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STABILITY ROBUSTNESS CRITERIA AND REDUNDANCY MANAGEMENT OF AN ACTIVE VIBRATION ISOLATION SYSTEM

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1 Abstract

An active control technique has been developed for the elimination of the helicopter vibrations. Potential applications of this technique, which has been tested in an experimental rig, also exist.A control strategy and an on - line identification algorithm have been developed with stability robustness criteria in order to implement the vibration isolation system.

Redundancy management is very important, because it will make the active control system for the vibration elimination ultra - reliable. Three planes of fault detection, isolation and reconfiguration are suggested. These planes include a generalised likelihood ratio test, majority voting algorithms and electrohydraulic servovalves and actuators with adequate hardware redundancy. Such a system will minimise also any problems related to the airworthiness issue.

2. Introduction

The helicopter vibration problem is very important and an active control technique was developed in order to reduce the rotor - induced vibrations[1]. Some other applications of this technique include vibration control of submarines and military aircraft. The analytical and simulation results have shown 100% vibration isolation. The simplicity and the effectiveness of this technique were successfully demonstrated on an experimental rig, where vibration reduction levels of up to 80% were achieved.The control method and the identification algorithm proved to be robust with respect to various system perturbations i.e. system parameter changes. Stability robustness criteria were set for the vibration isolation at the blade passing frequency and its harmonics. These criteria provided the necessary conditions for the on - line convergence of the parameter identification algorithm.

But still the performance of the system depends on the normal (error free) operation of several other factors. Problems created by these factors include sensor failures, loss of hydraulic power, actuator hardover or even one or more actuators out of condition. Therefore the redundancy management is imperative for the normal operation of an active vibration isolation system. Its objective is to ensure that an ultra - veliable active control system for the vibration reduction is achieved. The redundant information i.e. sensors, microprocessors, controllers, filters can be provided by two extra channels for each isolation unit used. Three planes of fault detection, isolation and reconfiquration are suggested. The first plane is at the sensor level and verifies the validity of the sensor data. This detection and isolation can be accomplished using a generalised likelihood ratio test. This test uses parity equations, which are linear combinations of sensors. The second plane is designed to detect and isolate any inaccurate implementation of the control law. It engages majority voting algorithms where each channel exchanges the data with the other two channels. Finally the third plane includes a triplex electrohydraulic actuator in a mid - value redundant configuration. The introduction of this redundancy management system is necessary not only for reliability and fault tolerance purposes, but also to overcome all the control system airworthiness related problems.

3. Concept and principle of the Active Control Technique.

The proposed Active vibration isolation technique is universal and it can be described as a nodalisation technique. There is a set of isolation units placed between the gearbox and the fuselage. Each isolation unit consists of an electrohydraulic servoactuator in parallel with an elastomeric unit. Using the appropriate control law the actuator forces oppose the equivalent spring forces to create a node, or a point of zero vibration motion at the airframe attachment point. Therefore if there is no transmitted force at a node, the fuselage will not vibrate. The principle is similar to the well known Nodalisation method [2] except the fact that the node is now created by active means.

The helicopter which was investigated was the Westland W30 - 100 type. This was scaled down to a 1/16 scale model. This model is shown in figures 1.a and 1.b. It consists of two parts. The upper part is rigid and represents the engines, the gearbox and the raft. The lower part represents the fuselage. The fuselage and the raft are connected by a set of isolation units. The system is symmetrical about two vertical planes (x-z) and (y-z). These planes intersect each other along the central suspension of the structure (fig 2).

4. Control theory for the Vibration Isolation

It is well known that the more significant vibration levels are those created by rotor dynamic effects [3]. These vibrations occur at discrete frequencies, the blade passing frequency and its harmonics and generally increase in magnitude with the airspeed and the rotor blade angle of attack increase. Their frequencies, usually are ranging from about 10 to 50 hz and directly depend on the rotor speed ω and the number of blades b. The blade passing frequency is the product of the number of blades b and the rotor speed ω.

The proposed control theory achieves two results [4] a. Rejection of sinusoidal disturbances due to the blade passing frequency (21.6 hz) and its harmonics (43.2 hz, 64.8 hz,e.t.c.) b. Rejection of constant disturbances during sudden manoeuvres. These objectives are achieved using a simple stabilisation theory which is preferred to an optimisation theory. In order to fulfill the above design objectives the control force U(s) must have the form

$$U(s)=[\ G_{21}(s)\ G_{BPF}(s)\ [\ G_{HDC}(s)+G_0(s)+G_{CDC}(s)]\ +G_{11}(s)\ G_3(s)]\ \ \Delta(s)$$

where

$\Delta(s)$ is the relative displacement between the transmission and the fuselage
U(s) is the control force provided by the actuators,
$G_{11}(s)$ and $G_{21}(s)$ depend on the servovalve/actuator modelling,
$G_0(s)$ and $G_3(s)$ depend on the applied feedback and they are related to the stability of the overall system,
$G_{BPF}(s)$ is acting as a feedback dynamic compensator and its presence is dictated by the demand for arbitrary pole placement. Physically it is represented by a bandpass filter (BPF) wich is preferred to a notch filter and it is very crucial for the disturbance rejection at the frequency of interest.
$G_{HDC}(s)$ is the transfer function of the harmonic disturbance conpensator (HDC) responsible for the rejection of sinusoidal disturbances. Physically it is implemented by a proportional - derivative controller.
$G_{CDC}(s)$ represents the transfer function of the Constant Disturbance Compensator (CDC) and its purpose is the rejection of constant disturbances during manoeuvres and landing. It is implemented by an integrator,
s is the Laplace operator.
The ratios of the relative displacement versus the disturbance force and the transmitted force $F_{ti}(s)$ versus the disturbance force $F_{di}(s)$ take now the form [1]

$$\frac{\Delta_i(s)}{F_{di}(s)}=\frac{1/m}{m_R/ms^2+K_p+C_ps+G_{2i}(s)G_{BPF}(s)G_{HDC}(s)+G_{CDC}(s)+G_0(s)+G_{1i}(s)+G_3(s)} \qquad (1)$$

and

$$\frac{F_{ti}(s)}{F_{di}(s)}=\frac{1/m\{K_p+C_ps+G_{2i}(s)G_{BPF}(s)[G_{HDC}(s)+G_{CDC}(s)+G_0(s)]+G_{1i}(s)G_3(s)\}}{m_R/m\ s^2+K_p+C_ps+G_{2i}(s)G_{BPF}(s)[G_{HDC}(s)+G_{CDC}(s)+G_0(s)]+G_{1i}(s)\ G_3(s)} \qquad (2)$$

where
m is the dimensionless term $1+m_R/m_F$
m_R is the equivalent rotor mass
m_F is the equivalent fuselage mass
K_p, C_p are the stiffness and damping coefficients of the elastomeric unit.
The harmonic disturbance controller has the form

$$G_{HDC}(s)=K_{op}+C_{op}s \qquad (3)$$

Where K_{op} and C_{op} are the optimum proportional and derivative coefficients respectively. Therefore by equating the numerator of equation 2 to zero and solving versus the parameters of the HDC the criterion (a) is always guaranteed. In practice these parameters can be estimated on line with an identification algorithm whose general characteristics are shown in fig. 3. The rejection of constant disturbances is achieved by selecting a controller of the form

$$G_{CDC}(s) = K_u/s \quad (4)$$

as can be easily verified from equation 1.
K_u is a suitable constant gain.
Figure 4 illustrates the instrumentation used for the implementation of the control strategy.

5. Simulation Results

Fig 5 shows the transmitted force versus the excitation force (1000N) after the application of control. Fig 6 shows the transmitted force versus the excitation force without the application of the control force U(s). Fig 7 shows the excitation during sudden manoeuvres corresponding to 3g within 1sec. The system response is shown in fig 8. The relative displacement tends to zero after 0.6 sec. Fig 9 shows the excitation of the system in 3 different vibration levels within 1 sec. After the application of control the isolation is 100% and the transmitted force is shown in fig 10. The frequency response of the system is shown in fig 11 where the vibration isolation is in excess of 90dB for the blade passing frequency and its first harmonic. Fig 12 shows the convergence of the on - line ideutification algorithm to the optimum controler values.

6. Stability Robustness Criteria applicable to the helicopter vibration envivonment

The system design must also take into account some other factors which will guarantee that sufficient vibration isolation takes places under the existence of some unpredictable effects. These effects can include:
a. Perturbations which can lead eventually to instabilities and as a consequence to the destruction of the vibration isolation system.
b. Insensitivities to parameter variations which might lead to the nonconvergence of the identification algorithm to the optimal solution. As a result the system will not only have inadequate levels of vibration isolation but also there is a possibility of an instability of the overall system. Therefore the system must be structurally robust in order to avoid any stability related problems. This robustness can be found quantitively if either multiplicative or additive perturbations $\Delta G(s)$ are considered [5]
For a multiplicative perturbation $\Delta G(\omega)$ the closed loop feedback system will remain stable if $\Delta G(\omega)$ is stable and

$$\|\Delta G(j\omega)\| < \frac{1}{\|[I+G(j\omega)^{-1}]^{-1}\|} \quad \text{for all } \omega \quad (5)$$

Also for an additive perturbation $\Delta G(s)$ the closed loop system will remain stable if $\Delta G(s)$ is stable and

$$\|\Delta G(j\omega)\| < \frac{1}{\|[I+G(j\omega)]^{-1}\|} \quad \text{for all } \omega \quad (6)$$

Where $G(j\omega)$ is the open loop gain matrix,and $\| . \|$ corresponds to the spectral norm of a matrix, I is the unity matrix. The multiplicative and additive feedback configurations are shown in figures 13.a and 13.b. There is no need to apply the above theorem for broad band frequencies of the helicopter vibration problem, but only for the blade passing frequency and its harmonics. Hence for the multiplicative case

$$\|\Delta G(jn\omega_b)\| < \frac{1}{\|[I+G(jn\omega_b)^{-1}]^{-1}\|} \quad \text{where } n=1,2,\ldots \quad (7)$$

and for the additive case

$$\|\Delta G(jn\omega_b)\| < \frac{1}{\|[I+G(jn\omega_b)]^{-1}\|} \quad \text{where } n=1,2,\ldots \quad (8)$$

and ω_b is the blade passing frequency.

When the spectral norm of the perturbation frequency response is always less than some value, stability is maintained and the convergence of the on-line identification algorithm is secured. The identification algorithm ensures that there is no need for the exact system model and the accurate knowledge of the system parameters is not necessary.

The application of the small gain and phase theorems can also give an answer to the stability robustnes demand for the frequency range $[0,2\omega_b]$ and $[2\omega_b,\infty]$. From fig 11 and taking into account [6]

$$\sup_{\omega_b} \delta_m(\omega_b) < \frac{1}{\mathrm{Sup}_{am}(\omega_b)} = \frac{1}{21837} = 4.57*10^{-5}$$

for the blade passing frequency and

$$\sup_{2\omega_b} \delta_m(2\omega_b) < \frac{1}{\sup_{2\omega_b} a_m(2\omega_b)} = \frac{1}{6307} = 1.58*10^{-4}$$

for the first harmonic of the b.p.f.

Where $\delta 1 \leq \delta 2 \leq \ldots \delta_m(\omega)$ are the principal gains of $\Delta G(j\omega)$ and $\alpha_1 < \alpha_2 < \ldots\ldots \alpha_m(\omega)$ are the principal gains of $[I+G(j\omega)^{-1}]^{-1}$.

The structured singular value $\mu(G)$ can also be used for the helicopter vibration isolation problem. The reason is that there is a desired performance objective and a set of possible external input signal like disturbances and setpoints. It is defined at each frequency (blade passing frequency and its first harmonic) such that $\mu^{-1}(G)$ is equal to the smallest $\overline{\sigma}(\Delta G)$ needed to make $I+\Delta G$ singular where $\overline{\sigma}(\Delta G)$ denotes the maximum singular value of the perturbation. $\mu(G)$ depends on the matrix G and the structure of the perturbations ΔG[7].

7. Redundancy Management for the Vibation Isolation System.

If the robustness criteria set in the previous section are not met following component failures or unexpected changes in operating conditions, the performance of the active vibration isolation system will be degraded. Also the trend in many dynamic systems is towards more complexity to meet higher levels of performance. Such systems require increasing levels of reliability through redundancy. Therefore in order to maintain the highest level of performance it is important that any faults be promply detected and isolated. The next step will be then the remedy action(reconfiguration) and the normal operation of the vibration isolation system. Active redundancy is suggested through three parallel channels corresponding to each isolation unit. Triple redundancy of all channel components in order to make the system two - fail operative. The redundant information includes outputs from sensors, filters, microprocessors, controllers, which are physically segregated from each other in order to remove any possibility of inter - lane fault propagation. There are three sections which are very critical for the normal operation of the active vibration system. These sections require correct sensor data, accurate implementation of the control law and optimum servoactuator outputs respectively.

7.a First Plane of fault - detection and Isolation

The failure detection - isolation confiquration for the first plane is shown in fig 14. There are three similar sensors which are combined in three pairs. Each pair measures the difference of the two sensor outputs and provides the residual state necessary for the implementation of the Generalized Likelihood Ratio Test (GLRT). The residuals used are the innovations generated by the Kalman - Bucy filter . These innovations are the differences between the measured sensor residual outputs y(t) and the estimates obtained by the Kalman filtering. That is

$$e(t) = e_i(t) - e_j(t) =$$

$$= [\tilde{y}_i(t) - \tilde{y}_j(t)] - c\{\hat{x}_i(t/t-1) - \hat{x}_j(t/t-1)\} =$$

$$\tilde{y}(t) - c\,\hat{x}(t/t-1) \qquad (9)$$

Where $\hat{x}_i(t/t-1)$, $\hat{x}_j(t/t-1)$ are the Kalman estimates of the sensors i and j respectively,

$\hat{x}(t/t-1) = \hat{x}_i(t/t-1) - \hat{x}_j(t/t-1)$ is the residual state

$\tilde{y}_i(t)$, $\tilde{y}_j(t)$ are the sensor outputs i and j respectively.

The sensor faults are considered stochastic because they are indicated by jumps occuring at random intervals with random amplitudes. The test uses two hypotheses : H_0 is the null - hypothesis or no jump hypothesis and H_1 is the hypothesis under which a jump has occured indicating a faulty sensor . The residual can be also expressed in the form :

$$e(t) = G(t,\theta)\, v + \gamma_1(t) \qquad (10)$$

where G depends on the measurement matrix and the state transition matrix

γ_1 is a zero mean white noise

v is the unknown size of the random jump

θ is an unknown positive integer which assumes a value if a jump occurs.

The method used in [9] can be directly applied for the innovations of equation (9). The likelihood ratio test will be of the form :

$$l[t,\hat{\theta}(t)] \underset{H_0}{\overset{H_1}{\gtrless}} \epsilon \qquad (11)$$

Where $\hat{\theta}(t)$ is the maximum likelihood estimate of θ . If the three GLRT fulfill the null hypothesis or $l_{ij} < \epsilon$, $l_{jk} < \epsilon$, and $l_{ki} < \epsilon$, where i,j and k correspond to the i,j,k sensors respectively and ϵ is the threshold level, then all the sensors provide the correct data. Hence there is not any discruption to their normal operation if a jump is detected, then a sensor is faulty i.e there is a zero output, or a bias. This means that two of the likelihood ratios will be greater than ϵ i.e $l_{jk} > \epsilon$ and $l_{ki} > \epsilon$. It is obvious that the faulty sensor is the sensor k, which is isolated immediately. The output of this sensor is replaced by the average value of the rest two error free sensors.

7.b. Second plane of detection and Isolation.

The vibration isolation system includes three controllers interfacing with the three microprocessors via a majority voting node (fig 14). The importance of the plane is highlighted by the fact that the optimum values of these controllers are very crucial for the correct implementation of the control law. The microprocessors send the computed optimum values to the controllers and exchange them with the rest processors in a cyclic way. The 12 bit digital values created by the residuals are running in a majority voting system. Three parity equation residuals exist which are digitally cross-linked so that each processor has one redundant source of information. The pattern of the parity equations is very simple and effective. It compares the difference of any two like processor outputs to an optimal threshold level. This optimal threshold should be related to the noise properties, modelling errors and the parameter uncertainties, of the system. The parity equations have the form :

$$P_{1n} = Y_{1n} - Y_{2n} \qquad (12)$$

$$P_{2n} = Y_{2n} - Y_{3n} \quad (13)$$

$$P_{3n} = Y_{3n} - Y_{1n} \quad (14)$$

Where P_{1n}, P_{2n}, P_{3n} are the residuals and Y_{1n}, Y_{2n}, Y_{3n} correspond to the optimum controller values. When there is an inaccurate implementation of the control law the parity equations involving it will be violated, whereas those excluding it will still hold. The failure detection is achieved if the parity equation residual P_{mn} is greater than the threshold F_{th} and is reset to zero otherwise.

$$F_{mn} = 1 \text{ if } P_{mn} > F_{th} \quad (15)$$
$$= 0 \text{ otherwise}$$

The faulty controller is identified if the Boolean variable Lmn is one

$$L_{1n} = F_{1n}\,\overline{F}_{2n} \quad (16)$$

$$L_{2n} = F_{2n}\,\overline{F}_{3n} \quad (17)$$

$$L_{3n} = F_{3n}\,\overline{F}_{1n} \quad (18)$$

The selected signal at the node is mechanized to output three identical signals in all channels to drive the harmonic - disturbance compensators. The controller outputs pass through three identical BPF which are linked to the servoamplifiers.

7.c. Third plane of fault detection and Isolation.

This plane includes hardware redundancy exclusively. There is a triplex electrohydraulic servoactuator which is double fail operative. It will continue to work satisfactorily after any two failures which might be loss of function of any element within the servoactuator, loss of command signal to the servoactuator, or an erroneous hardover electrical command. The electrohydraulic servovalve has three coils each of which is driven by one of the electrical signals provided by the bandpass filters (fig 14). If one coil current goes hardover, the high gain of the electrical feedback from the remaining good channels will limit the actuator output transient until the channel is shut down. This is a mid - value system preferred to a majority voting system because it will not be affected by any single failure like a hardover or a channel drift. The hydraulic outputs from the three servovalves are flow summed at the triplex actuator.

8. Conclusions.

An active vibration isolation system for a helicopter has been developed. This system provided 100% vibration isolation under normal flight and zero deflection during manoeuvres. Stability robustness criteria were set for the system in order to guarantee that the design objectives are fulfilled always. The introduchion of the redundancy management configuration is evident not only for better reliability and fault tolerance purposes, but also to minimise any problems related with the more stringent airworthiness issues.

Further improvements to the system performance can be achieved by adopting the expert system approach [8] and by applying higher level languages (Ada) [10].

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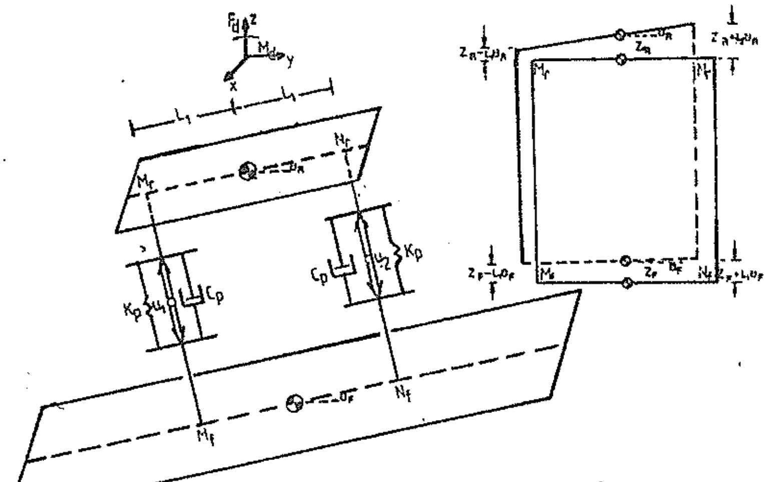

Figure 1.a Two mass-model of the W30-100 Helicopter

Figure 1.b Position of the centre of gravities of the gearbox and fuselage before and after the excitation

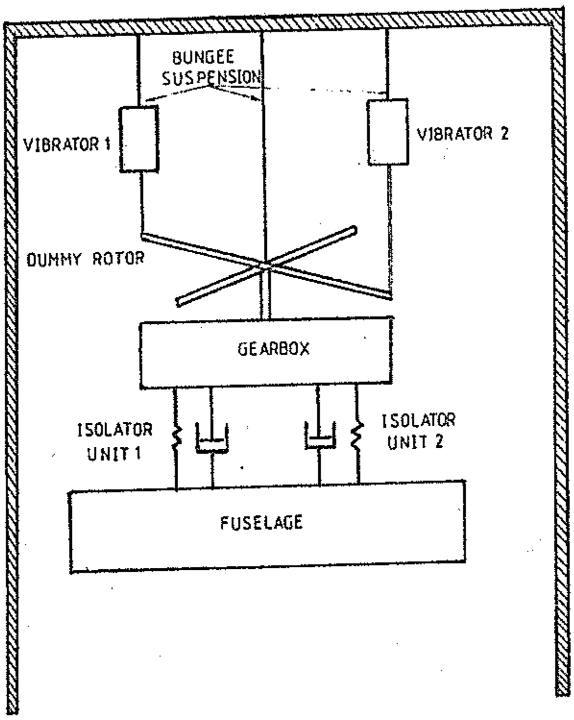

Figure 2 Cross section of the experimental rig

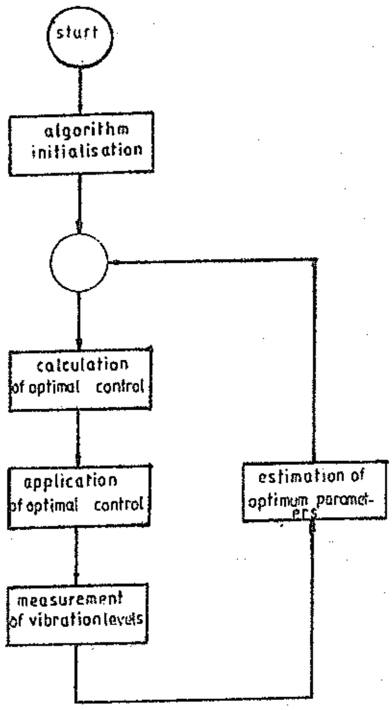

Figure 3 Structure of the on-line identifica

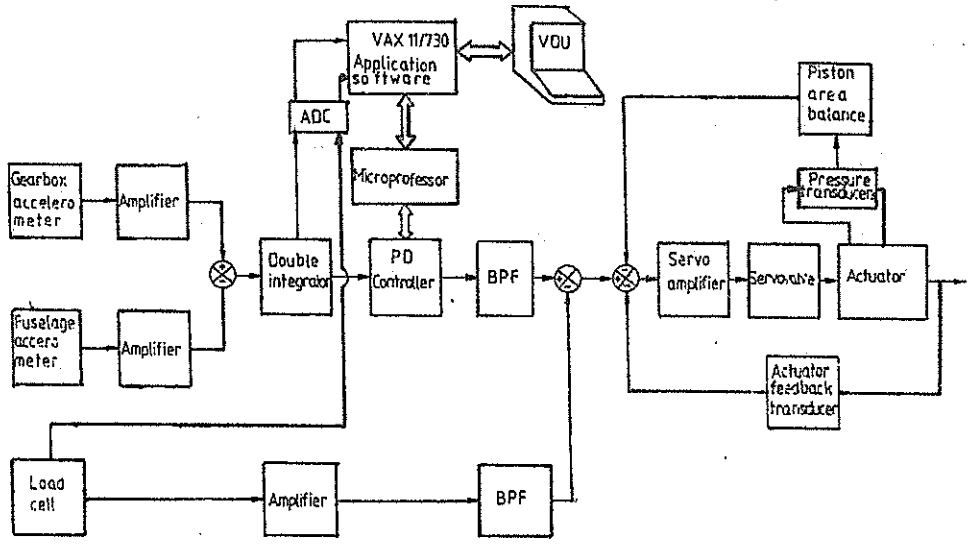

Figure 4 Block diagram of the instrumentation used for the control theory

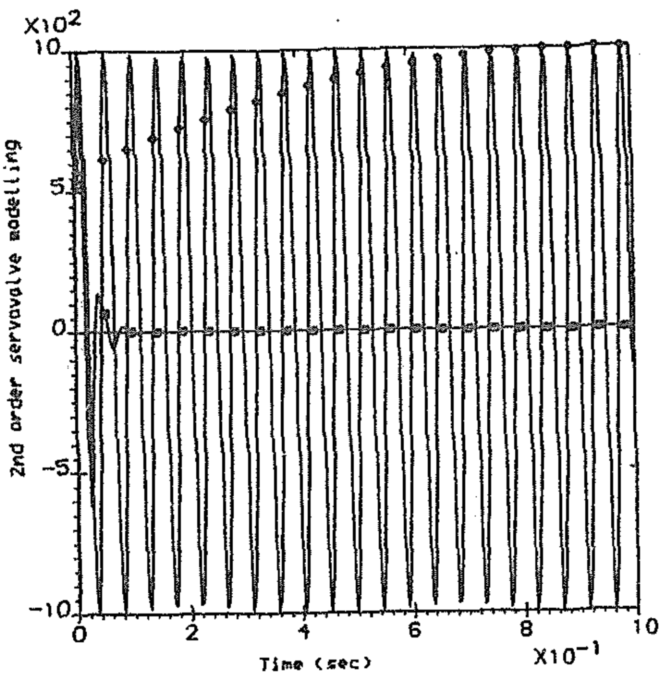

FIGURE 5 Transm. Force versus exc. Force for the one d.o.F.
□ Transm.Force(N) to Fuselage
◇ Excit. Force:1000sin(135.72t)

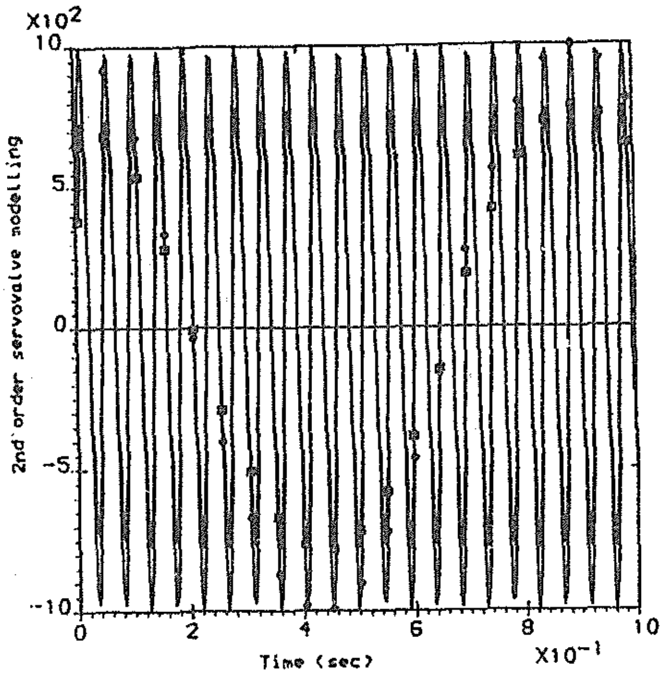

FIGURE 6 Transm. Force without control versus the excitation Force
□ Transm. Force (N)
◇ Excit. Force:1000sin(135.72t)

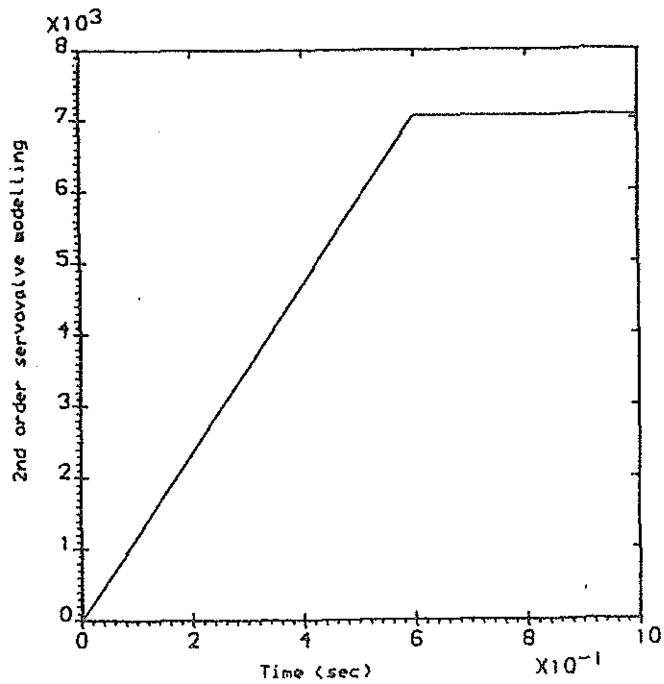

FIGURE 7 Excitation Force during sudden manoeuvres (N).

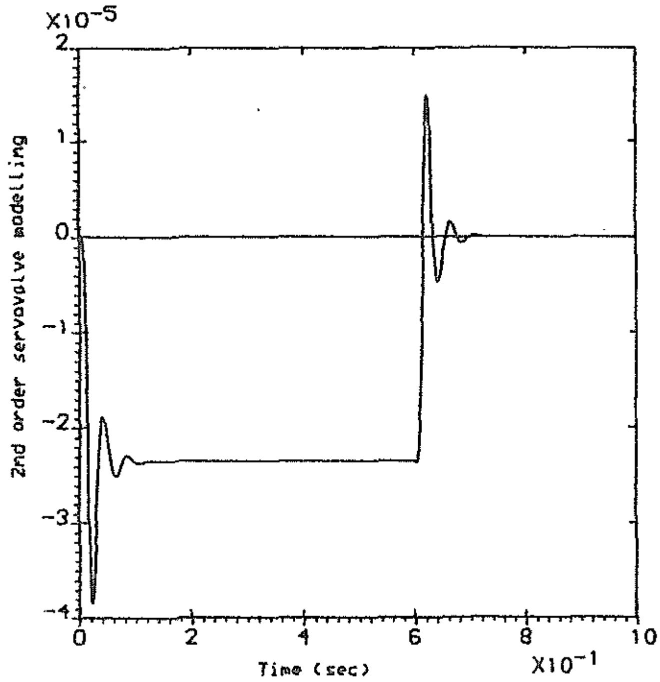

FIGURE 8 Relative displacement during manoeuvres (m).

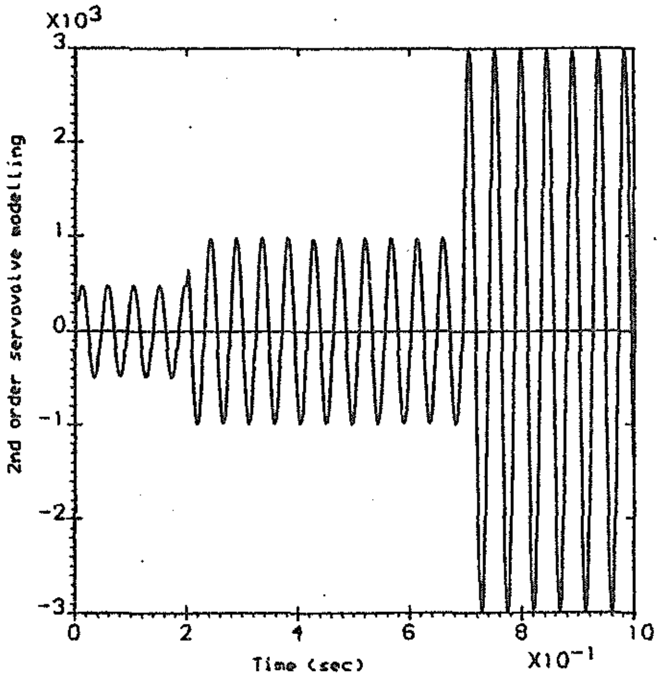

FIGURE 9 Disturbance force with 3 different levels in 1 sec.

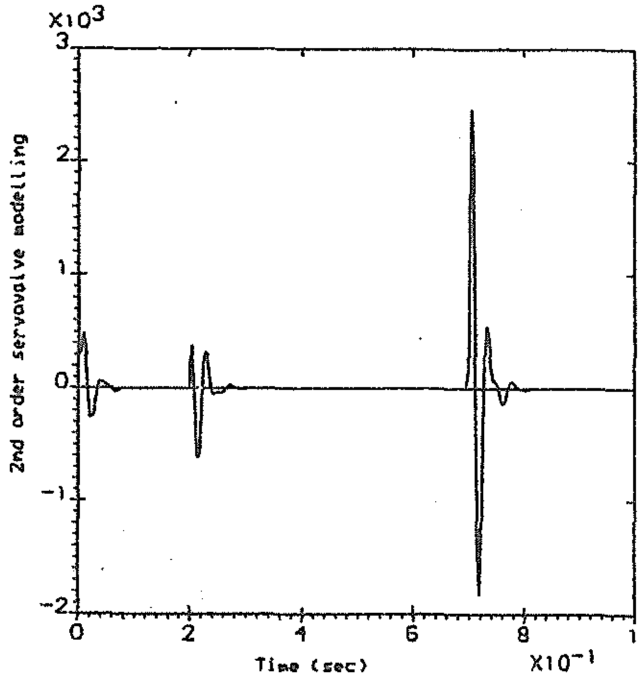

FIGURE 10 Transmitted Force(N).

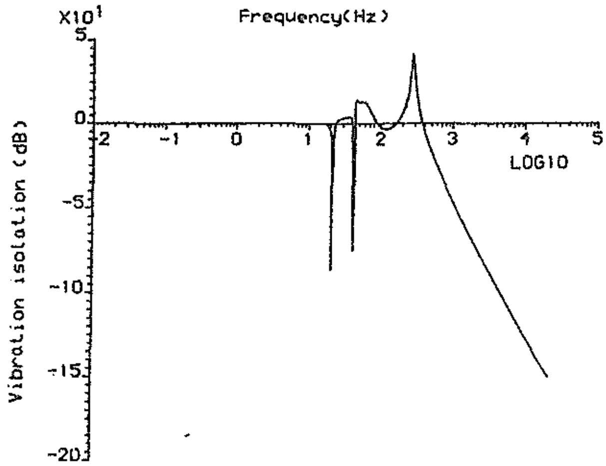

Figure 11 Vibration Isolation versus frequency

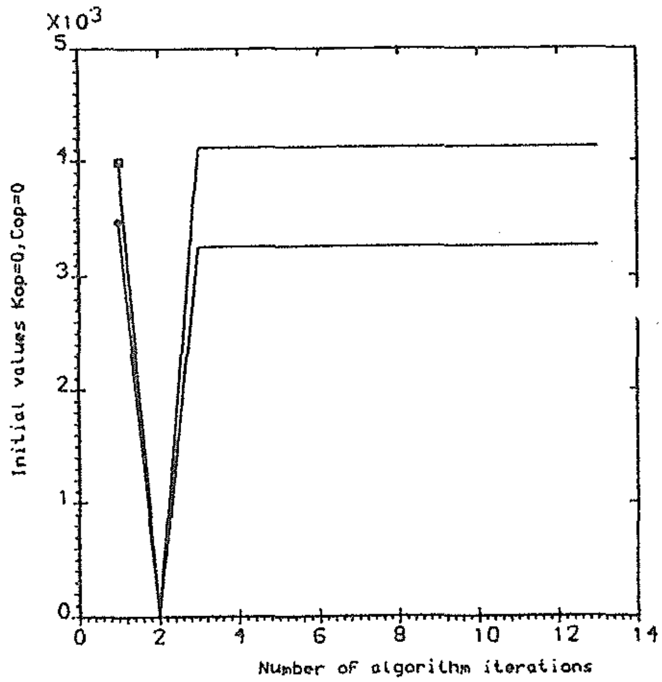

FIGURE 12 Identification algorithm convergence.
□ Prop. optimum gain, Kop
◇ Der. optimum gain, Copx100

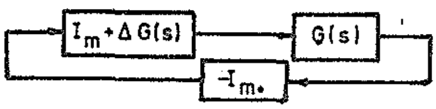

Figure 13a Feedback configuration with multiplicative perturbation

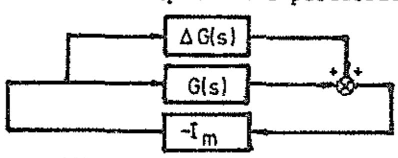

Figure 13b Feedback configuration with additive perturbation

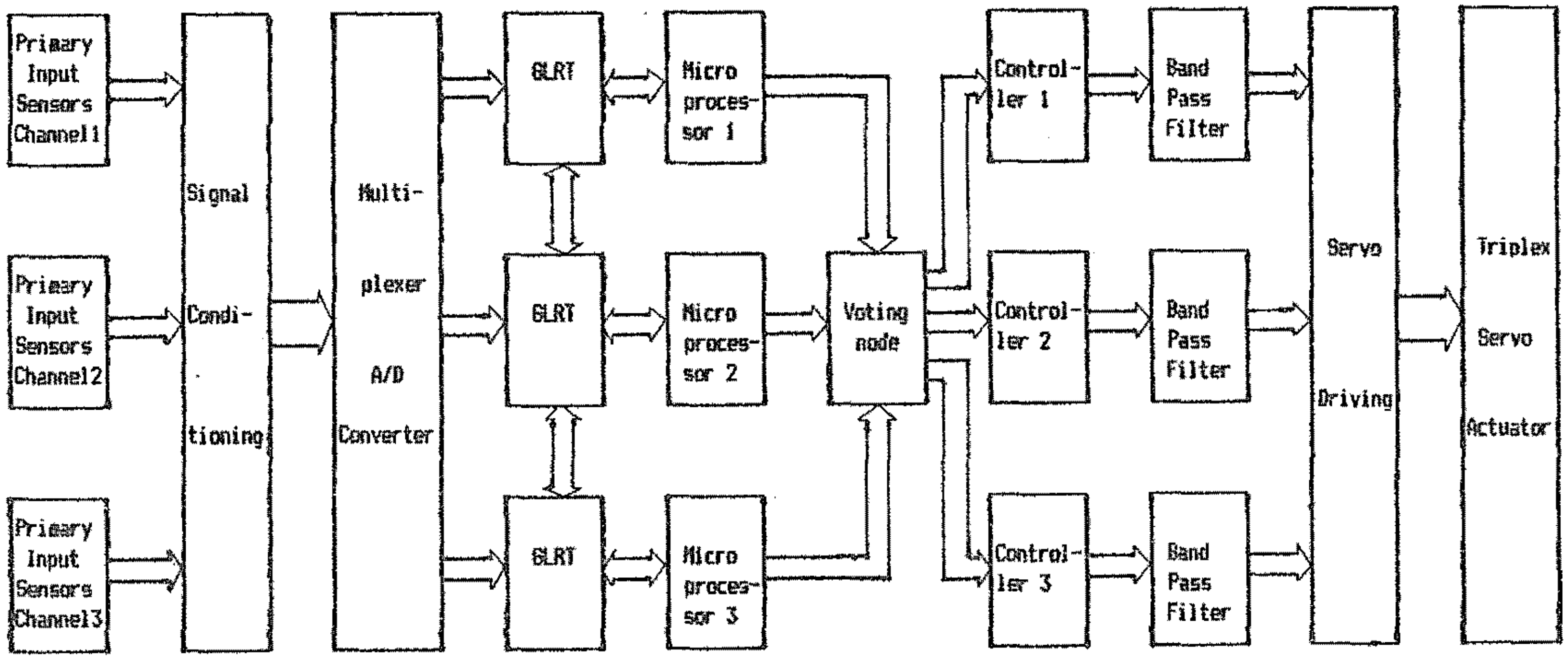

Figure 14 Redundancy Management of the vibration isolation system