Flight control system damage simulation using freely flying models

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ABSTRACT

This article deals with a damage simulation of flight control systems during a flight. As a subject of flight tests the model of a classic aerodynamic aircraft scheme is considered. This aircraft has highly swept wings a moderate wing elongation and all-moving horizontal tail surfaces. The diagrams of flight parameters from flight tests of the freely flying models with loss of a control surface in flight are given. The investigation results for the failures leading to a sharp drop in the stiffness of the control units in the pitch channel are given. The damage investigation in horizontal flight and in complex manoeuvres is given.

NOMENCLATURE

$G_{\dot{q}}, G_{\vartheta}, G_{\bar{\vartheta}}$	gains of Flight Control System (FCS) by pitch rate, pitch angle and by integral of pitch angle respectively.
K_L	length scale coefficient
L	length, m
M_T	torque moment
$M_{\perp}$	bending moment which acts at right angle to the flatness of chord
$M_{\|\|}$	bending moment that act in the flatness of stabiliser chords
Ni	acceleration, ms^{-2}
q	pitch rate, deg/s
t	time, s
V	velocity, ms^{-1}

Greek letters

α	angle-of-attack, deg
δ	surface deflection, deg
ϑ, γ, ψ	pitch, roll and yaw angles respectively, deg

Subscripts

a	aileron
H	altitude
M	model
N	aircraft (natural)
r	rudder
st	stabiliser
x,y,z	body frame axes index

Superscripts

$\bullet$	derivative with respect to time

Paper No. 2898. Manuscript received 8 July 2004, revised version received 14 October 2004, accepted 29 November 2004.

https://doi.org/10.1017/S000192400000052X Published online by Cambridge University Press

1.0 INTRODUCTION

The Flight Control System (FCS) flight test is often the final phase of the test cycle and its aim is to evaluate the FCS's actual behaviour after functional degradation of one or several elements. Aircraft motion in emergency situations displays a complex three-dimensional nature and may be accompanied by energetic manoeuvres, large angles-of-attack and the appearance of flow separation areas.

The computation of such flight regimes is highly complicated and the mathematical model developed for this purpose always requires experimental validation. But, testing of the FCS in real conditions is a costly task and presents a risk for test pilots since it must be conducted with the presence of both the actual control system and the prototype aircraft.

The limited dimensions of wind tunnels and the presence of the model's fixing attachments inside a wind tunnel do not allow the reproduction of all features of an aircraft in an emergency flight simulation, the proper interaction of the aircraft's manoeuvres with its induced aeroelastic deflections, and with structural vibrations.

On the other hand utilising freely flying models is a highly effective method for the investigation of the flight dynamics of an aircraft with partial FCS failures and certain control surface damage. Dynamically similar freely flying models (DSM) have been utilised in practice in all steps of the process of generation of a new aircraft, for which basic data and performances completely differ from previous designs[1,2,3]. This method has considerable advantages: Firstly, is the unlimited space that surrounds the model (it excludes the interference between the walls and boundary layer of the flow inside a wind tunnel). Secondly, is the absence of suspension and attachment which distorts natural oscillations of the model and introduces additional masses and friction. Last, but not least, the possibility of investigating the most dangerous regimes of flight before the first prototype of the aircraft under development is created, as well as saving large investments and time for the construction of unique installations (such as special wind tunnels). The flutter research on manoeuvre were conducted by NASA within the HiMAT programme, but in the release the experiments with changes of the aeroelastic characteristics connected to damages of a control-surface are not mentioned[4].

The given work had three main purposes:

1. The first is to develop a special tool for the aircraft damage simulation. In other words, it deals with the research of the problem concerning the capability of practical implementation of devices for imitating damage on DSM.
2. The second aim is to improve technique of the aircraft damage simulation by means of DSM. This purpose includes: injection of the DSM to a pre-programmed flying path; implementation of the given manoeuvre; definition of the moment of the beginning of aircraft damage simulation; checking of procedures for prevention of irreversible DSM damages and estimation of a capability of DSM reuse for aircraft damage simulation.
3. The third purpose is to carry out damage simulation of the particular aircraft and to draw the conclusion about critical damages.

2.0 DSM FLIGHT TEST TECHNIQUE

A DSM is an autonomous, pilotless, research, multiple mission aircraft. It may perform automatic or remotely controlled flights, according to a given program and has the capability to record all flight relevant information. The DSM is geometrically similar to the full-scale aircraft under investigation. Its mass, moments of inertia and other parameters ensure exact presentation of the aircraft performances; including cases when an adequate simulation of surface´s aeroelastic behaviour is required. DSMs are launched from a carrier (aircraft or helicopter) or from a ground-based launcher with the help of a solid-propellant ascent stage as shown in Fig. 1.

After separation of assisted take-off unit, the DSM accomplishes pre-programmed manoeuvres and in the case of an emergency or after completion of the flight program a recovery parachute and soft landing system assure a safe return.

The existence of necessary and sufficient conditions of similarity of the FCS of an aircraft, and that of its DSM has been proven[5]. The generalised scale coefficients for applying the aircraft FCS's gain factor, to those of the FCS of the DSM have been obtained, and it has also been shown that the coefficients depend only on the linear scale of the DSM.



Figure 1. DSM with solid-propellant ascent stage.

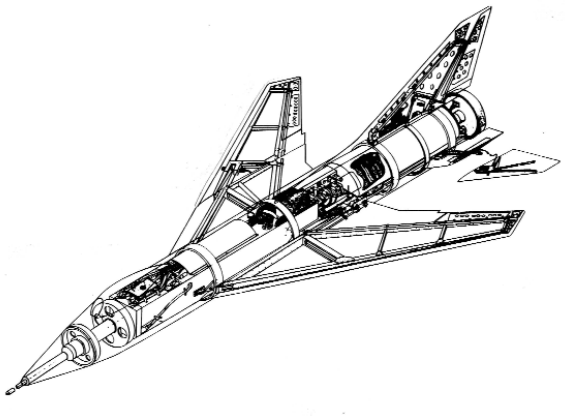


Figure 2. Diagram of the DSM structure.

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Figure 3. DSM structure with the 75% stabiliser loss.

For example, if the control law for a longitudinal zero-constant-error system is taken as follows,

$$\delta = G_q \cdot \dot{\vartheta} + G_{\vartheta} \cdot \vartheta + G_{\bar{\vartheta}} \cdot \int \vartheta \mathrm{d}t \qquad \ldots (1)$$

the relation between gains of the model's FCS (subscript '*M*') and of the aircraft (subscript '*N*') follows as:

$$G_{qM} = \sqrt{K_L} G_{qN}; \quad G_{\theta M} = G_{\theta N} \; ; \quad G_{\bar{\vartheta} M} = \frac{1}{\sqrt{K_L}} G_{\bar{\vartheta} N} \qquad \ldots (2)$$

where $K_L = L_M/L_N$ is the length scale coefficient.

This allows to simulate the FCS of the aircraft by means of the FCS of the DSM, and to investigate faults and damages of sensors, servos and control surfaces, the change of the control laws and of its gains. Figure 2 shows the diagram of the DSM structure of the Sukhoi-7 aircraft.

This particular model, which was designed and fabricated in the Physical Simulation Research Institute affiliate of National Aerospace University of Ukraine, had the following principal characteristics: length 3·36m; wing span 1·78m; mean aerodynamic chord 0·8m; mass 156kg; range of model's centre of gravity position from 29·5 to 38·1%; test speed from 30 to 150ms^{-1}.

Damage simulation of a horizontal control surface is the most complex regime from the point of view of DSM flight test techniques. In flight tests the investigation of the subcritical and supercritical damage is necessary. If the damage is subcritical and the overload of a disturbed movement does not exceed of the destructive limit of the structure it is harmless for the DSM.

If the damage is supercritical, the model loses stability and controllability in a pitch angle. In this case aerodynamic load is significantly increased and can cause destruction of a model.

Flight experiments researching the flight dynamics and controllability of the aircraft with all-moving tail surface damage were done by means of a model of the Sukhoi-7 aircraft. The structure of the all-moving tail surface was developed in two variants. First, it was used for damage simulation of the tail with small cross-sectional height of profiles. In this case the separation of a surface was by means of a linear hollow charge. The second variant of the structure was used for modelling 75% stabiliser loss.

In this case the cross-sectional height allowed small pyrotechnic charge to be used and the application structure to be reusable. Figure 3 shows the DSM structure with the 75% stabiliser loss.

Modelling such dangerous modes as damage of a horizontal control surface required the creation of a special system of emergency stabilisation as an additional part of the control system.

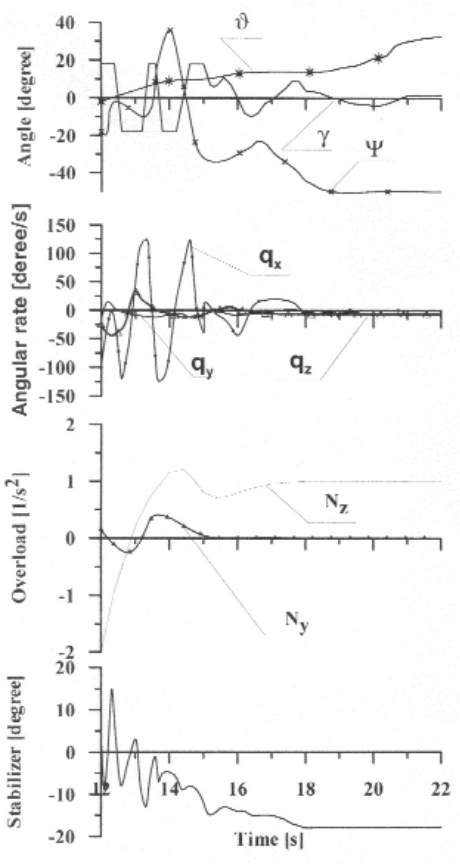


Figure 4. Parameters of horizontal model flight with simulation of a 75% stabiliser loss.

This system provides a change of the FCS control laws and in addition defines a mode of experiment start and the time of its emergency interruption. If the experimental flight follows the program and the longitudinal acceleration reaches the programmed magnitude, the 'fire' signal for separating the control surface is applied. Simultaneously the system follows the limit magnitude of acceleration Nz, and in the case of the first level overshoot the system provides the change of the FCS control laws (gain change). In case of second level overshoot ($N_{Z1} \leq N_{Z2}$) the system sends a command pulse for stabilising-chute ejection.

3.0 FLIGHT TEST RESULTS

The test flights were carried out by the following technique. The amount of stabiliser damage was identical for all flights. Subcritical and supercritical regimes were defined in the experiments. The flight programmes of the experiments are described below.

3.1 Experiment No 1. 75% stabiliser loss. Regular overload

The most typical experiment in subcritical regime was the flight, characteristics of which are presented on a Fig. 4.

In this case the flight program included the following stages:

a. Take-off from a ground-based launcher by means of a solid-propellant ascent stage. The inclination of a trajectory is equal to 45 degrees.

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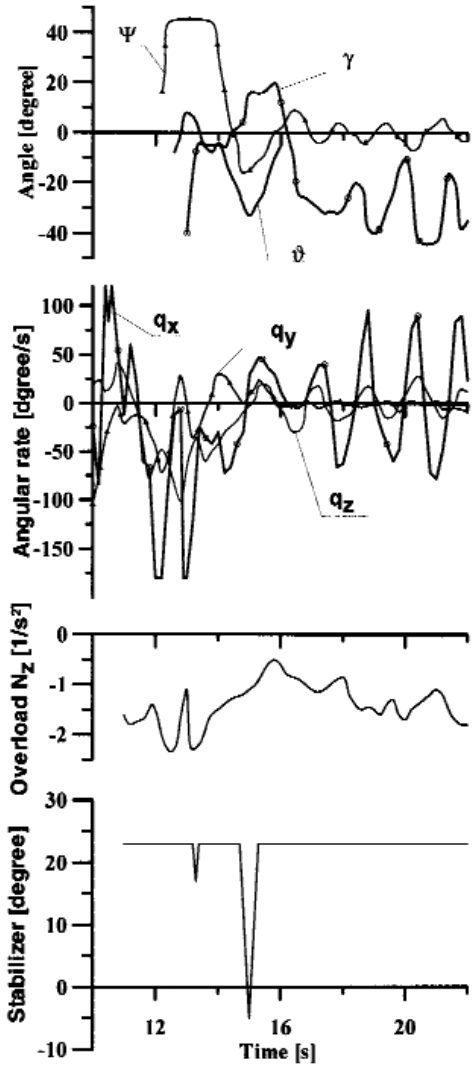


Figure 5. Parameters of model flight with $N_z = 2{\cdot}3$ and imitation of a 75% stabiliser loss.

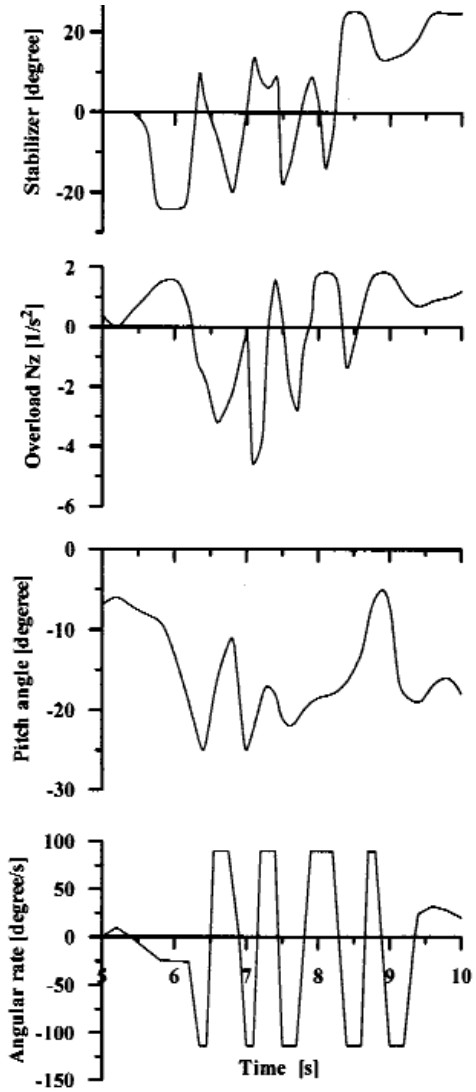


Figure 6. Low frequency process.

b. Separation of assisted take-off unit (7th second of flight);
c. Manoeuvre for transition in to horizontal flight. The stabilisation of the initial conditions of the experiment. (10·4 – 12·4s);
d. Energising of the imitation damage device (14s);
e. Energising of the parachute landing system (26s).

The separation of the 75% stabiliser's half span is carried out at 14·5 seconds in to the flight (flight altitude is 67m and speed of the flight is about 266km/hour). The longitudinal manoeuvre was completed and, during 0·8s, the magnitude of normal acceleration Nz is in an interval from +1 up to +1·2. The all-moving horizontal tail surface supports a given flight trajectory and was moving in an interval from –10° up to –5° (at the moment of damage imitation the stabiliser was in a position $\delta_{st} = -9°$).

The perturbation in lateral motion (roll angle γ) which appeared on the manoeuvre for transition in to horizontal flight is not yet completely finished.

The roll angle is equal to about +18°. The horizontal flight of the model after damage simulation is stabilised with an insignificantly modified heading (ψ). The normal acceleration is reduced up to 0·7 and then, after 2·6s, returns to assigned value $Nz = +1$ and conserves this value up to the end of flight.

The stabiliser was deflected to the maximum angle for conservation of a horizontal flight regime. The change of angular velocity q_y was insignificant (from –10°/s up to +4°/s). The rudder (δ_r) was deflected completely for compensation of a disturbing moment, which was caused by the damage. The magnitude of angular velocity q_z was insignificant (about +7°/s). The perturbations in the roll angle were damped. The deflection angle of aileron (δ_a) which was necessary for the model balance, did not exceed 18° and angular velocity q_x was limited to 100°/s magnitude

The results obtained were confirmed in several flights. On the basis of these results, it is possible to reach the following conclusions; the investigated damage of the stabiliser (in flight with $Nz \cong +1$) is subcritical. The model keeps initial steady-state flight. It means that the simulated aircraft is capable of counteracting the modification of aerodynamic characteristics by means of the undamaged part of the control surfaces and it can continue the flight on a given trajectory.

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3.2 Experiment No 2. 75% stabiliser loss. High overload

The aim of the following flight tests was data generation about the flight dynamics of an aircraft with the same damage as in the previous experiment, but at an instant of time of energetic manoeuvre in the longitudinal channel. In Fig. 5, the flight characteristics in supercritical regime are shown.

In this case the flight program included the following stages:

a. take-off from a ground-based launcher by means of a solid propellant ascent stage. The inclination of a trajectory is equal to 45 degree.
b. separation of assisted take-off unit (6th second of the flight);
c. start of a pitch angle manoeuvre (From regime $\vartheta = +45°$ to pitch-down. Negative acceleration). (9th s);
d. Unlocking of the imitation damage device (9th s);
e. Termination of the manoeuvre with negative acceleration, transit to manoeuvre with positive acceleration (deflection of the stabiliser in an extreme position $\delta_{st} = +20°$);
g. Energising of parachute landing system (26th s).

The damage imitation device was adjusted for two thresholds of acceleration. The first threshold was equal $Nz \geq 1{\cdot}7$ and corresponded to the beginning of damage simulation. The second was just the same but with $Nz \geq 4{\cdot}9$ which corresponded to the instant of the emergency stabilisation system energising. The imitation of damage was at 12·6 seconds with acceleration $Nz = -2{\cdot}3$. After that, the model carried out a rotation to a roll angle equal to 180° (on the Fig. 5 the value of 0 degrees corresponds to 180 degrees for roll angle) and continued the flight in this situation (inverted flight) until the parachute was deployed.

The model flight trajectory was near to straight flight with altitude reduction. The stabiliser was deflected to the maximum angle for diving. The magnitude of normal acceleration, which was near to 1·0, and its fluctuation was caused by a change of the roll angle (axis of the sensor sensitivity changes the attitude with respect to gravitation direction). The cause of a long-duration inverted flight could be the high stability of the aircraft with a positive V aerodynamic scheme. The change of angular velocity q_x was significant and hunting of the DSM roll angle was increased up to 100°/s, but the model did not come back to a normal attitude. The model flight time in the inverted position was just the same 10 seconds that corresponded to 23 seconds of aircraft flight.

The data received in this flight showed, that with 75% damage to the stabiliser half span and with acceleration $Nz \cong 2$ the model cannot keep the initial flight mode.

3.3 Experiment No 3. Stiffness decreasing of the control units in the pitch channel

As a second characteristic example of a DSM flight test, it is possible to consider the results of investigating the failures leading to a sharp drop in stiffness of the control units in the pitch channel (with the preservation of friction couplings, installed on the shafts of an all moving stabiliser)[6].

A special attachment for stabiliser's surfaces with elastic similarity was developed for the investigation of the aeroelastic processes caused by feathering of the all-moving stabiliser during severe pitch manoeuvres. This attachment is installed on the sides of the fuselage by means of arms and beams equipped with self-aligning spherical bearings. The stabiliser's shafts pass through these bearings and have inside the fuselage a rectangular cross-section that act as elastic elements, providing a given bending rigidity in the chord's plane. Close to the spherical bearing on the shaft a second elastic element is attached. Its deformations provide torsional rigidity of the control surface. This elastic element through a control rod and crank is attached to the servo drive of the all-moving stabiliser. Moreover, a strong elastic element with a 'V' shape is attached to the end of the shaft. The deformations of this last element provide rigidity of the surface attachment in the plane perpendicular to the chord.

Thus, the system of flexible elements provides a given rigidity of the stabiliser's surfaces and allows the investigation of the special form of 'flutter on manoeuvre' that is characterised by intensive oscillations of the all-moving stabiliser in all its degrees of freedom (including oscillations in the chord plane). This kind of flutter also has interdependence between the loss of aeroelastic stability and the aircraft's manoeuvres. Mechanical oscillation amplitude limiters, as well as a system of emergency flutter suppression avert the danger of excessive deformations that may break the airframe.

The DSM flight tests allowed investigation of flight stability up to a speed of $130 ms^{-1}$ with the simulation of control unit damages in the pitch channel. It also allowed study of the stability of elastic oscillations of the stabiliser with a pre-established rigidity of its joint.

It was determined that during a simulation two aeroelastic processes take place:

a. low frequency phenomena, and
b. high frequency oscillations on determined phases of the same low frequency phenomena.

3.3.1 Experiment No. 4. Low frequency phenomena

The low frequency process represents oscillations of both surfaces of the all-moving horizontal stabiliser, around of position that was predetermined by the flight control system.

An example of flight data is presented in Fig. 6. The rigidity of the control drive was diminished 17·3 times. The flight experiment was initiated at a velocity head of $8{,}830 N/m^2$ (velocity of the model is $120 ms^{-1}$). The cause of low frequency oscillations was the disturbance of the acceleration that went from 0 to +2 during 0·7s.

The lift force of the stabiliser increased respectively (maintained by servo-drives in a trim position). The hinge axis of the stabiliser is situated between subsonic and supersonic aerodynamic centers of the console.

In range of subsonic speed the increase in stabiliser lifting force caused an additional torque around the hinge that in turn caused further deflection which then produced yet more lift force. The servo drives could not prevent this tendency after simulation of rigidity loss in spite of the fact that the stabiliser was deflected almost to its limit. Only during the DSM manoeuvre could the low rigidity servo drives deflect the stabiliser to the initial position, because the sharp change of angular velocity q_y, (from 15deg/s to 100deg/s) and more important, a change of load factor Nz (from + 2·1 to –2) diminished the lift force of both surfaces.

The process of stabiliser reversal from one limit to the other continued up to the end of test flight. In the discussed test flight four cycles of low frequency processes of the stabiliser reversing were registered.

It is typical in normal flight that the output signal of the pitch sensor (without failure simulation) is an input signal for FCS and parameter δ_{st} (angular deflection of stabiliser) with some delays follows changes of pitch angle. But in the case of simulation of servo drive rigidity loss, the stabiliser is practically deprived of a control action from its servo drive; it carries out the reversal oscillations from limit to limit. In this case the DSM oscillations follow the stabiliser oscillations with little delay. This can be seen from the pitch angle ϑ, the angle-of-attack α and the normal load factor N_z variations. It proves the fact that the oscillation of the feathering stabiliser is the exact cause of the model's manoeuvres.

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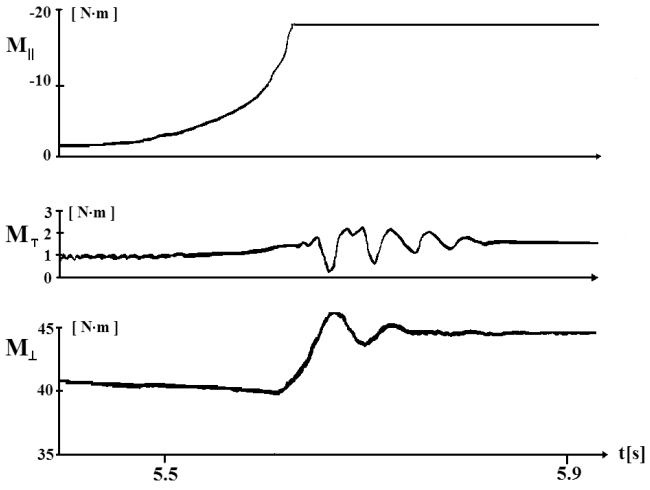


Figure 7. High frequency process.

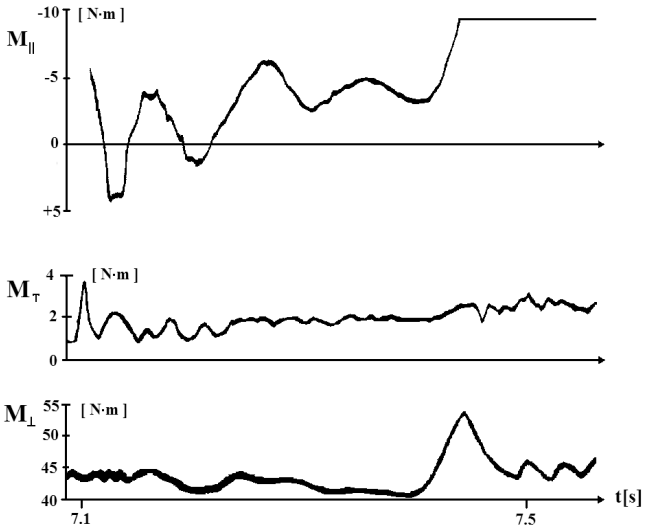


Figure 8. High frequency process of elastic oscillations.

3.3.2 Experiment No 5. High frequency phenomena

A high frequency process of elastic oscillations was registered in flight tests at certain phases of the low frequency process. The typical episodes of the beginning of such kinds of oscillations are presented on Fig. 7 (flight from 5·38s to 5·93s) and Fig. 8. (flight from 7·08s to 7·59s) in greater scale. In the flight test from 5·55s to 5·75s (see Fig. 6) the stabiliser achieved the extreme position ($\delta_{st} = -25$) and the lifting force of one of its blades achieved 240 Newtons. Flutter appeared. Its progress had an intense character. In the first period of oscillations the bending moment that acts in the flatness of stabiliser chords ($M_{||}$) achieved the maximum allowed value. The stabiliser's structure destruction (usual for real aircraft) was prevented on the model by the work of limiter of elastic oscillations.

One more typical episode of the flutter is presented on Fig. 8. At 7·46s of the flight the stabiliser with diminished drive rigidity was deflected by airflow to an angle of –18·5° during pitching.

In this case, the bending moment $M_{||}$ that acts in the flatness of stabiliser chords was changing not so intensely as in the previous flight and its oscillatory character is precisely presented in Fig. 8. The same oscillatory character has the bending moment $M_{\perp}$ which acts at a right angle to the flatness of chords, and torque moment M_T. The oscillation frequencies of the stabiliser are close to the frequencies found during ground tests.

The cause of the stabiliser's flutter is the energetic manoeuvre of the model, which in turn was caused by the deflection of the feathering stabiliser. The angular velocity of this manoeuvre changed from 100 to –130 deg/s, and the load factor change was from –4·4 up to +2.

The reliability of the results obtained was proved by their repetition. In a given DSM flight test five cases of initialisation of the high frequency flutter-nature oscillations were registered. The data obtained allowed the conclusion that the simulated damage is critical for an aircraft and does not permit continued flight.

4.0 CONCLUSIONS

These investigations and tests have shown the advantages of researching using DSM for adequately representing aircraft flight with loss of a control surface part in interaction with the most complex manoeuvres.

The three main purposes of this work were successfully achieved. DSM with the special device for simulating damage (partial loss of a control surface) and flutter (reduction of rigidity of the drive) was developed. The technique of aircraft damage simulation by means of DSM was developed and tested. The design philosophy of the special systems for damage simulation used to protect the structure of control surface from destructions has allowed executing all flight test program on a single copy of DSM.

The damage simulation of the particular aircraft was carried out and conclusion about critical damages was drawn. The simulation showed that the investigated damage with 75% damage to the stabiliser half span in flight with $Nz \cong 1$ is subcritical and with acceleration $Nz \cong 2$ is supercritical. In a given DSM flight test five cases of initialisation of the high frequency flutter-nature oscillations were registered. The obtained data allow to draw the conclusion that after reducing the rigidity of the control driver by a factor of 17·3 the aircraft could not continue the flight, the damage is supercritical.

The physical modelling of a FCS by means of large scale dynamically similar free-flying models has a long-range perspective, which allow for a significant increase in the quality of design and development of new types of aircraft, and their FCS; with the advantage of the physical model better approximating the natural object being simulated.

The results show the possibility of physically modelling the dynamics of flight, taking into account all peculiarities of aerodynamics, possible damage, as well as real atmospheric conditions. In addition, the unique data obtained in flight tests can be used for the validation of mathematical models for calculation of all classes of emergency situations of the same nature.

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