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Experimental characterisation of flutter and divergence of 2D wing section with stabilised response

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ABSTRACT

The first experimental application of the Parametric Flutter Margin method for identification of aeroelastic instabilities is presented. The experiment was performed in two steps using a two degree-of-freedom wing segment mounted in the wind tunnel. First, the reference flutter and divergence conditions were found by increasing the free-stream velocity until the observed response diverged. Then, the system was stabilised according to the Parametric Flutter Margin methodology, and the flutter and divergence conditions of the original test model were identified positively while being in a stable regime demonstrating excellent agreement with the reference instability conditions. Although the new experimental methodology is not model based, the results were compared with a theoretical model showing good agreement as well. The acquired data demonstrates both the accuracy of the Parametric Flutter Margin method as well as its capability to test for aeroelastic instabilities, both flutter and divergence, in stable and predictable testing conditions.

1. Introduction

Flight testing, used to prove that the aircraft flight envelope is flutter free, is a risky task. The flutter boundary is cautiously approached by gradually increasing the flight speed until the flight envelope is reached or a damping coefficient reaches the 3% threshold [1]. Meanwhile, the aircraft response to various sources of excitation like atmospheric turbulence or control surface deflections is continuously monitored and analysed. In some cases, such as explosive flutter, damping might suddenly rapidly decrease. Hence flutter might be encountered by accident causing severe damage to the aircraft. Consequently, such tests are accompanied by numerous numerical analyses, wind tunnel and ground testing to avoid bringing the tested aircraft too close to the flutter boundary by accident [1].

Various flight-test data-analysis methods are available for application in on- and off-line manner to identify the flutter conditions. Among others: damping extrapolation [2], envelope function [3], the Zimmerman–Weissenburger flutter margin [4], the model-based flutterometer method [5], and using a discrete-time autoregressive moving average (ARMA) model [6]. Operating at flutter conditions might have catastrophic results. Therefore, all the approaches rely, in one way or another, on extrapolation to predict the flutter conditions while staying at safe flying conditions, which makes the tests expensive, time-consuming and risky. On the contrary, the recently developed Parametric Flutter Margin (PFM) method [7] is based on analysing frequency-response functions (FRFs) at and beyond the nominal flutter onset conditions, but with the system modified such that it is actually stable. This allows us to identify flutter positively without exceeding the pre-determined safe vibration levels. It is anticipated that the PFM methodology will be very instrumental in the design of future flutter-test campaigns improving their safety and reduce the time and effort required to ensure that the flight envelope is indeed flutter free.

Karpel and Roizner [8] proposed a novel method for finding the flutter boundary experimentally based on their numerical PFM method [7]. The experimental PFM mitigates some of the deficiencies of the currently established methods, namely the need to approach the flutter boundary cautiously, and the fact that the flutter boundary is never positively identified unless erroneously encountered. The PFM method is based on the idea that the stability point of an aeroelastic system can be offset by adding a stabilising element. In the case of wing flutter, such a stabilising element could be an added mass at the leading edge of the wing tip. Such an augmented system is then subjected to harmonic excitation to obtain the FRF of the stabilising element, for instance, the acceler-

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Nomenclature

$[A(i\omega)]$	PFM system matrix
$\{B_f\}$	Distribution vector, $\{B_f\} = [1, x_s]^T$
$C(k)$	Theodorsen function
C_0	Airfoil centre of gravity
$[C_f(i\omega)]$	Acceleration sensor, $[C_f(i\omega)] = -\omega^2[1, x_s]$
C_s	Location of the stabilising mass
F_{ih}	Force measured by the impedance head
F_s	External excitation force
$H(\omega; v_0)$	Frequency response function
J_0	Airfoil moment of inertia around C_0
L	Lift
M_0	Constant torque applied in divergence test
$M_{c/4}$	Aerodynamic moment about point Q
P	Airfoil hinge point
Q	Quarter-chord point
$Y_f(\omega)$	Gain function
a	Non-dimensional location of P, $P = (1 + a)b$
a_1, a_2	Acceleration measured by Accelerometer 1 and 2
a_{ih}	Acceleration measured by the impedance head
a_s	Acceleration of the stabilising mass
b	Airfoil semi-chord, $b = c/2$
c	Airfoil chord length
d_h	Heave damping
d_θ	Pitch damping
e_0	Non-dimensional location of C_0, $C_0 = (1 + e_0)b$
e_s	Non-dimensional location of C_s, $C_s = (1 + e_s)b$
h	Heave DOF
h_{LVDT}	Heave DOF measured by the LVDT
k	Reduced frequency, $k = \omega b/v_0$
k_h	Heave stiffness
k_s	Stiffness of the extensional spring
k_θ	Pitch stiffness
k_θ^s	Stabilising torsional stiffness
m_*	Support mechanism mass
m_0	Airfoil mass
r	Dimensionless radius of gyration about P, $r^2 = (J_0 + m_0 x_0^2 b^2)/(m_0 b^2)$
r_p	Pulley radius
r_p^s	Pulley radius in divergence test for stabilisation stiffness
r_{p1}	Pulley radius in divergence test with wind-on conditions
t	Time
u_f	Excitation input
v_0	Freestream velocity
v_d	Divergence speed
v_f	Flutter speed
x_0	Eccentricity of the airfoil section, $x_0 = e_0 - a$
x_s	Eccentricity of the stabilising mass, $x_s = e_s - a$
y_f	Acceleration in heave DOF at C_s
$\zeta(i\omega)$	Vector of DOFs, $\zeta(i\omega) = [\xi, \theta]^T$
η_θ	Normalised damping coefficient of the pitch DOF, $\eta_\theta = d_\theta/(m_0 b^2)$
η_ξ	Normalised damping coefficient of the heave DOF, $\eta_\xi = d_h/(m_0 b)$
θ	Non-dimensional pitch DOF
θ_{RVDT}	Pitch DOF measured by RVDT
θ_s	Pitch DOF deflection due to M_0
μ_*	Support to section mass ratio, $\mu_* = m^*/m_0$
μ_s	Stabilising to section mass ratio, $\mu_s = m_s/m_0$
ξ	Non-dimensional heave DOF, $\xi = h/b$
ρ	Air density
$\Phi_f(\omega)$	Phase function
χ_θ	Force in the pitch DOF
χ_ξ	Force in the heave DOF
ω_{pco}	Phase cross-over frequency
ω_θ	Circular frequency of the pitch DOF, $\omega_\theta = \sqrt{k_\theta/J_0 + m_0 x_0^2 b^2}$
ω_ξ	Circular frequency of the heave DOF, $\omega_\xi = \sqrt{k_h/m_0}$
Æ	Wing aspect ratio
ARMA	Autoregressive moving average
CG	Centre of gravity
DOF	Degree of freedom
FFT	Fast Fourier transform
FRF	Frequency response function
LVDT	Linear variable differential transformer
PFM	Parametric flutter margin
RVDT	Rotary variable differential transformer
SDOF	Single degree of freedom

ation of the added mass which is then analysed for gain margin at phase-cross-over (pco) frequency. The flutter boundary of the original system excluding the added stabilising mass is reached when the gain margin of the stabilising element equals 0 dB. The FRF analysis is repeated at various flight conditions to obtain the gain margin vs flight speed characteristics. The flutter speed is read from the graph at 0 dB. Details on the theoretical foundation of the PFM method and its formulation are provided in Roizner and Karpel [7] while the key equations and their application related to this experiment are outlined in this paper.

It is worth pointing out that the PFM method allows for the flutter boundary identification of the original system excluding the added mass while the augmented system remains stable. This greatly reduces the risk of such experimental efforts.

The contribution to the state of the art of this paper is a proof of concept and validation of the proposed PFM method using a typical wing section with pitch and plunge degrees of freedom (DOF) mounted in the wind tunnel. The paper is organised as follows: in Sec. 2 the mathematical formulation of the 2DOF aeroelastic system along with its PFM implementation related to the experiment is presented, Sec. 3 describes the experimental setup and the testing procedure. The results are shown in Sec. 4, and the conclusions of this work are given in Sec. 5.

2. Theoretical model of the aeroelastic system

The mathematical formulation of the 2DOF airfoil along with its PFM implementation relevant to this experiment is presented in this section. The mathematical model had three main purposes. First, to configure the experimental setup to obtain the aeroelastic instability at a velocity within the wind tunnel capabilities. Second, to size and position the stabilising weight such that the flutter velocity would increase by at least 15%, and the third purpose was the comparison with the experimental results. It has to be stressed, however, that the experimental PFM method or its results do not depend on the application of this mathematical model. The experimental PFM method is not model-based and does not require any mathematical model of the aeroelastic system to identify the nominal flutter conditions if they exist in the test velocity range.

First, the governing equations of motion are presented, followed by the presentation of the PFM methodology.