Development of a Tilt-Rotor Aircraft Model Using System Identification Technique

Antonio Vitale, Nicola Genito, Giovanni Cuciniello, Ferdinando Montemari

***Abstract*—**The introduction of tilt-rotor aircraft into the existing civilian air transportation system will provide beneficial effects due to tilt-rotor capability to combine the characteristics of a helicopter and a fixed-wing aircraft into one vehicle. The disposability of reliable tilt-rotor simulation models supports the development of such vehicle. Indeed, simulation models are required to design automatic control systems that increase safety, reduce pilot's workload and stress, and ensure the optimal aircraft configuration with respect to flight envelope limits, especially during the most critical flight phases such as conversion from helicopter to aircraft mode and vice versa. This article presents a process to build a simplified tilt-rotor simulation model, derived from the analysis of flight data. The model aims to reproduce the complex dynamics of tilt-rotor during the in-flight conversion phase. It uses a set of scheduled linear transfer functions to relate the autopilot reference inputs to the most relevant rigid body state variables. The model also computes information about the rotor flapping dynamics, which are useful to evaluate the aircraft control margin in terms of rotor collective and cyclic commands. The rotor flapping model is derived through a mixed theoretical-empirical approach, which includes physical analytical equations (applicable to helicopter configuration) and parametric corrective functions. The latter are introduced to best fit the actual rotor behavior and balance the differences existing between helicopter and tilt-rotor during flight. Time-domain system identification from flight data is exploited to optimize the model structure and to estimate the model parameters. The presented model-building process was applied to simulated flight data of the ERICA Tilt-Rotor, generated by using a high fidelity simulation model implemented in FlightLab environment. The validation of the obtained model was very satisfying, confirming the validity of the proposed approach.

***Keywords*—**Flapping Dynamics, Flight Dynamics, System Identification, Tilt-Rotor Modeling and Simulation.

I. Introduction

TILT-ROTOR is a relative new category of aircraft and, although today they are used exclusively for military applications, there is a lot of interest in the development of this kind of aerial vehicles for civilian applications. Tilt-rotor combines into a single aircraft the advantages of both helicopter (hovering, vertical take-off and landing) and fixed-wing airplane (high cruise speed). Therefore, it is very flexible, time saving, and capable of a cost-competitiveness solution with respect to small aircraft or helicopters in medium range movement. Several research activities [1]-[4] highlighted the beneficial effects of introducing tilt-rotor into the existing air transportation system.

Reliable simulation models represent a critical asset for the evolution of tilt-rotor. Indeed, they allow rapid assessment of vehicle's performance and support the design of dedicated flight control systems that can help pilots during the most critical phases of flight. Several approaches are available in the literature to model tilt-rotor aircraft. Simplified models represent the vehicle through 6 degree of freedom (DOF) rigid body equations [5]-[7], which do not simulate the rotor dynamics and do not take into account the effects of inertia coupling during conversion. Multi-body equations of motion are proposed in [8], where the aircraft, the nacelles and the rotors are considered as independent bodies that influence each other. This model still treats the aircraft and the blades as rigid body with negligible elastic deformation. The model equations are implicit and represent an increased number of DOF. Multi-body approach is also presented in [9], where the bodies are assumed to be rigidly connected to each other. The model includes elastic beam equations, to simulate rotor blade and wing flexibility, and finite-state inflow model for each rotor. More complex tilt-rotor model implementation is feasible by using the MBDyn multi-body dynamics code [10], or the commercial environments FlightLab [11], [12] and CAMRAD II [13]. These software environments allow performing detailed simulation of the tilt-rotor dynamics, including high order multi-body dynamics, nonlinear finite elements, elastic blade and structural dynamics, wake models and rotor aerodynamics, and deriving linearized models, which are very useful in control system design. Indeed, model-based control system design requires the availability of simulation models that on the one hand shall be able to catch all the main relevant vehicle dynamics and on the other hand shall be enough simple to allow easily understanding the physical phenomena and keep limited the computational burden.

This paper presents the development process to build up a simplified tilt-rotor simulation model from the analysis of experimental data. The purpose is to obtain a simulation tool able to reproduce the tilt-rotor dynamics during the conversion phase, in order to support the design of nacelle control systems for automatic conversion. Indeed, the disposability of a reliable simulation model is essential in order to reduce the flight test time, cost and risk. The modeling problem is divided into two parts: the first one aims to reproduce the rigid body dynamics for the conversion phase, including Stability and Control Augmentation System (SCAS) and standard

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autopilot (AP) functions emulation; the other one has the purpose to model the rotor blades flapping phenomenon. A set of scheduled linear transfer functions between AP reference inputs and most relevant vehicle state variables simulates the highly nonlinear dynamics of the rigid body in the conversion phase, whose complex behaviour depends on both aircraft and helicopter configuration characteristics. For what concerns the rotor flapping model, physical analytical equations, which are applicable to helicopter configuration, are coupled to parametric empirical corrective functions. It permits to fit the actual rotor functioning, to compensate for inaccurate values of tilt-rotor synthetic parameters, and to balance the differences existing between helicopter and tilt-rotor during flight. A key innovation of the proposed approach is the application to both model parts of time-domain system identification techniques, in order to select the model structure and to estimate the model's parameters. In this process, a suitable trade-off allows balancing the complexity of the reproduced target system and the need to keep the simulation model as simple as possible. The proposed approach was validated through the application to simulated flight data of the ERICA Tilt-Rotor, obtained by using a high fidelity simulator developed in FlightLab environment. The analysis of several conversion maneuvers enabled the modeling of two effects (the vehicle air speed and the nacelle rotation angle) that constitute the main relevant rigid body dynamics during the conversion phase. Obtained model also reproduced the flapping dynamics. The validation results were very satisfying and they confirmed the validity of the proposed approach. Indeed the model was capable to fit simulated flight data along maneuvers not used during the model definition process.

The paper presents, in Section II, the mathematical model formulation, including hypotheses and assumptions on which the model relies. Section III addresses the system identification methodology applied to estimate model parameters. Section IV describes the case study, simulated data set used for model development and validation, and validation results. Finally, a conclusion section ends the paper.

II. Tilt-Rotor Model Formulation

A. Model Assumptions and Architecture

The aim of the simulation model is to support the design of a nacelle control system, able to perform automatic conversion from helicopter to aircraft and vice versa. Therefore, the model shall represent the vehicle dynamics only within a limited flight envelope, that is, the conversion corridor defined in the nacelle angle versus vehicle speed plane. Fig. 1 presents a typical tilt-rotor conversion corridor envelope. The flight conditions usually experienced during the conversion allow introducing the following simplification hypotheses:

- H1) Conversion maneuver occurs in purely longitudinal (negligible side acceleration), in neutral attitude and leveled wings (roll and yaw angles are null).
- H2) Conversion maneuver follows predefined altitude and Indicated Air Speed (IAS) profiles.
- H3) SCAS and AP work during conversion maneuvers, and AP is in charge to track the reference altitude and IAS profiles.
- H4) Nacelle rotation follows a "step movement" logic. That is, the nacelle rotation can only stop at some predefined nacelle angle values, denoted as detent; the movement between two following detents is continuous and at constant rotation speed. Both detent points and nacelle rotation speed are design parameters of the vehicle.
- H5) Rotor model only includes flapping dynamics; blade lag and torsion are not simulated. Specifically, the model assumes that blade twist angle is constant and represents a characteristic of the vehicle.
- H6) A mixed theoretical-empirical approach describes the flapping phenomenon, whose theoretical formulation is discussed in [14]. The model assumes a center-spring equivalent rotor with straight and rigid blades, linear rotor lift force with respect to local blade incidence, and drag force quadratic function of lift. It neglects the unsteady aerodynamic effects, tip losses, non-uniform spanwise inflow distribution and reversed flow effects. The introduction of empirical correcting factors into the model permits to compensate the difference between helicopter (to which the basic theoretical formulation refers) and tilt-rotor during conversion.

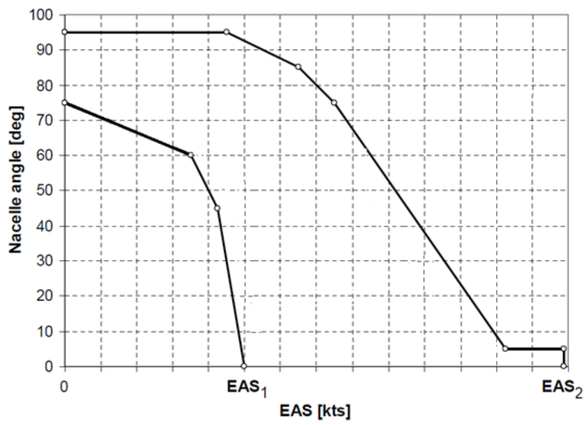

Fig. 1 Tilt-rotor conversion corridor envelope

Although the detailed simulation of a tilt-rotor system would require complex and strongly nonlinear models, in the above listed hypotheses, a set of suitably scheduled linear functions well approximate the relevant vehicle dynamics, satisfying the rule of thumb that model should be as simple as possible.

Concerning rigid body dynamics (to which hypotheses H1 to H4 refer), hypothesis H3 requires that the simulation model shall represent the closed loop of the whole system composed of AP, SCAS, and bare airframe. It helps to define the inputs to the model, which are restricted only to the parameters or commands that are necessary for the conversion simulation. The inputs are the reference IAS profile, the reference altitude profile, and the nacelle angle. Indeed, this last parameter influences the dynamics and is the output of the nacelle control system whose design uses the model described in this paper. Hypotheses H1 and H2 allow selecting the state variables that shall be modeled. They are actual IAS value (different from the reference command provided as input to the standard AP) and pitch angle, which are two notable parameters whose behavior is function of the conversion flight dynamics. Further than IAS and pitch angle, the rigid body model computes the longitudinal acceleration.

Regarding rotor dynamics, hypothesis H5 identifies the state variables of the model. They are the blades flapping angles, represented in multi-blade coordinate system [14] in the hypothesis of quasi-steady flap equations (differential coning is neglected). H6 allows simplifying the structure while well approximating the rotor dynamic associated with low-frequency fuselage motion for helicopter.

In addition to rigid body and flapping dynamics, the model also includes the following auxiliary modules, which calculate intermediate variables required as inputs by the flapping model:

- The AP Emulator is a control system that emulates the performance of standard AP and SCAS. It evaluates the rotor low-level commands (collective command, longitudinal and lateral cyclic commands) and rotor thrust commanded by the AP in the actual flight condition to track the reference input (IAS and altitude).
- The Inflow Computation evaluates the inflow velocities.

Fig. 2 presents the proposed model structure, whereas Tables I and II summarize the model's inputs and outputs, respectively. The following sections describe the mathematical formulation and the system identification process applied to derive each module showed in Fig. 2.

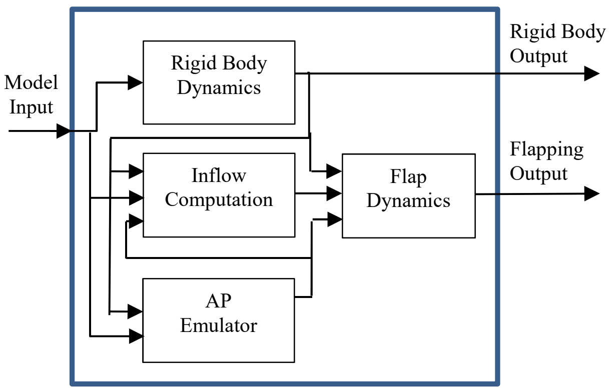

Fig. 2 Tilt-rotor model structure

B. Mathematical Formulation

The Rigid Body Dynamics module computes actual IAS, pitch angle, and longitudinal acceleration. The basic idea is to evaluate IAS and pitch angle variations with respect to their initial values (IAS_0 and Θ_0, respectively) as summation of two contributions, both expressed through single input single output (SISO) transfer function. The first contribution reproduces the effect of the reference IAS on the model outputs; the introduction of the second term takes into account the disturbances on the outputs due to the nacelle motion. The model computes the longitudinal acceleration as the time derivative of actual IAS.

TABLE I
MODEL INPUTS

Symbol	Quantity
IAS_{ref}	Reference Indicated Air Speed
H_{ref}	Reference altitude
Θ_{NAC}	Nacelle angle

TABLE II
MODEL OUTPUTS

Symbol	Quantity
IAS	Actual Aircraft Indicated Air Speed (included in Rigid Body Output)
N_x	Longitudinal acceleration (included in Rigid Body Output)
Θ	Actual aircraft pitch angle (included in Rigid Body Output)
β_0	Coning flapping angle (included in Flapping Output)
β_{1C}	Longitudinal cyclic flapping angle (included in Flapping Output)
β_{1S}	Lateral cyclic flapping angle (included in Flapping Output)

The mathematical formulation of this model is:

$$IAS(t) - IAS_0 = \mathbf{G_1}\left(\Theta_{NAC-\det}, sign\left(\dot{\Theta}_{NAC}\right)\right) \cdot IAS_{ref}(t) + \\ + \mathbf{G_2}\left(\Theta_{NAC-\det}, sign\left(\dot{\Theta}_{NAC}\right)\right) \cdot \Theta_{NAC}(t) \tag{1}$$

$$\Theta(t) - \Theta_0 = \\ K\left(IAS(t), sign\left(\dot{\Theta}_{NAC}\right)\right) \cdot \left[\mathbf{G_3}\left(\Theta_{NAC-\det}, sign\left(\dot{\Theta}_{NAC}\right)\right) \cdot IAS_{ref}(t) + \\ + \mathbf{G_4}\left(\Theta_{NAC-\det}, sign\left(\dot{\Theta}_{NAC}\right)\right) \cdot \Theta_{NAC}(t)\right] \tag{2}$$

$$N_X(t) = I\dot{A}S(t). \tag{3}$$

K in (2) takes into account the dependence of the pitch angle gain on the value of the IAS. Indeed, different airspeeds correspond to different pitch trim angles. The gain is therefore a two-dimensional table, whose independent variables are the IAS itself and the sign of nacelle angular rate $\dot{\Theta}_{NAC}$, which identifies the type of conversion (from helicopter to aircraft or vice versa). The number of breakpoints for this tabular gain shall be low, in order to avoid introducing too many model parameters. $\mathbf{G_i}$ (i = 1, 2, 3, 4) is defined by (4):

$$G_i\left(\Theta_{NAC-\det}, sign\left(\dot{\Theta}_{NAC}\right)\right) = \frac{\sum_{k=0}^{ni\left(\Theta_{NAC-\det}, sign\left(\dot{\Theta}_{NAC}\right)\right)} b_{ki}\left(\Theta_{NAC-\det}, sign\left(\dot{\Theta}_{NAC}\right)\right) D^k}{\sum_{k=0}^{ni\left(\Theta_{NAC-\det}, sign\left(\dot{\Theta}_{NAC}\right)\right)} a_{ki}\left(\Theta_{NAC-\det}, sign\left(\dot{\Theta}_{NAC}\right)\right) D^k}. \tag{4}$$

The symbol D^k in (4) denotes the k-th order time derivative; *ni* is the order of the transfer function represented by the functional $\mathbf{G_i}$, and a_{ki} and b_{ki} are its parameters. Order and parameters generally depend on nacelle angle and conversion type; therefore, a different implementation of $\mathbf{G_i}$ is defined for each value of nacelle detent angle $\Theta_{NAC-\det}$ and sign of nacelle angular rate $\dot{\Theta}_{NAC}$. It is worthy to note that usually few nacelle detents are designed (nacelle ends and one or two intermediate positions); consequently, the model requires the definition of few realizations for each functional $\mathbf{G_i}$. A scheduling logic allows selecting during the simulation the applicable implementation and the introduction of suitable merging functions guarantees a smooth transition when $\mathbf{G_i}$ switches between two scheduled functional realizations. Based on tilt-rotor dynamic characteristics, some of the above-described scheduling could be not necessary. On the other hand, some further correcting factors (such as saturation or rate limiter on some of the addend on the right hand side of (1) and (2)), could be added to the model if the analysis of flight data highlight this need. The evaluations on scheduling simplification or additional correction factors inclusion, as well as the estimation of gain, functionals order and parameters, is part of the model identification process described in Section III.

The Inflow Computation module is an auxiliary model introduced to compute the inflow velocity components, which are inputs needed for flapping dynamics evaluation. The applied model relies on classical local-differential momentum theory [14], which does not require the definition of any parameter. The symbol λ_0 denotes the rotor normalized uniform inflow velocities with respect to the plane orthogonal to the rotor shaft. In the hypothesis of small rotor's tip path angle, λ_0 is computed by solving the following equation [14]:

$$4\lambda_0^4 - 8\cdot\mu_Z\cdot\lambda_0^3 + 4\cdot\left(\mu_X^2+\mu_Z^2\right)\cdot\lambda_0^2 - C_T^2 = 0 \tag{5}$$

where μ_X and μ_Z are the longitudinal components of velocity, orthogonal and parallel to the rotor shaft, respectively, normalized with respect to the product between rotor speed (Ω) and radius (R), that is:

$$\mu_X = \frac{IAS_X}{\Omega\cdot R} \tag{6}$$

$$\mu_Z = \frac{IAS_Z}{\Omega\cdot R}. \tag{7}$$

IAS_X and IAS_Z are the projection of the IAS in the longitudinal plan, orthogonal and parallel to the rotor shaft, respectively. In the hypothesis of negligible vertical wind, their computation only depends on the outputs of the Rigid Body Dynamics module (IAS and pitch angle) and on the model inputs (reference IAS and altitude profiles, to compute the flight path angle, and nacelle angle). C_T is the thrust coefficient, given by [12]:

$$C_T = \frac{T}{\rho\cdot\pi\cdot\Omega^2\cdot R^4} \tag{8}$$

where ρ is the air density corresponding to the input altitude, and T is the vehicle thrust provided by the AP Emulator module. Solution of (5) changes at each simulation step and if more than one real solution exists at one step, then the solution closest to the one computed at the previous step is selected. The inflow model also includes the equations of the normalized first harmonic (λ_{1c} and λ_{1s}) inflow velocities with respect to the plane orthogonal to the rotor shaft [14]:

$$\lambda_{1C} = C_{lp}\cdot\frac{a_0\cdot\sigma}{16\cdot\lambda_0}\cdot\overline{q} \tag{9}$$

$$\lambda_{1S} = C_{lp}\cdot\frac{a_0\cdot\sigma}{16\cdot\lambda_0}\cdot\delta_{lon} \tag{10}$$

where a_0 is the main rotor blade lift curve slope, σ is the rotor solidity ratio, δ_{lon} is the cyclic longitudinal command (provided by the AP Emulator module), $\overline{q}$ is the rotor pitch rate (tilt-rotor pitch rate plus nacelle pitch rate, being the flight purely longitudinal) normalized with respect to rotor speed Ω, and C_{lp} is [14]:

$$C_{lp} = \left(1 + a_0\cdot\frac{\sigma}{16\cdot\lambda_0}\right)^{-1}. \tag{11}$$

The AP Emulator module implements a proportional-integral-derivative (PID) controller, to emulate the standard AP and SCAS dynamics. It aims at determining the values of thrust module (T), rotor collective (δ_{col}) and longitudinal cyclic (δ_{lon}) commands that allow tracking the reference IAS and altitude profiles with pre-defined performance (which are design specifications or deduced from the analysis of flight data). The tuning of the controller follows classical control design methods [15], not detailed in the present paper. If a black box of the actual standard AP and SCAS systems is available, the model can include this black box, which replaces the AP Emulator module.

The Flap Dynamics module consists of a classic analytic model of helicopter rotor flapping augmented with empirical gains, in order to balance the differences existing between helicopter and tilt-rotor during flight and to compensate for inaccurate values of tilt-rotor theoretical synthetic parameters, which are not measurable. The following linear static equations model the flapping components (β_0, β_{1C}, β_{1S}) in multi-blade coordinate system in the hypothesis of quasi-steady flap equations (negligible differential coning):

$$\begin{bmatrix}\beta_0\\ \beta_{1C}\\ \beta_{1S}\end{bmatrix} = \mathbf{G}_\Delta\circ\mathbf{A}_\Delta(\mathbf{P},\mu,\rho)\cdot\begin{bmatrix}\delta_{col}\\ \delta_{twist}\\ \delta_{lon}\end{bmatrix} + \mathbf{G}_\Lambda\circ\mathbf{A}_\Lambda(\mathbf{P},\mu,\rho)\cdot\begin{bmatrix}\mu_Z-\lambda_0\\ \lambda_{1S}\\ \lambda_{1C}\end{bmatrix} + \\ + \mathbf{G}_\Omega\circ\mathbf{A}_\Omega(\mathbf{P},\mu,\rho)\cdot\begin{bmatrix}\dot{\overline{q}}\\ \overline{q}\end{bmatrix}. \tag{12}$$

In (12) the symbol ° denotes the Hadamard (or element-wise) product, δ_{twist} is the blade twist angle, assumed constant according to hypothesis H5, and $\dot{\overline{q}}$ is the time derivative of the rotor pitch rate normalized with respect to Ω^2. The elements of matrices $\mathbf{A_\Delta}$, $\mathbf{A_\Lambda}$ and $\mathbf{A_\Omega}$ depend on air density (ρ), rotor's advance ratio (μ) and a vector of rotor synthetic parameters, denoted with $\mathbf{P}$, which includes rotor radius, rotor speed, blade inertia moment, flap hinge eccentricity and stiffness. The matrix expression is available in [14] and not reported here for the sake of brevity. $\mathbf{G_\Delta}$, $\mathbf{G_\Lambda}$ and $\mathbf{G_\Omega}$ are the empirical matrix gains introduced to correct the theoretical formulation. Based on sensitivity analysis, most of the elements of the gains could be fixed to one, in order to keep limited the number of parameters to be estimated. The tuning of the remaining gains is part of the model identification process.

III. System Identification Strategy

The application of system identification methodologies allows the complete definition of the model, by estimating from the analysis of flight data the model parameters included in (1), (2), (4), and (12). In particular, two different identification strategies avail to perform this task for Rigid Body Dynamics and Flapping Dynamics.

A. Rigid Body Model Identification

Model identification is carried out independently for IAS and pitch angle. The following steps compose the identification process:

1) Data Collection: Three different sets of flight data are collected. Each data set shall include all the inputs and outputs of the model, gathered in different maneuvers that cover the whole conversion corridor envelope.
2) Preliminary Model Identification: First data set is exploited to perform system identification of several models, which differ each other only for the order *ni* of the functionals $\mathbf{G_i}$ defined in (4). Minimum and maximum model order shall be defined before starting the process. For each model structure the proposed strategy executes the following steps:
a) It estimates all the model parameters (scheduled with respect to the respective independent variables). The estimation computes in two stages first the parameters of transfer function from IAS_{ref} to the output, next the parameters of the transfer function from Θ_{NAC} to the output. Estimation exploits equation error method that fits continuous-time transfer function models to discrete-time data [16].
b) It evaluates the Normalized Root Mean Square Error (NRMSE) [17] that is an identification performance metrics.
c) It selects the best model structure by discarding unstable transfer functions and choosing, among the stable functions, the one associated to minimum NRMSE (if different structures have close NRMSE, differing less than a predefined threshold, the one having minimum order is chosen).
3) Model Refinement: First, the tabular gain *K* (only for (2)) is determined by analyzing the trimmed conditions at the end of the conversion maneuvers included in the second set of data. In addition, if the model is not able to fit the flight data in local region of the flight envelope, then local correcting factors are included based on engineering evaluation. These factors could be saturation and/or rate limiter acting (in specific regions of the conversion corridor) on one or both the addends on the right hand side of (1) and (2).
4) Model Simplification: A sensitivity analysis evaluates if it is possible to remove some of the scheduling, in order to simplify the model, without degrading significantly the model performance.
5) Final Validation: It assesses the model performance on the third data set by evaluating the NRMSE between the identified model output and the corresponding flight data. If validation results are satisfying (that is, the NRMSE is below a pre-defined threshold for all the modeled variables) then the process ends and the identified model is released. Otherwise, the process discards the selected model structure and returns to step 2-c, where a new model structure is picked out. The process iterates until it finds a satisfying model or assesses all the possible structures among the one defined by minimum and maximum selected model order (that is, the process reaches the maximum number of iterations). This last case could happen when the data set used for identification contains poor dynamic information or the selected range of model orders is not able to represent correctly the tilt-rotor dynamic behavior. Both these issues shall be checked and, if possible, shall be corrected before restarting the process.

Fig. 3 summarizes the identification strategy for this model.

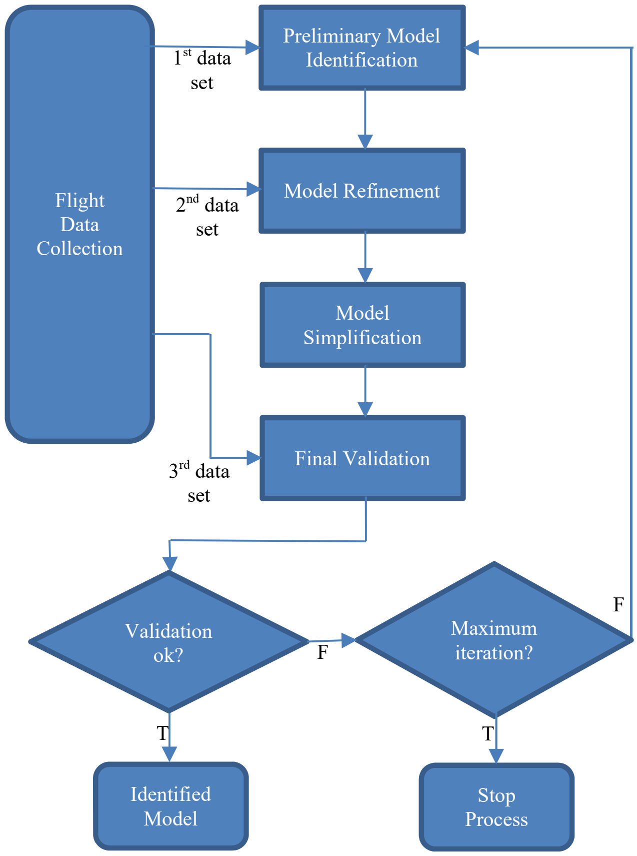

Fig. 3 Rigid Body model identification strategy

B. Flapping Model Identification

Equation (12) is static and it simplifies the identification process. The estimation of related parameters requires the execution of the following steps:

1) Data Collection: Two different sets of flight data are

collected. Each data set shall include all the inputs and outputs of the model, gathered in different maneuvers that cover the whole conversion corridor envelope.

2) Parameters Selection: A sensitivity analysis on the first data set allows identifying which gains mainly affect the model output. The estimation will only concern the gains (elements of matrices $\mathbf{G}_{\Delta}$, $\mathbf{G}_{\Lambda}$ and $\mathbf{G}_{\Omega}$) that produce variation of the output bigger than prefixed threshold (all the other gains are set to 1).
3) Parameters Estimation: A linear least square (LS) technique [18] in the time domain provides the estimation of all the selected gains by exploiting the first data set.
4) Model Validation: It assesses the model performance on the second data set by evaluating the NRMSE between the identified model output and the corresponding flight data. If validation results are satisfying, then the process ends and the identified model is released. Otherwise, the process returns to step 2) and selects for estimation a wider set of gains, by relaxing the sensitivity analysis threshold. The process iterates until it finds a satisfying model or it estimates all the elements of matrices $\mathbf{G}_{\Delta}$, $\mathbf{G}_{\Lambda}$ and $\mathbf{G}_{\Omega}$. This last case could happen if the examined data set has a poor flapping dynamics excitation.

Fig. 4 summarizes the identification strategy for this model.

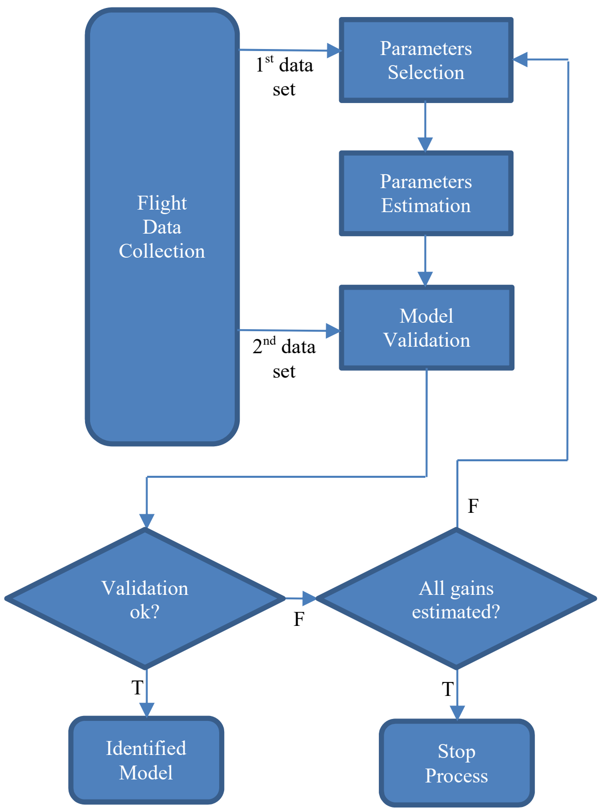

Fig. 4 Flapping model identification strategy

IV. Case Study

This section presents the results of the application of the proposed model development process to simulated flight data of the ERICA Tilt-Rotor concept [19]. The ERICA tilt-rotor is an advanced concept of a civil tilt-rotor aircraft developed within the European NICETRIP (Novel Innovative Competitive Effective Tilt Rotor Integrated Project) project. It is a medium large tilt-rotor size. The vehicle is characterized by high performance, in terms of speed (350 knots), range (650 nautical miles) and cruise altitude (7500 m), with a capability of 19-22 passengers. The ERICA configuration has two four-bladed rotors with a gimbaled system. In addition to a relatively small rotor, the ERICA configuration has the additional feature of tilting the outboard portion of the wing, in order to improve performance and handling qualities in helicopter mode. The ERICA model in FlightLab combines a heavily customized model, created using the tool available components, with various components specifically created for this tilt-rotor aircraft. It enables executing very detailed simulation of the tilt-rotor aircraft, including standard AP and SCAS systems. The considered conversion logic uses only three detents at 0 degree, 75 degrees and 90 degrees. The FlightLab ERICA model allowed collecting 30 simulated flight data sets for system identification purpose, gathered at different flight conditions. Table III presents the data sets list, including some relevant characteristics of the maneuvers. The simulations always start from a stable trim condition for a clean definition of tilt-rotor flight dynamic and stop when the vehicle reaches again a complete stabilization on the final point.

The IAS model, obtained by applying the proposed identification process to these data, has fourth order transfer functions for both the contribution depending on IAS_{ref} and Θ_{NAC}. The first transfer function does not require any scheduling. It means that $\mathbf{G}_1$ in (1) does not vary on the whole conversion corridor. Instead, the transfer function related to nacelle angle requires two different implementations with respect to the sign of nacelle angular rate (one implementation for each conversion type) and, only for conversion from aircraft to helicopter, two different implementations with respect to the detent angle from which nacelle rotation starts (0 degree or 75 degrees). Globally, the model includes three different forms of the functional $\mathbf{G}_2$, managed by a suitable merging logic. Finally, this last transfer function includes a local correction term implemented through a rate limiter on the output. It only applies when the nacelle angle is close to zero degree in conversion from helicopter to aircraft. The preliminary validation of the model (step 3) of Section III *A*) highlighted the need for this additional factor.

The obtained pitch angle model is more complex. The tabular gain K requires nine breakpoints (four points for conversion from aircraft to helicopter and five for the opposite conversion). Concerning the transfer function related to the reference IAS, three different implementations of the functional $\mathbf{G}_3$ in (2) are required, and the applicable implementation depends on the detent angle from which the nacelle starts the rotation. $\mathbf{G}_3$ is not dependent on the conversion type and all its implementations have fourth order. The transfer function from nacelle angle to pitch angle includes four implementations of $\mathbf{G}_4$, whose maximum order is five. The applicable functional form derives from both the sign of nacelle angular rate and the detent angle, from which the nacelle starts the rotation. Summarizing, a different form of $\mathbf{G_4}$ is applicable in each of the following conditions:

- Nacelle starts rotation from 0 degree;
- Nacelle starts rotation from 90 degrees;
- Nacelle starts rotation from 75 degrees with positive angular rate;
- Nacelle starts rotation from 75 degrees with negative angular rate.

TABLE III
FLIGHT MANEUVERS

N°	Configuration	IAS range, Kts	Nacelle range, Deg	Maneuvers duration, s
1	Conversion	200 – 5	0 – 90	200
2	Conversion	200 - 30	0 – 90	150
3	Conversion	180 - 20	0 – 90	120
4	Conversion	150 – 0	0 – 90	200
5	Conversion	130 – 10	0 – 90	200
6	Conversion	0 – 200	90 – 0	100
7	Conversion	0 - 200	90 – 0	200
8	Conversion	0 – 200	90 – 0	50
9	Conversion	50 - 200	90 – 0	100
10	Conversion	80 -200	90 – 0	100
11	Conversion	20 - 180	90 – 0	200
12	Conversion	20 – 170	90 – 0	260
13	Conversion	20 – 170	90 – 0	310
14	Helicopter	20	90	200
15	Helicopter	20 – 90	90	100
16	Helicopter	20 – 120	90	150
17	Helicopter	20 – 160	90	100
18	Helicopter	10 – 40	90	10
19	Helicopter	10 – 60	90	30
20	Helicopter	5 – 60	90	60
21	Helicopter	10 – 50	90	215
22	Helicopter	20 - 60	90	200
23	Helicopter	10 – 60	90	170
24	Helicopter	10 - 45	90	180
25	Aircraft	120 – 220	0	135
26	Aircraft	130 – 210	0	140
27	Aircraft	120 – 180	0	130
28	Aircraft	140 – 220	0	220
29	Conversion	30 – 120	75	120
30	Conversion	30 – 120	75	150

Finally, regarding the flapping model, the sensitivity analysis selected for estimation all the elements of matrix $\mathbf{G_\Delta}$ and of the first column of matrix $\mathbf{G_\Lambda}$. All the other parameters in (12) do not require an estimation and are set to one.

Figs. 5-10 present the results of the identified model validation, performed by comparing (along maneuvers not used during the identification process) the outputs of the model with the corresponding simulated flight data provided by the Flightlab ERICA simulator. Specifically, Figs. 5-7 refer to simulation N°1 of Table III, which reproduces a conversion from aircraft to helicopter, whereas Figs. 8-10 refer to simulation N°6 of Table III, which reproduces a conversion from helicopter to aircraft. Figs. 5 and 8 show the inputs to the simulations, except for the reference altitude profile, which is constant for both the maneuvers.

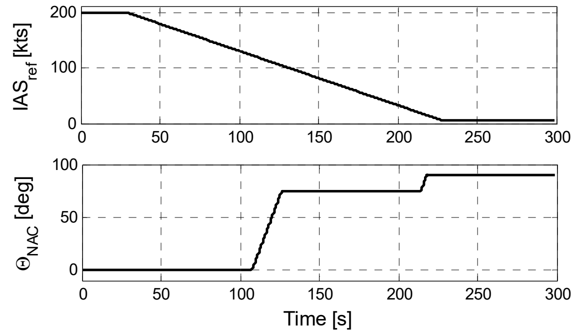

Fig. 5 Model inputs for conversion from aircraft to helicopter: time histories of reference IAS and nacelle angle

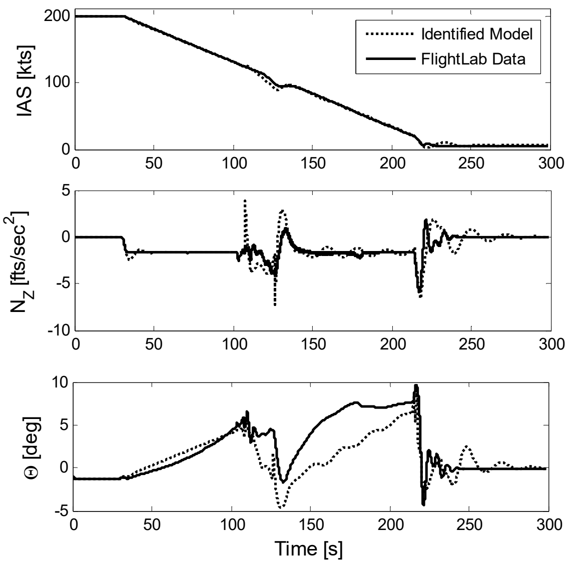

Fig. 6 Comparison between identified model outputs and simulated flight data for conversion from aircraft to helicopter: rigid body dynamics

The models of IAS and longitudinal acceleration perfectly fit the simulated flight data for both the conversion maneuvers. Some differences exist between pitch angle model output and FlightLab data, limited to the conversion from aircraft to helicopter (Fig. 6). Probably, the introduction of additional terms or scheduling in the model could compensate these differences. However, since the model is already able to catch the trends of the dynamic and in addition, it works perfectly during conversion from helicopter to aircraft (Fig. 9), the current model formulation represents a balanced trade-off between complexity and accuracy.

Concerning flapping dynamics, the model outputs fit quite well the simulated flight data. Some small deviations only appear on lateral cyclic flapping angle, limited to the conversion from aircraft to helicopter (Fig. 7). This result could derive from a poor excitation of this dynamic during the maneuvers used for model identifications. Globally, the identified model performance are fully satisfying, with excellent results for what concerns the conversion from helicopter to aircraft.

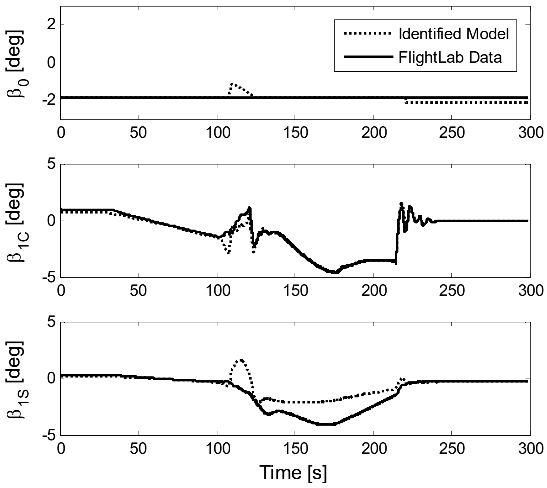

Fig. 7 Comparison between identified model outputs and simulated flight data for conversion from aircraft to helicopter: flapping dynamics

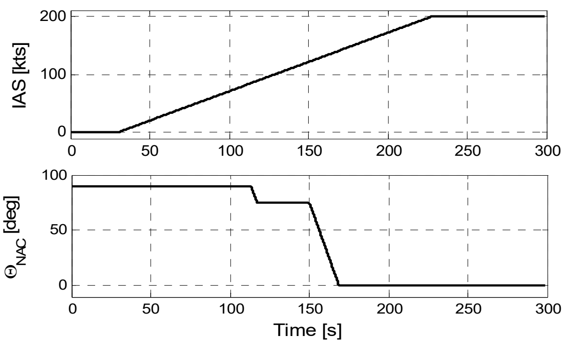

Fig. 8 Model inputs for conversion from helicopter to aircraft: time histories of reference IAS and nacelle angle

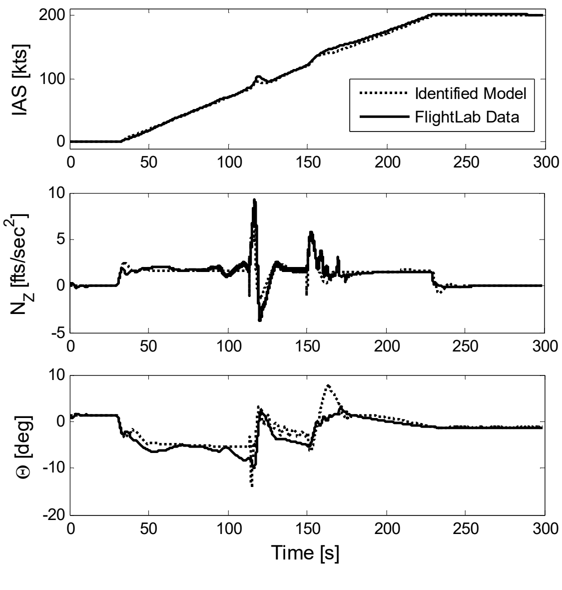

Fig. 9 Comparison between identified model outputs and simulated flight data for conversion from helicopter to aircraft: rigid body dynamics

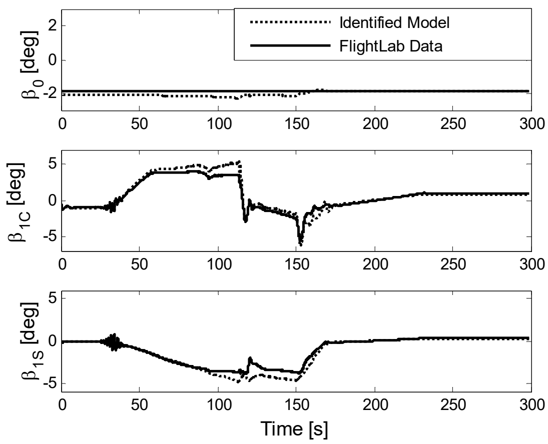

Fig. 10 Comparison between identified model outputs and simulated flight data for conversion from helicopter to aircraft: flapping dynamics

V. Conclusion

This paper presented a process to develop a tilt-rotor simulation model from the analysis of flight data. The obtained model aims at representing through simplified dynamic equations the complex behavior of the tilt-rotor in the conversion phase of flight. This model is applicable to the design of control system, which shall be able to perform automatic conversion from aircraft to helicopter and vice versa.

The paper focuses on the modeling of two main phenomena, the closed loop rigid body dynamics and the rotor blades flapping. A mixed theoretical-empirical approach and classical system identification methodologies are exploited to build the dynamic equations of some selected variables, which represent these phenomena.

The application of this process to simulated flight data, generated through a high fidelity simulator of the ERICA tilt-rotor concept, highlighted the reliability of the proposed approach and the capability of the identified model to fit the dynamics of interest. This result confirms that the presented development process provides an identified model that represents a good compromise between model complexity and accuracy.

Future steps of this work will concern further validations of the proposed model development process, by its application to other test vehicles and possibly to actual tilt-rotor flight data.

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