ALL-ELECTRIC FLIGHT CONTROL SYSTEM AND LANDING GEAR SYSTEM MODELS FOR POWER ASSESSMENT STUDIES

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ABSTRACT
The complete electrification of aircraft power systems entails the implementation of smart logics for sharing the available energy among the loads, and the design of these logics requires the characterisation of the power absorption of each on-board system as a function of mission phase and aircraft operating point, also taking into account the level of criticality of the function implemented by the system itself.

The paper describes the models of the electro-mechanical systems used for the landing gear extension/retraction and the flight control actuation of a regional aircraft, with the basic objective of evaluating the power requests that have to be fulfilled both continuously and completely for these safety-critical equipments.

The Flight Control System (FCS) model is composed of both primary and secondary flight controls. The Landing Gear System (LGS) model refers to the landing gear of a typical turboprop regional aircraft, with a single nose and two main gears. Both the landing gears (LGs) and the control surfaces (CSs) are driven by Electro-Mechanical Actuators (EMAs). All the EMA models refer to actuators with a 3-phase synchronous brushless motor and mechanical transmission.

Simulation tests have been performed to assess the maximum power flows characterizing the systems, with reference to severe operative conditions.

INTRODUCTION
The design of modern airborne systems and equipments constantly tends towards both technology innovation and integration of functions. This trend has been recently emphasized by research programs oriented to the conversion of aircraft systems to the “all-electric” solution [1][2][3][4], which, together with well-known advantages in terms of eco-sustainability and cost reduction, point out important issues on power management for systems [5][6][7]. Actually, in the conventional aircraft technological frame, different types of power are derived from the engine for supplying the on-board systems: pneumatic, hydraulic, mechanical and electrical. The All-Electric Aircraft (AEA) concept is instead based on the replacement of hydraulic and pneumatic power with the electrical one, so that the suggestive acronym PbW (Power by Wire) has been coined for referring to a solution that satisfies all power requests by cables. As for any technological innovation, new issues related to performances and reliability are expected, as well as systems interactions concerns in terms of electrical power flows. The challenging all-electric objective can be thus achieved by appropriately monitoring and managing the power requests (e.g. by temporarily reducing the power supplied to some systems during those flight phases in which the total request of electrical power could overcome the maximum available). Energy Management System (EMS) logics needed to be used, starting from the characterisation of the power absorption of each on-board system as a function of mission phase and aircraft operating point. In this context, a strong effort is required for system engineers to develop models that are capable of predicting the aircraft systems power flows. Modelling and simulation activities play a key role in this design loop: firstly, because the power management can be simulated in operating conditions that are difficult to be tested in flight or via on-ground rigs, pointing out criticalities; secondly, because the electro-mechanical actuation is a novel technology for flight controls and the landing gears, and the influence on performances of electrical power quality, actuator thermal behaviour and power electronics efficiency needs to be investigated and predicted.

LANDING GEAR SYSTEM DATA
The model has been developed with reference to a LGS of a typical turboprop regional aircraft with one nose LG and two main LGs.

The LGS parameters have been defined starting from the basic design parameters reported in Table 1.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Nose LG mass</td>
<td>40 kg</td>
</tr>
<tr>
<td>Main LG mass (right+left)</td>
<td>400 kg</td>
</tr>
<tr>
<td>Extension (Retraction) time</td>
<td>8 (12) sec</td>
</tr>
<tr>
<td>Nose LG EMA working stroke</td>
<td>148 mm</td>
</tr>
<tr>
<td>Main LG EMA working stroke</td>
<td>310 mm</td>
</tr>
<tr>
<td>Position error steady-state</td>
<td>&lt; 1%</td>
</tr>
</tbody>
</table>

Table 1: Basic requirements for the landing gear system
**FLIGHT CONTROL SYSTEM DATA**

The model is developed with reference to a FCS composed of the following control surfaces (Figure 1):

- **Primary flight controls**
  - Ailerons and Steering Spoilers for roll control;
  - Elevators for pitch control;
  - Rudder for yaw control.
- **Secondary flight controls**
  - Inboard and outboard Flaps.

![Figure 1: Flight control system layout](image)

**BASIC SIMULATION MODEL**

Both FCS and LGS models have been developed by using for each landing gear and each control surface the basic Matlab-Simulink scheme depicted in Figure 2.

Essentially, the basic scheme is composed of two main models:

- an EMA model and a dynamics and kinematics model of landing gear (or control surface). It is worth noting the strong interaction existing between these two models. EMA model evaluates the actuator rod dynamics, whose acceleration, velocity and position are needed to the dynamics and kinematics model to provide the force acting on EMA itself. This interaction determines an algebraic loop in the Simulink scheme that has been solved by applying an integration step delay (memory block) on the force signal.

**EMA MODEL**

The functional model of the EMA system has been developed under the following assumptions:

- the 3-phase brushless motor dynamics is described in the dq0 frame (no Park transforms are used) neglecting the homopolar axis
- the power electronics is ideal (no voltage drop on a closed switch, no current through an opened switch, perfect voltage tracking)
- the mechanical transmission is perfectly rigid

The electric equations along the direct and quadrant axes of the motor are thus given by Eqs. (1)-(2),

\[
V_d = R i_d + L_d \frac{di_d}{dt} - L_p i_q \omega_m \\
V_q = R i_q + L_q \frac{di_q}{dt} + L_p i_d \omega_m + K_b \omega_m
\]

where \( R \) is the motor phase resistance, \( p_m \) is the number of magnet pole pairs, \( \omega_m \) is the motor shaft speed, \( K_b \) is the motor back-electromotive force coefficient, while \( V_d, V_q, L_d, L_q, i_d \) and \( i_q \) are the voltages, the inductances and the currents along the direct and the quadrant axes respectively.

The motor torque is then provided by Eq. (3),

\[
T_m = [(L_d - L_q) p_m i_d + K_b] i_q
\]

and the system dynamics can be described with a single momentum equation, referred to the motor shaft, Eq. (4),

\[
J_{tot} \ddot{\omega}_m = T_m - T_{ext} - B_{tot} \omega_m - T_{f\,tot} \text{sgn}(\omega_m)
\]

where

\[
J_{tot} = J_m + J_g \frac{1}{\tau_g}
\]

\[
T_{ext} = I_s F_{ext}
\]

In Eq. (4), \( J_{tot} \) is the EMA inertia, \( B_{tot} \) is overall viscous damping coefficient, \( T_{ext} \) accounts for sliding friction effects (Coulomb, Striebeck, etc.), \( T_{f\,tot} \) is the external loading torque, scaled to the motor shaft; while in Eqs. (5)-(6), \( I_s \) is the screw lead pitch, \( \tau_g \) is the gearbox ratio, \( F_{ext} \) is the external force (coming from the
dynamics and kinematics model), and \( J_m \) and \( J_g \) are the inertia of the motor shaft and gears, respectively.

Concerning the EMA control, four simply proportional loops are used (position, motor speed, direct current and quadrant current), Eqs. (7)-(10),

\[
V_d = -K_i i_d \tag{7}
\]

\[
V_q = K_i (i_{qi} - i_q) \tag{8}
\]

\[
i_{qi} = K_o (\omega_m - \omega_m) \tag{9}
\]

\[
\omega_m = K_x (x_{ai} - x_a) \tag{10}
\]

in which \( K_i \), \( K_o \), \( K_x \) are the control gains of the current, speed and position loops respectively, and \( x_a \) is the actuator position, Eq. (11).

\[
\dot{x}_a = \frac{l}{\tau_g} \omega_m \tag{11}
\]

In addition, both voltage and current demands are limited, by saturating the related values to \( V_{\text{max}} \) and \( i_{\text{max}} \) respectively. The power balance is given by Eq. (12).

\[
P_e^{(\text{in})} = P_e^{(s)} + P_e^{(i)} + P_m^{(i)} + P_m^{(out)} + P_m^{(\text{out})} \tag{12}
\]

In Eq. (12), \( P_e^{(\text{in})} \) is the electrical power input, \( P_e^{(s)} \) is the electrical power stored in the system (only related to inductance, no capacitors are considered), \( P_e^{(i)} \) is the electrical power lost in the circuitry, \( P_m^{(i)} \) and \( P_m^{(\text{out})} \) are the mechanical powers stored and lost respectively, while \( P_m^{(\text{out})} \) is the mechanical power output, Eqs. (13)-(18).

\[
P_e^{(\text{in})} = V_d i_d + V_q i_q \tag{13}
\]

\[
P_e^{(s)} = L_d \frac{di_d}{dt} + L_q \frac{di_q}{dt} \tag{14}
\]

\[
P_e^{(i)} = R (i_d^2 + i_q^2) \tag{15}
\]

\[
P_m^{(i)} = J_m \dot{\omega}_m \omega_m \tag{16}
\]

\[
P_m^{(\text{out})} = B_{\text{tot}} \omega_m^2 + T_{\text{ext}} \text{sgn}(\omega_m) \omega_m \tag{17}
\]

As an example, table 1 reports the basic data of the EMA system referred to the aileron control surfaces.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
<th>Unit</th>
</tr>
</thead>
<tbody>
<tr>
<td>( R )</td>
<td>0.74</td>
<td>Ohm</td>
</tr>
<tr>
<td>( p_m )</td>
<td>1</td>
<td></td>
</tr>
<tr>
<td>( K_b )</td>
<td>0.4</td>
<td>N m / A</td>
</tr>
<tr>
<td>( L_d = L_q )</td>
<td>5e-3</td>
<td>H</td>
</tr>
<tr>
<td>( J_m = J_g )</td>
<td>8e-4</td>
<td>kg m²</td>
</tr>
<tr>
<td>( I_s )</td>
<td>8e-4</td>
<td>m / rad</td>
</tr>
<tr>
<td>( \tau_g )</td>
<td>5.5</td>
<td>-</td>
</tr>
<tr>
<td>( B_{\text{tot}} )</td>
<td>4e-3</td>
<td>N m sec / rad</td>
</tr>
<tr>
<td>( T_{\text{tot}} )</td>
<td>0.01</td>
<td>N m</td>
</tr>
<tr>
<td>( K_i )</td>
<td>9</td>
<td>V / A</td>
</tr>
<tr>
<td>( K_o )</td>
<td>0.2</td>
<td>A sec / rad</td>
</tr>
<tr>
<td>( K_x )</td>
<td>2e5</td>
<td>rad / (sec m)</td>
</tr>
<tr>
<td>( V_{\text{max}} )</td>
<td>270</td>
<td></td>
</tr>
<tr>
<td>( i_{\text{max}} )</td>
<td>30</td>
<td></td>
</tr>
</tbody>
</table>

**Table 1 – EMA system data (aileron control surfaces)**

**LANDING GEAR DYNAMICS AND KINEMATICS MODEL**

The same simplified architecture and the same extension/retraction kinematics are considered for both the nose LG and the main LGs (Figures 3 and 4). Each LG is represented by a generic rigid structure, which includes the fully extended shock absorber, bound to the A/C in the point \( C \). The extension / retraction is obtained thanks to a rotation about an axis whose direction is assigned within the input parameters of the landing gear. The LG rotation is actuated by an EMA acting on the LG structure, and bound to the LG and to the A/C structure by spherical hinges (points 2 and 3 in Figure 4).
The LG mass and the aerodynamic force are assumed to be concentrated in the point $H$. The latter is modelled as a drag force, with a constant drag coefficient (derived from [10][11]), not dependent on the LG stroke. For each LG (nose, left, right), the points $H$ and 3 are assigned through their coordinates on a LG-fixed reference frame $(C; X_{LG}, Y_{LG}, Z_{LG} – \text{Figure 5})$, while the origin $C$ of such a frame and the point 2 are assigned through their coordinates on the A/C body-fixed reference frame $(O; X_B, Y_B, Z_B, \text{Figure 6})$. The $X_{LG}$ axis of the LG reference frame coincides with the axis of rotation of the landing gear during extension and retraction. The axis $Z_{LG}$ has been chosen so that the $X_{LG}-Z_{LG}$ plane contains the point $H$.

The orientation of the LG reference frame is defined by the Euler’s angles with respect to the A/C reference frame (Figure 3).

The aerodynamic hinge moment on a generic control surface is given by:

$$H_{CS} = \frac{\rho U_{CS}^2}{2} C_{H_{CS}} C_{Sh} \alpha_{CS} \delta_{CS}$$  \hspace{1cm} (19)

where $\rho$ is the air density, $U_{CS}$ is the local airspeed magnitude, $C_{H_{CS}}$ is the mean aerodynamic chord of the CS considered as an isolated wing, $s_{f,CS}$ is the span of the flapped portion of the aerodynamic surface under consideration (wing, horizontal tail or rudder) and $C_{Sh}$ is the hinge moment coefficient. The last coefficient is assumed to be a function of the Mach number ($M$), the Reynolds number ($Re_{CS}$), evaluated by means of the mean aerodynamic chord of the aerodynamic surface containing the specific CS and the local airspeed, the local angle of attack ($\alpha_{CS}$) and the deflection of the control surface ($\delta_{CS}$).
The hinge moment coefficient \( C_{HC} \) depends both on incidence \( \alpha_{CS} \) and on control deflection \( \delta_{ACS} \) through relations that are generally non-linear. Typically, the flight control dynamics is studied with reference to small control deflection (<10°), where a linear relation can be used. This approach is not adequate for the purpose of this study, where extreme operations have to be examined. The envelopes of deflection of the aircraft control surfaces exceed the linear limits, thus the hinge moment model is based on the following assumptions:

a. the linear range with respect to the control deflection is: \( \delta_{ACS} \leq \delta^{\text{lin}} \)

b. out of the linear range, the hinge moment coefficient derivative \( b_{2ACS}(M, Re_{CS}) \) grows linearly with \( \delta_{ACS} \), and doubles its value for a control deflection equal to \( \delta^{\text{double}} \);

c. the contribution of \( \delta_{ACS} \) to the hinge moment coefficient, for high control deflections, is limited by a "saturation function", that is

\[ C_{HC} \leq K_{CS} \cdot \sin(\delta_{ACS}) . \]

By applying these assumptions, the evaluation of hinge moments for the control surfaces is provided by the following relation:

\[ C_{HC} = C_{H0CS} + b_{1CS}(M, Re_{CS}) \cdot \alpha_{CS} + b_{2ACS}(M, Re_{CS}, \delta_{ACS}) \cdot \delta_{ACS} \tag{20} \]

where \( b_{1CS}(M, Re_{CS}) \) is a constant coefficients evaluated as a function of \( M \) and \( Re_{CS} \), while \( b_{2ACS}(M, Re_{CS}, \delta_{ACS}) \) is evaluated

\[ \begin{align*}
 b_{2CS}(M, Re_{CS}) & \quad \text{for } |\delta_{ACS}| \leq \delta^{\text{lin}} \\
 b_{2CS}(M, Re_{CS}) \left[ 1 + \frac{|\delta_{ACS}|}{\delta^{\text{double}} - \delta^{\text{lin}}} \right] & \quad \text{for } |\delta_{ACS}| \geq \delta^{\text{lin}}
\end{align*} \tag{21} \]
In addition the product $b_{2\text{CS}} (M, \text{Re}_{\text{CS}}) \cdot \delta_{\text{hCS}}$ is limited by the following saturation function:

$$|b_{2\text{CS}} (M, \text{Re}_{\text{CS}}) \cdot \delta_{\text{hCS}}| \leq |K_{\text{sat}}^{\text{CSh}} \cdot \sin(\delta_{\text{hCS}})|$$  \hspace{1cm} (22)

The coefficients depending on the Mach number and the Reynolds number have been calculated by means of the ESDU method [8][9] on the basis of the Net equivalent wing geometry.

Figure 8 shows a diagram of the variation of the hinge moment coefficient ($C_{\text{hCS}}$) with respect to control deflection ($\delta_{\text{hCS}}$) for a given Mach number and a given incidence of the control surface ($\alpha_{\text{CS}}$).

**Simulation Tests**

Two simulation tests are presented in this section, as examples of assessment of LGS and FCS power flows in severe operative conditions. Concerning the LGS simulation, the test (Figure 9) is characterized by an extension phase and a retraction phase, performed in levelled flight conditions, at an A/C speed of 70 m/sec and an A/C load factor of 2.5. During the extension phase, the nose LG EMA retracts, while the EMAs of the main LGs extend.

Referring to FCS simulation, the commands are provided in levelled flight conditions, at A/C speed of 100 m/sec, A/C load factor of 2.5 and with zero angle-of-attack and sideslip. The command deflections are characterized by large amplitudes and speeds (figure 12) and, for each CS, a motion inversion is imposed when the actuators achieve the maximum deflections (maximum hinge moments). In addition, such commands have been built in order to synchronize the control surfaces power peaks, thus to have an assessment of the maximum FCS power requests.

Even if it could be argued that these tests are referred to excessively severe conditions (e.g. contemporary extreme deflections on all CS), this approach provides an estimation of the maximum system power flows, by using the systems specifications only.

**Results**

Figure 10 report the time histories of power flows for nose LG and left main LG during the extension and retraction phases. The time history of the actual EMAs stroke is shown in Figure 9, together with the commanded ones. For the considered LGS configuration, the nose LG EMA works in brake mode (negative mechanical power) during all the extension phase and in motor mode during all the retraction phase. The behaviour of the main LGs is slightly different because the EMA works in motor mode during the last part of the extension phase and in brake mode during the first part of the retraction phase.

In the test, the total maximum LGS power absorption ranges about 2.5 kW, and a similar value is generated by the EMAs in brake mode (to be dissipated or available for regeneration). Considering that the LGs move at constant speeds, the power flows variations can be interpreted by using the results in terms of applied forces (Figure 11). For the nose LG, weight and drag contribute in equal measure, while for the main LGs the weight contribution is predominant. For all LGs, the inertial contributions demonstrate to be significant with respect to the other forces.

Figure 13 shows the power flows for each CS of the FCS. The models evaluate identical power flows for EMAs driving the CSs located symmetrically with respect to the A/C longitudinal plane (e.g. left and right ailerons), for the same commanded deflections. For this reason, all the power flows of figure 13, except those referred to rudder (single surface) are the sum of equal contributes. In addition, it is relevant to observe that the aerodynamic interaction among the different CS is neglected, while the flaps deflection modifies the total lift coefficient that affects the down-wash angle and the Elevators aerodynamic.

As for the LGS test case, Figure 13 highlights the phases in which the EMAs work in brake mode or in motor mode. In the test, the total maximum FCS power absorption ranges about
20 kW, and this value can be considered as a reference also for the maximum generated power.

**CONCLUSIONS**

The developed models, referred to an aircraft with tricycle landing gear system and a flight control system with nine control surfaces, are used within the Clean Sky GRA project as sub-models of a Shared Simulation Environment, which is a tool able to support the design and the validation of the electrical energy management strategies.

The dynamics and kinematics models of each flight control surface take into account inertial, weight and aerodynamic hinge moment contributions. Similarly, each LG has been modeled as a rigid structure and the forces transferred to the EMAs are evaluated through models that take into account the load weight, aerodynamic forces and the acceleration effects due to A/C manoeuvres and load motions.

The tests are referred to severe operating conditions (e.g. landing gear activation at high load factor, flight controls commanded with large-amplitude/high-speed demands and motion inversion at maximum deflections), to identify the maximum electric power flows. In terms of total maximum power absorption, the FCS requires about 20 kW, while the LGS power rating is about 2.5 kW. Similar values are generated by the EMAs when working in brake mode (to be dissipated or available for regeneration).

The analysis highlighted that the electric power flows of both landing gears and flaps are essentially dominated by the mechanical power output, since the EMA motions are characterised by low speed/high torque cycles. On the other hand, the electric power flows of primary flight controls (ailerons, elevators, rudder, spoilers) also depend on transient mechanical power, producing peaks when high speed motions are required to the EMAs.

**ACKNOWLEDGEMENTS**

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**REFERENCES**


Figure 9 LGS position dynamics

Figure 10 LGS power flows (same results for right main LG)

Figure 11 Force applied to LGS

Figure 12 FCS position dynamics

Figure 13 FCS power flows