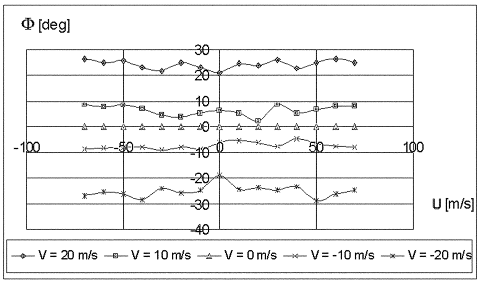

Fig. 23. Roll angle of tiltrotor

In this flight conditions the tiltrotor roll angle depends on side velocity. The deflections of flaps and elevators are small. The value of cyclic pitch control depends on the value of side speed, and its sign.

STABILITY AND CONTROLLABILITY

The tiltrotor equations of motion for stability analysis are linearized numerically with respect to state and control variables for given trim values. This allows analyzing the values of control matrix and calculation of eigenvalues and eigenvectors for stability analysis. This option of model analysis allows to investigate the stability and controllability of the aircraft.

For the assumed tiltrotor design data, both stability and controllability analysis was made for whole forward speed range of tiltrotor. In range of velocity considered the complex eigenvalues with negative real parts were obtained only in low velocity of forward flight (from 60 to 70 m/s), when conversion mode occurs. In other forward speed real, negative values of eigenvalues were obtained. These results show that in the steady flight, the aircraft is stable in the whole range of flight velocities.

Analysing the control matrix it may be stated, that the control variables influence the loads related to them with minor cross coupling effects. For airplane mode this is summarized in Table 2.

Tab. 2. Controllability analysis in airplane mode

Loads	Control elements		
	airplane mode	transition	helicopter mode
F_x	rotor collective pitch	nacelles tilt angle rotor collective pitch	rotor longitudinal cyclic pitch
F_y	rudders, (no lateral cyclic pitch)	rudders, (no lateral cyclic pitch)	rotor lateral cyclic pitch
F_z	flaps, elevators (not used nacelles tilt, nor longi-tudinal cyclic pitch)	flaps, nacelles tilt, rotor collective pitch, elevators (smaller influence)	rotor collective pitch
M_x	flaps, elevators, rotor collective pitch	deflection of flaps, rotor collective control	asymmetric rotor collective pitch
M_y	elevators, (rotor longitudinal cyclic pitch not used)	deflection of elevators, rotor collective control	rotor longitudinal cyclic pitch (symmetric)
M_z	rotor collective pitch, rudders	rotor collective pitch, rudders	rotor collective pitch rotor longitudinal cyclic pitch

RESULTS OF NUMERICAL FLIGHT SIMULATIONS

For the calculated steady flight parameters, the simulations of flight were carried out to check their accuracy. The steady flight in the airplane mode was simulated for: $U = 180$ m/s, $V = W = P = Q = R = 0$, $\Phi = \Theta = \Psi = 0$ and trim parameters obtained from calculations:

Deflection of wing flaps $\delta_w = -6.75°$, elevators $\delta_h = 7.23°$ and collective pitch of rotor $\Theta = 45.91°$.

Tiltrotor rotor displacements are presented: in Fig. 24 – horizontal and in Fig. 25 – vertical. The variations of attitude angles were very small (below 0.1°).

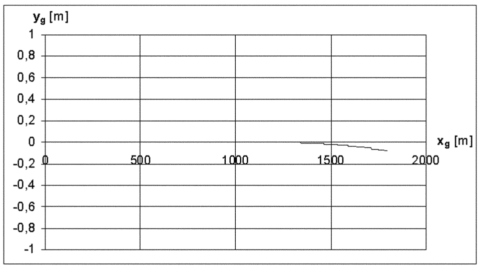

Fig. 24. Horizontal displacement in airplane mode

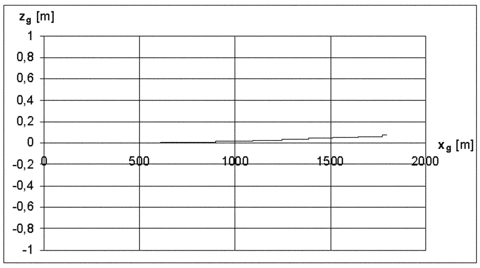

Fig. 25. Vertical displacement in airplane mode

It may be seen that during a 1800 m distance the flight altitude decreased by about 0.1 m and side translation was about 0.1 m to the left. These values are attributed to numerical errors in calculating trim values. Similar results were obtained in transition and helicopter modes.

CONCLUSIONS

The tiltrotor computer model was developed for flight simulation, trim stability and control analysis. The model is constructed of rigid elements: fuselage, wings, empennage, rotors, but due to modularity of the code this assumptions may be easily released. The designed parameters of V-22 tiltrotor were used for simulations. Some data of tiltrotor had to be assumed, and there was no possibility to compare results of numerical simulations and flight data. From the results of calculations performed it may be concluded, that the model of tiltrotor developed during this research works properly.

A tiltrotor is a complex rotorcraft, and several simplifying assumptions were applied. They may be released adjusting model for specific needs of particular helicopter.

Acknowledgement

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ZASTOSOWANIE OGÓLNEGO MODELU OBIEKTU RUCHOMEGO DO ANALIZY STATECZNOŚCI TILTROTORA

Streszczenie

Opracowano symulacyjny model statku powietrznego typu tiltrotor dla potrzeb analizy osiągów, stabilności i sterowania oraz symulacji lotu. Model wiropłata złożony jest z kadłuba, skrzydeł, usterzenia ogonowego, gondoli silnikowych i wirników. Równania ruchu zostały uzyskane z sumowania obciążeń bezwładności, grawitacyjnych i aerodynamicznych działających na każdy element statku powietrznego. Obciążenia aerodynamiczne skrzydeł, stateczników i łopat wirników zostały obliczone z zastosowaniem quasistacjonarnego modelu opływu. Do wyznaczania prędkości indukowanej wirników zastosowano model Glauerta. Wpływ na-czany z wykorzystaniem aktualnej wartości prędkości indukowanej. Program do modelowania wiropłata został opracowany w środowisku MatLab. Program złożony jest z modułów służących do obliczeń obciążeń poszczególnych elementów wiropłata, które wykorzystywane są również do wyznaczania warunków stanów ustalonych, stateczności i macierzy sterowania. Podczas pierwszego etapu badań wyznaczono warunki lotu ustalonego wiropłata typu tiltrotor w różnych konfiguracjach co pozwoliło zbadać zachowanie i potwierdzić poprawność modelu.

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ПРИМЕНЕНИЕ ОБЩЕЙ МОДЕЛИ МОБИЛЬНОГО ОБЪЕКТА ДЛЯ АНАЛИЗА УСТАЙЧИВОСТИ TILTROTORA

Резюме

Разроботана имитационная модель воздушного судна типа tiltrotor для анализа характеристик, устайчивости и управления а также имитации полета. Модель винтокрыла состоит из фюзелажа, крыльев, хвостового оперения, мотогондол и винтов. Уравнения движения получены суммированием инерциальных, гравитационных и аэродинамических нагрузок действующих на каждый элемент воздушного судна. Аэродинамические нагрузки крыльев, стабилизатора и киля и лопастей винтов рассчитаны с применением квазистационарной модели обтекания. Для определения индуктивной скорости винтов применена модель Gluerta. Влияние наплыва за винтом на крылья, стабилизатор и киль определено с использованием актуальной величины индуктивной скорости. Программа для моделирования винтокрыла разработана в среде MatLab. Программа состоит из модулей для расчетов нагрузок отдельных элементов винтокрыла, которые используются также для определения условий установившихся состояний, устойчивости и матрицы управления. Во время первого этапа исследований определены условия установившегося полета винтокрыла типа tiltrotor в разных конфигурациях, что дало возможность исследовать поверение и подтвердить правильность модели.