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Flight Control Frequency Response Testing of the Shuttle Ascent Vehicle

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This paper presents results of flight control frequency response testing of the space shuttle vehicle conducted at NASA's Marshall Space Flight Center (MSFC) in Alabama. The purpose of this testing was to verify the response of flight control accelerometers and rate gyroscopes to vibratory structural motion induced by rocket thrust control forces. This transmissibility is exceptionally important to the shuttle vehicle because it employs accelerometers for control purposes to alleviate aerodynamic loads during a portion of flight, and accelerometers are particularly responsive to vibratory motion. Test results are compared with theoretical calculations and a limit criterion. These comparisons demonstrate the adequacy of the flight control design, or provide a basis for modifying control system characteristics to ensure that response to flexural motion is adequately suppressed.

Nomenclature

a = control system numerical scaling factor, deg/s/deg, deg/ft/s^2
F = control signal processing function, nondimensional
G = system transfer function, nondimensional
H = system return signal transfer function, deg/s/lb, ft/s^2/lb
δ = rocket nozzle deflection, deg

Subscripts

a = amplifier
A = main rocket actuators
C = compensation
D = digital dynamics
E = main rocket engine
N = booster rocket nozzle
R = booster rocket actuators

Introduction

THE response and sensitivity of flight control instruments to vehicle vibrations during flight must be known to ensure that they will not affect control system operations. In the shuttle vehicle, this knowledge is particularly important because, in addition to rate gyroscopes and inertial platforms, the flight attitude control system utilizes accelerometers as primary sensing instruments. The accelerometers, which are particularly sensitive to flexible motion, are utilized during the portion of flight where aerodynamic loads in excess of structural limitations could occur. Since the shuttle ascent flight control system design was predicated on theoretical flexible structure characteristics, the magnitude of vibratory motion transmitted through the flight control instruments needed to be verified by testing. The use of flight control instruments to verify response characteristics was conceived and proposed in October 1974, but the tests were conducted in late 1978 and early 1979, a 4 to 5 year delay between concept and execution. The experimental verification of the transmission of vibration through the flight control instruments is believed to be a novel procedure for such a system. References 1-4 discuss similar test concepts and related theory, which indicates others were thinking along similar lines in the same period. The testing was conducted on the ascent vehicle under conditions representing various flight configurations.

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The ascent phase of flight occurs in three parts, or stages, that involve different configurations of shuttle vehicle elements. During the first stage, the orbiter, external fuel tank, and two solid-propellant booster rockets ascend until the booster rockets have expended their propellant. These are then separated from the external tank, and the second stage of flight begins. The orbiter and external tank proceed until the liquid fuel is consumed and the empty tank is separated from the orbiter. In the third stage, the orbiter continues the ascent on its own into Earth orbit.

This paper deals with vibration testing of the first- and second-stage vehicle configurations at NASA's Marshal Space Flight Center (MSFC), located on the Redstone Arsenal in Alabama.

Test Procedure and Evaluation Criteria

The vibration tests were simple in concept. Electromechanical vibrator devices (Fig. 1) were used to shake the structure at discrete frequencies from 6 to 157 rad/s. The force applied to the structure by these devices was recorded as the test proceeded. At the same time, output signals from the flight control rate gyroscopes and accelerometers were recorded.

Tests were conducted as shown in Table 1 for each of five test configurations that represented the vehicle under various flight conditions: the beginning and end of stage 1 and the beginning, middle, and end of stage 2. The different test weights were obtained by loading water in the liquid oxygen tank. Also, the booster rockets were filled with an inert

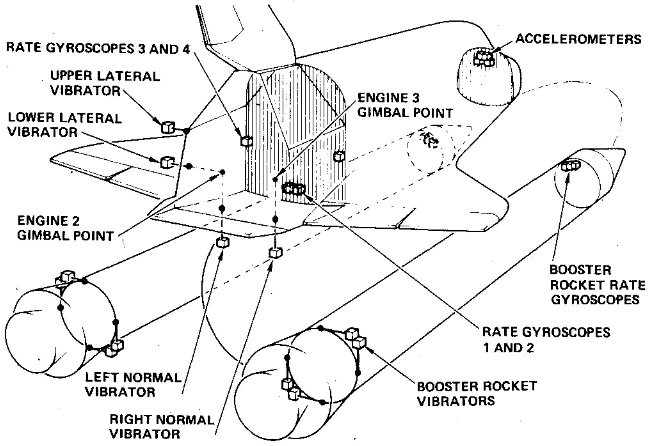


Fig. 1 Location of vibrators, rate gyroscopes, and accelerometers.

Table 1 Summary of flight control transfer function vibration testing

Test configuration	Flight condition represented by test	Approximate vehicle weight, lb	Date of testing
Stage 1	Begin stage 1	3,890,000	Nov. 1978
	End stage 1	1,670,000	Feb. 1979
Stage 2	Begin stage 2	1,120,000	June 1978
	Midstage 2	650,000	June 1978
	End stage 2	350,000	June 1978

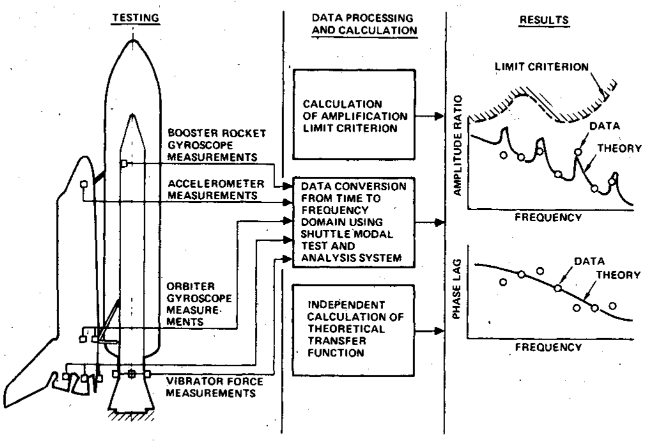


Fig. 2 Test arrangement and data processing.

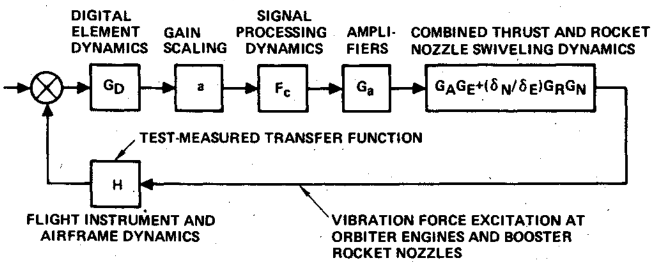


Fig. 3 Flight control rate gyroscope and accelerometer feedback dynamics for evaluating frequency response tests.

material for the "liftoff" test condition, but were empty for the end of stage 1 test condition.

Vibration forces were applied during each test condition to excite rate gyroscopes sensitive to pitch, yaw, and roll axis rates of rotation as well as accelerometers sensitive to normal and lateral direction accelerations.

Upon completion of the tests, these data were processed to produce the ratio of response between the flight control instrument electrical output signal and vibration input forces over the given frequency range (Fig. 2). These data were then compared with predicted results, as well as a flight control amplification limit criterion.

This criterion was developed in order to assess and interpret the flight control frequency response vibration test results. An amplification limit criterion was conceived as a figure of merit that would indicate the acceptability of test data for flight control purposes. The criterion is based on suppressing or attenuating signal content below a specified level. Criteria were developed for rate gyroscopes and accelerometers used in pitch, yaw, and roll control functions corresponding to the attitude control and load alleviation control modes that are operative during stage 1 and 2 flight. A block diagram illustrating the amplification of motion in the flight control system caused by structural flexibility is shown in Fig. 3. Transfer function H is the characteristic measured by the vibration tests. This transfer function is measured in units of deg/s/lb for rate gyroscopes and unit of g/lb for accelerometers.

If the loop gain or amplification factor of the control system, including the test measured transfer function, from input to return signal is less than unity, it cannot be unstable. Therefore, the product of transfer function elements in this system

$$aF_C G_D G_A [G_A G_E + (\delta_N/\delta_E) G_R G_N] H = 1$$

is significant for use as a criterion to discriminate amplification factors.

The term (δ_N/δ_E) represents the ratio of angular deflection in degrees of booster rocket nozzles (δ_N) to deflection of orbiter rocket engines (δ_E). This term appears because both sets of rockets are used for control purposes during stage 1.

Rearranging the product of transfer function elements so that

$$H \leq 1/\{aF_C G_C G_a [G_A G_E + (\delta_N/\delta_E) G_R G_N]\}$$

delimits values of the test transfer function H, which ensures that the amplification factor is less than unity at any frequency.

Table 2 Normalized[a] average values of selected prominent response ratios for rate gyroscopes

Test condition	Pitch rate gyroscope response (Mag at freq)[b]	Yaw rate gyroscope response (Mag at freq)[b]	Roll rate gyroscope response (Mag at freq)[b]
Begin stage 1	0.008 @ 13	0.005 @ 13	0.26 @ 14
	0.009 @ 19	0.009 @ 26	0.064 @ 16
	0.028 @ 33	0.035 @ 33	0.032 @ 29
End stage 1	0.011 @ 16	0.015 @ 26	0.22 @ 16
	0.007- @ 21	0.011 @ 28	0.062 @ 25
	0.009- @ 43	0.020 @ 33	0.013 @ 55
Begin stage 1	0.18 @ 21	0.12 @ 28	0.04 @ 6
	0.04- @ 39	0.04 @ 37	0.02 @ 28
	0.03 @ 57	0.04- @ 50	0.02- @ 50
End stage 2	0.10 @ 26	0.01 @ 16	c
	0.03 @ 48	0.01 @ 32	0.04 @ 30
	0.03 @ 59	0.01- @ 41	0.03 @ 50

[a] Compound forces were applied by multiple vibration devices during these tests. Results are normalized to a unit force basis.
[b] Magnitude of response is in mdeg/s/lb, and frequency of response is in rad/s.
[c] Data at less than 16 rad/s are not available from this test.

Even more stringent criteria can be applied to ensure attenuation below some aribitrary level. For example, the shuttle design criteria require attenuation of the amplitude ratio equal to or less than 0.32 (−10 dB) for frequencies greater than 25 rad/s. While discussion of aeroelastic effects is beyond the scope of this paper, it is noted that this attenuation criterion allows for substantial degradation due to these effects. The criterion then becomes

$$H \leq 0.32/\{aF_C G_D G_a [G_A G_E + (\delta_N/\delta_E) G_R G_N]\}$$

for stage 1, or

$$H \leq 0.32/(aF_C G_D G_a G_A G_E)$$

for stage 2 when only the shuttle rocket motors are available for control. These functions (with and without the compensation function included) were computed and plotted as the amplification limit criterion line that appears on the frequency response data. In many cases, the criteria with the compensation function included do not appear on the figures because they are beyond the scale of the data.

Limit functions were computed by using representative transfer function frequency response characteristics for actuators (G_A and G_R), thrust and nozzle inertia forces (G_E and G_N), compensation functions (F_C), digital system sample and hold dynamics (G_D), gain scaling factors (a), and electronic signal amplifiers (Ga) in the control system.

There are different limit function criteria corresponding to the stage of flight and the mode of control in operation at a particular time. These include an ordinary attitude control mode in pitch and yaw from 0 to 26 s and from 93 to 124 s during stage 1 flight and during all of stage 2 flight. Moreover, a load alleviation control mode (in which accelerometers are utilized in addition to rate gyroscopes) is operative for pitch and yaw control during first-stage flight from 43 to 65 s.

With the test procedure and evaluation criteria established, the test results can now be presented.

Table 3 Normalized[a] average of selected prominent responses of accelerometers from stage 1 tests

Test condition	Normal accelerometer response (Mag at freq)[b]	Lateral accelerometer response (Mag at freq)[b]
Begin stage 1	0.05 @ 13	0.09 @ 14
	0.08 @ 16	0.14 @ 16
	0.17 @ 19	0.18 @ 26
	0.17 @ 28	0.08 @ 33
	0.12 @ 33	0.05 @ 35
End stage 1	0.32 @ 14	0.45 @ 22
	0.10 @ 19	0.55 @ 28
	0.35 @ 21	0.42 @ 33
	0.32 @ 41	0.19 @ 39
	0.32 @ 50	0.19 @ 50

[a] Compound forces were applied by multiple vibrator devices during these tests. Results are normalized to a unit force basis.
[b] Magnitude of response is in mft/s/s/lb, and frequency of response is in rad/s.

Results

Selected representative results of these tests are tabulated in Tables 2 and 3. Table 2 shows the response characteristics of the rate gyroscopes, and Table 3 shows response characteristics of the accelerometers. Stage 1 pitch and yaw gyroscope data are from left and right booster rocket rate gyroscope installations; stage 2 pitch and yaw gyroscope data are from orbiter rate gyroscope installations. All of the roll rate responses are from the orbiter roll rate gyroscopes. (Roll rate gyroscopes are not mounted on the booster rockets.)

These tabulated data show the most significant or dominant frequencies of response and the corresponding amplitudes of the response ratio at that frequency for the various rate gyroscopes and accelerometers.

These data indicate the progression of modal frequencies to larger values as the vehicle becomes lighter in weight from the beginning to the end of stage test condition. For example, the first prominent mode at the beginning of the stage 1 test condition is 13 rad/s, whereas this mode has progressed to 16 rad/s at the end of the stage 1 test condition. Since these modes are the most prominent, a control system design that adequately suppresses these modes will also be adequate for other less significant modes of response.

In addition to the tabulated results, the following discussion presents selected frequency response characteristics from stage 1 and stage 2 test data.

Selected Results From Stage 1 Tests

Information from 49 test and flight control rate gyroscopes and accelerometers was recorded during vibration testing for each of the two stage 1 test conditions. These test conditions represented the launch vehicle at the beginning of stage 1 after liftoff from the launch complex and near the end of stage 1 before separation of the booster rockets. For each test condition, a series of ten tests was conducted by using a combination of two, four, or six vibrators. These ten tests, and the instruments of primary response, are given in Table 4.

Individual tests were designed to show response to booster rocket vibrators (tests 2, 4, and 6), orbiter-placed vibrators (tests 1, 3, and 5), and combined orbiter and booster rocket vibrators (tests 8, 9, and 10). Each of these tests, in turn, was planned to provide primary excitation of pitch, yaw, or roll rate gyroscopes, and lateral or normal direction sensitive accelerometers as indicated in Table 4.

Table 4 Designation of ten stage 1 vibration tests, and primary and secondary instruments of response[a]

Test no.	Vibrator operation	Pitch rate gyroscopes	Yaw rate gyroscopes	Roll rate gyroscopes	Normal accelerometers	Lateral accelerometers
1	4 lateral booster vibrators in unison		×			×
2	2 lateral orbiter vibrators in unison		×			×
3	4 normal booster vibrators in unison	×			×	
4	2 normal orbiter vibrators in unison	×			×	
5	2 left normal opposing 2 right normal booster vibrators			×		
6	Left normal opposing right normal orbiter vibrator			×		
7	1 longitudinal vibrator					
8	6 normal vibrators in unison	× (Figs. 3, 4)[b]			× (Fig. 10)[b]	
9	6 lateral vibrators in unison		× (Figs. 5, 6)[b]			× (Fig. 9)[b]
10	3 left normal vibrators opposing 3 right normal vibrators			× (Figs. 7, 8)[b]		

[a] × indicates primary instruments of response during tests; all others are secondary response.
[b] Designates figure that shows representative results from this test.

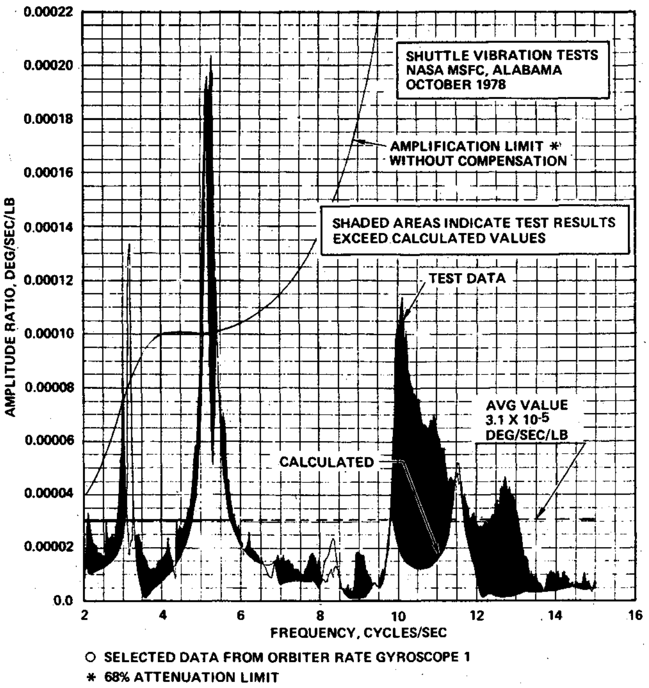


Fig. 4 Frequency response of left booster rocket pitch rate gyroscope 1 (begin stage 1 test condition).

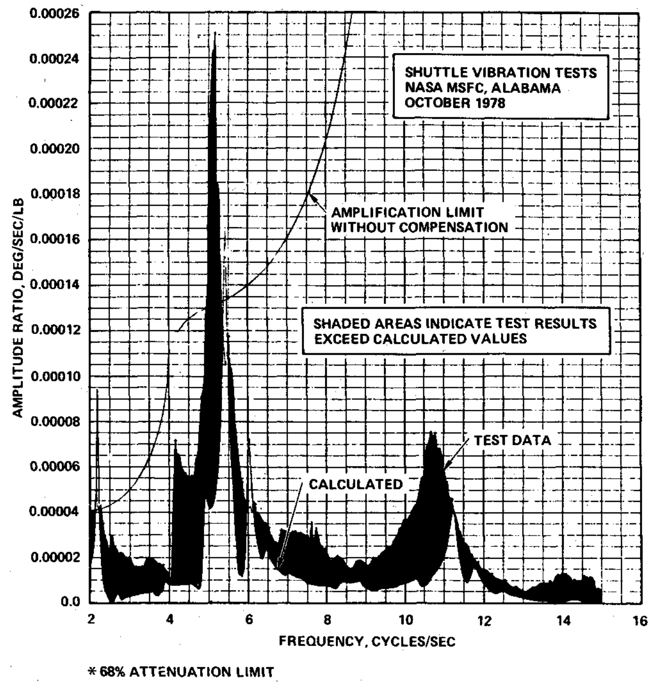


Fig. 6 Frequency response of left booster rocket yaw rate gyroscope 1 (begin stage 1 test condition).

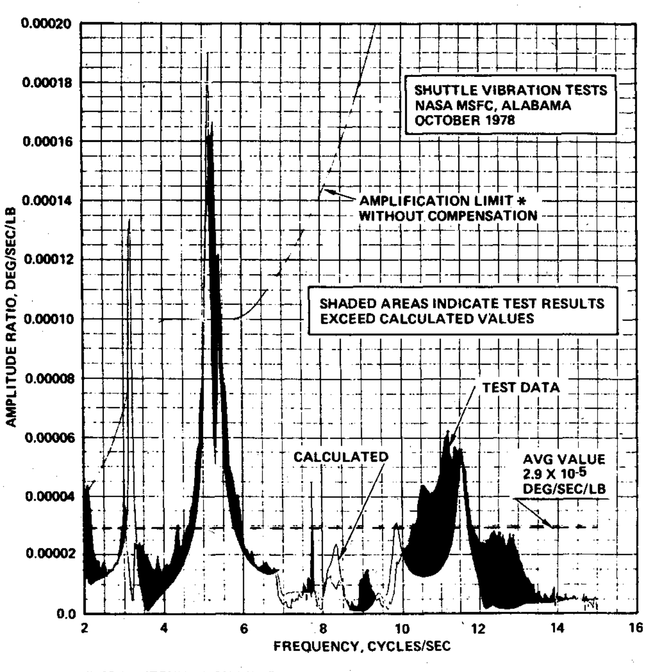


Fig. 5 Frequency response of right booster rocket pitch rate gyroscope 1 (begin stage 1 test condition).

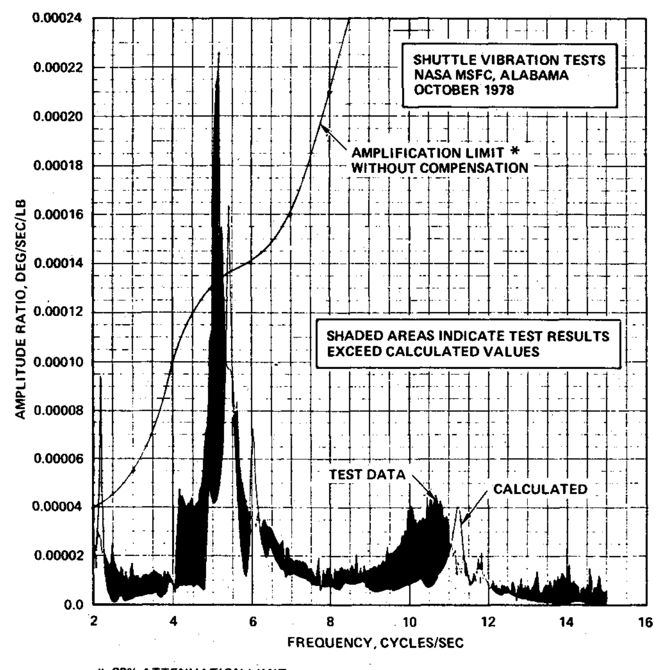


Fig. 7 Frequency response of right booster rocket yaw rate gyroscope 1 (begin stage 1 test condition).

For each test condition, 490 transfer functions are available. Only selected results are discussed here. These include data from tests 8, 9, and 10 to illustrate pitch, yaw, and roll rate gyroscope response, and normal and lateral direction accelerometer response. Results from other tests are not shown.

From these tests, only the responses of instruments actually used for flight control during stage 1 are presented. These include rate gyroscopes mounted on the left and right booster rocket for pitch and yaw axes control, and orbiter-mounted rate gyroscopes used for roll axis control. In the case of triple- or quadruple-redundant instruments, data from only one instrument are presented.

Figures 4 and 5 show the response of pitch rate gyroscopes mounted on the booster rockets. These results are from the stage 1 liftoff test condition. Figure 4 presents results from the No. 1 gyroscope on the left booster rocket, while Fig. 5 shows similar results from the No. 1 location on the right booster rocket. These data show good similarity between calculated and observed charcteristics as well as the similarity between the left- and right-hand elements. Calculated values were obtained by using a constant viscous damping factor of 0.01.

The average unnormalized level of response for these gyroscopes is 0.03 mdeg/s/lbf. These data are termed "unnormalized" because compound vibration forces were applied by multiple vibration devices during the tests, but

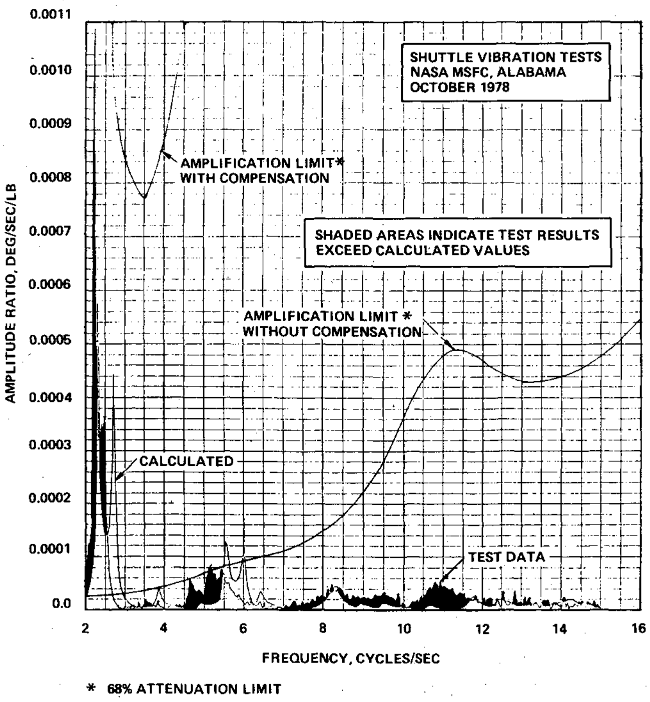


Fig. 8 Frequency response of flight control roll rate gyroscope 1 (begin stage 1 test condition).

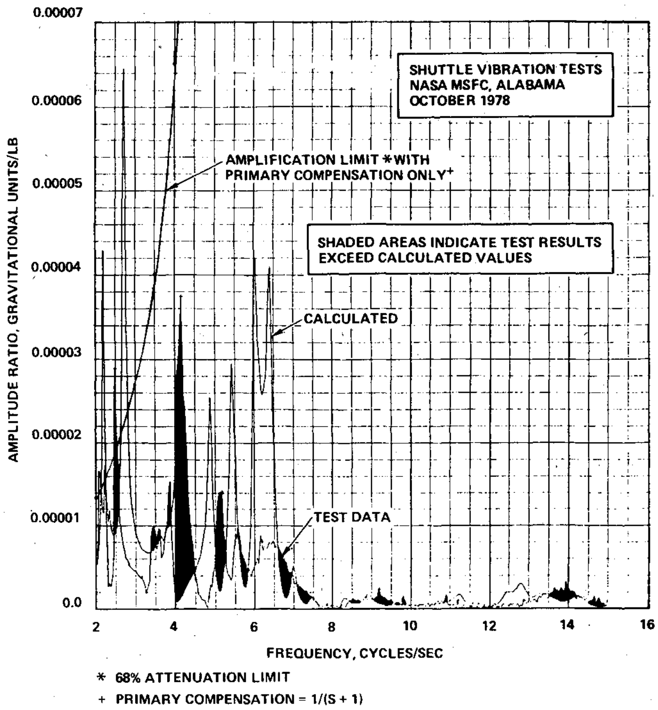


Fig. 10 Frequency response of lateral flight control accelerometer 1 (begin stage 1 test condition).

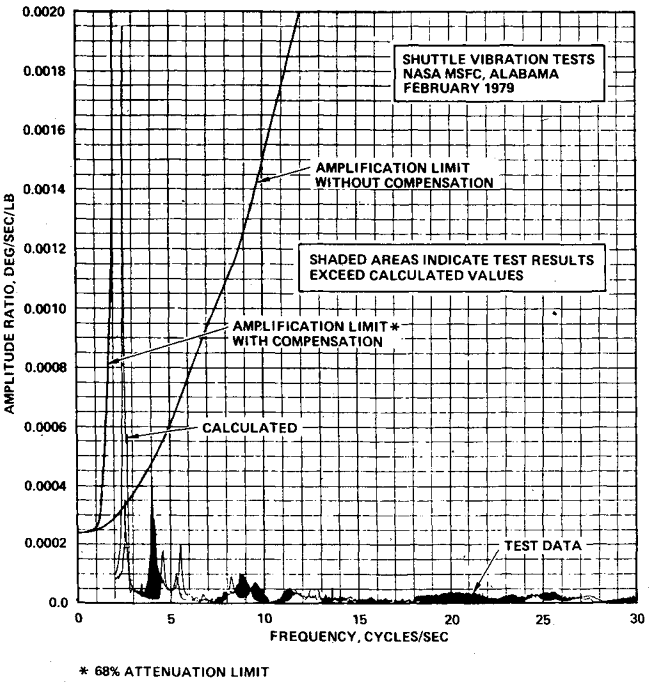


Fig. 9 Frequency response of flight control roll rate gyroscope 1 (end stage 1 test condition).

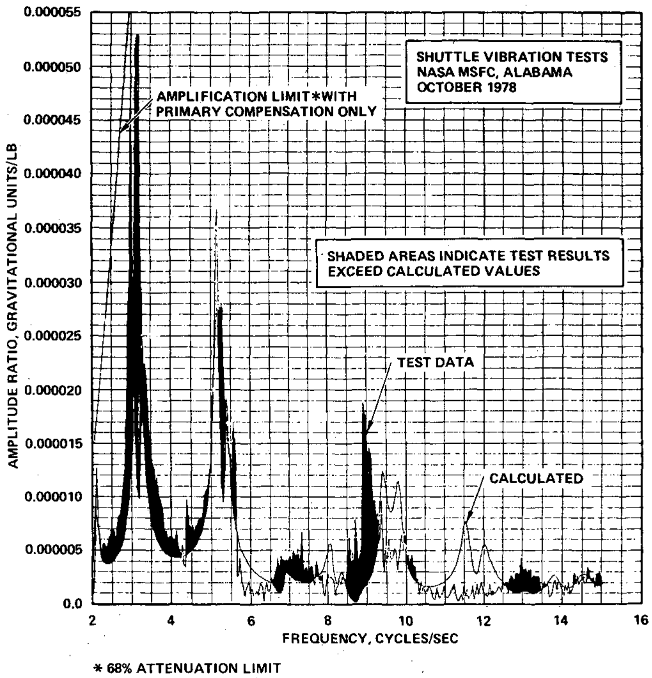


Fig. 11 Frequency response of normal flight control accelerometer 1 (end stage 1 test condition).

response ratios were calculated by dividing the instrument output by the force of a single vibrator. A normalizing factor of 6.5 can be used to reduce results to a unit force basis.

The normalizing factor is not an integer value because of circumstances of the tests. During first stage tests, fractionally dissimilar forces were applied on the orbiter and booster rockets in the approximate proportion of rocket forces in flight.

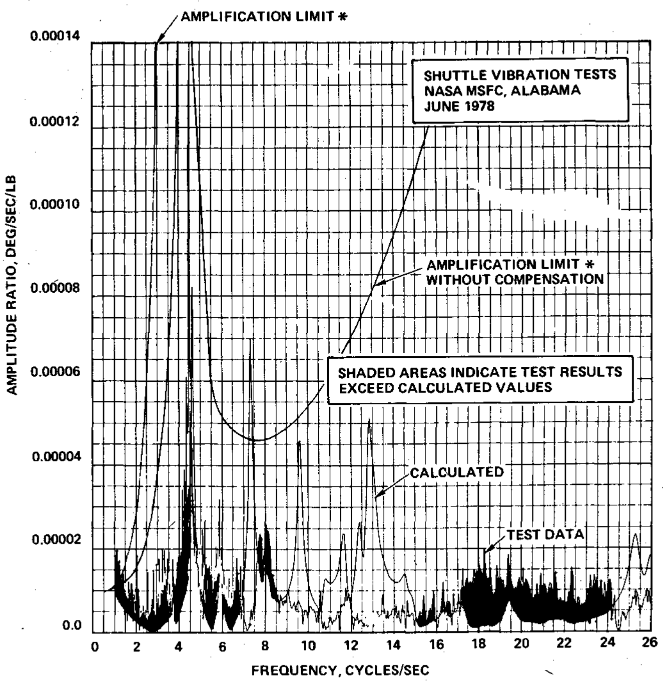
Figures 6 and 7 show the unnormalized response of yaw rate gyroscopes mounted on the booster rockets. These results are from the beginning of stage 1 test condition. Figure 6 shows data from the right booster rocket location, and Fig. 7 shows data from the left booster rocket location. A normalizing factor of 6.5 can be used to reduce these data to a unit force basis. The similarities in the data from the left- and right-hand locations are obvious, as are those between the calculated and observed results.

Since roll rate gyroscopes are not mounted on the booster rockets, Figs. 8 and 9 present unnormalized response of orbiter-mounted roll rate gyroscopes. Figure 8 shows data for the beginning of stage 1 test condition, and Fig. 9 presents results from the end of stage 1 test condition. The normalizing factor to reduce these data to a unit force basis is 8.2. The results show that only smaller frequency vibration modes

Table 5 Designation of six stage 2 vibration tests and primary instruments of response[a]

Test no.	Operating vibration device	Pitch rate gyroscopes	Yaw rate gyroscopes	Roll rate gyroscopes	Normal accelerometers	Lateral acceleromet
1	Upper lateral		× (Figs. 11, 12)[a]			×
2	Lower lateral		×			×
3	Left normal	×		×	×	
4	Right normal	×		×	×	
5	Left and right normal in unison	× (Figs. 13, 14)[b]			×	
6	Left and right normal in opposition			× (Figs. 15, 16)[b]		

[a] × designates primary response instruments.
[b] Designates figure that shows representative results from this test.



* 68% ATTENUATION LIMIT

Fig. 12 Frequency response of flight control yaw rate gyroscope 1 (begin stage 2 test condition).

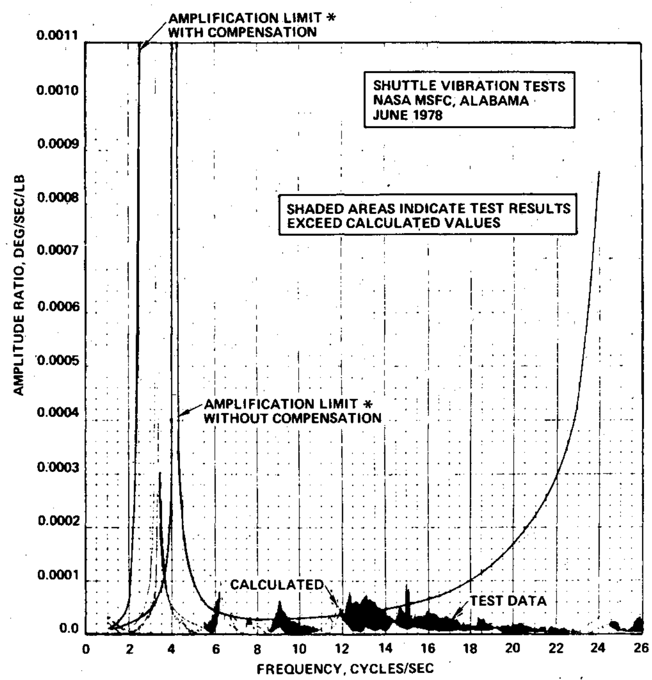


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Fig. 14 Frequency response of flight control pitch rate gyroscope 1 (begin stage 2 test condition).

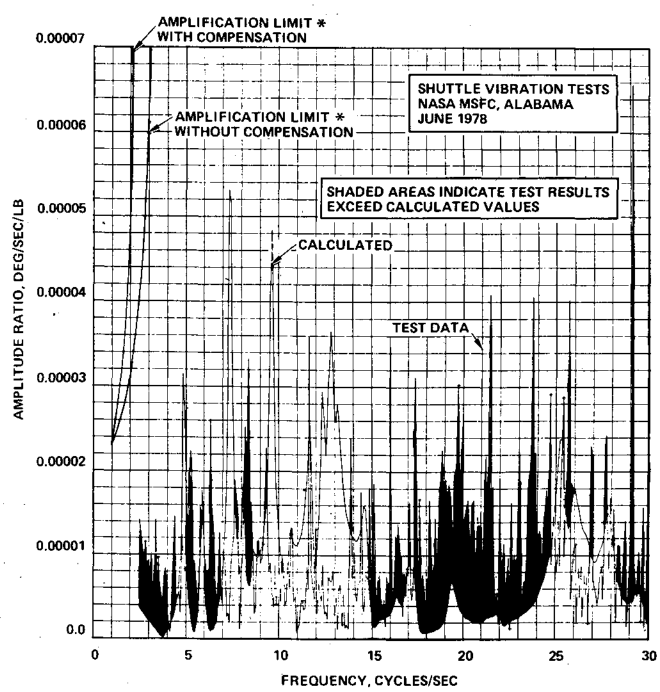

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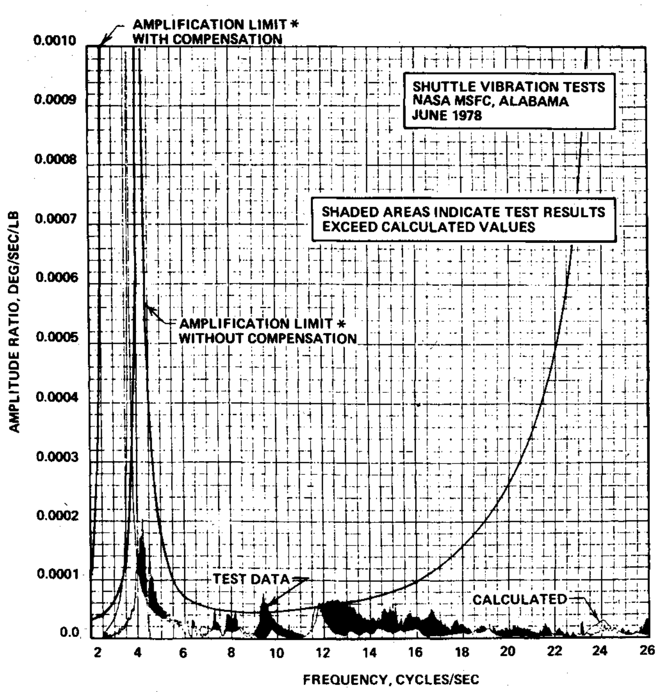
Fig. 13 Frequency response of flight control yaw rate gyroscope 1 (end stage 2 test condition).



* 68% ATTENUATION LIMIT

Fig. 15 Frequency response of flight control pitch rate gyroscope 1 (end stage 2 test condition).

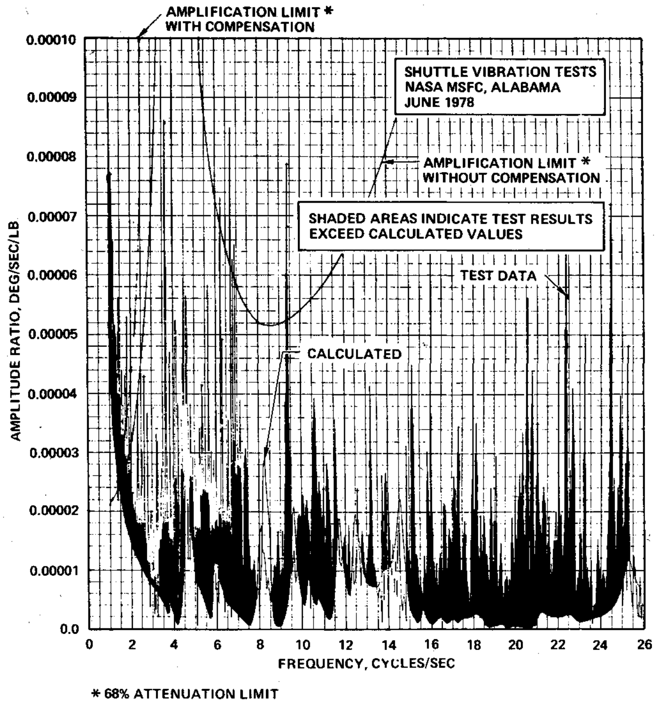


Fig. 16 Frequency response of flight control roll rate gyroscope 1 (begin stage 2 test condition).

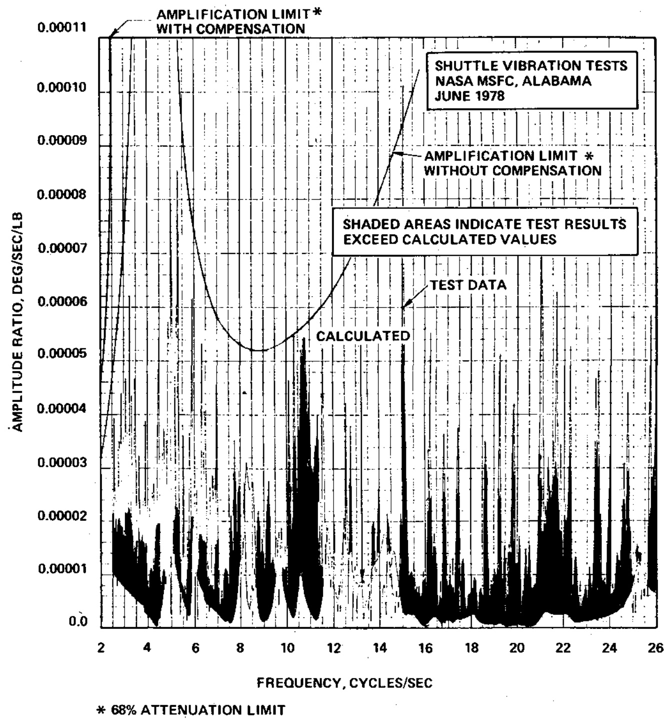


Fig. 17 Frequency response of flight control roll rate gyroscope 1 (end stage 2 test condition).

require additional (compensatory) attenuation to meet the control system design criterion, and proposed compensation functions place the amplification limit function well beyond the test data.

Figures 10 and 11 show unnormalized results from accelerometers sensitive to normal and lateral direction. Figure 10 shows response of the lateral accelerometer at the beginning of stage 1 test condition, whereas Fig. 11 shows the response of the normal accelerometer at the end of stage 1 test condition. The normalizing factor to reduce the data to a unit force basis (Fig. 10) is 6.5. The normalizing factor for the normal accelerometer data (Fig. 11) is 10.0.

Selected Results From Stage 2 Tests

The stage 2 test arrangement was similar to that shown schematically in Figs. 1 and 2, except the booster rockets and associated vibrators and instrumentation were excluded. The test procedure and data processing were the same as those described for the stage 1 tests.

During the vibration testing, information from 37 test and flight control rate gyroscopes and accelerometers was recorded for each of three test conditions. For each test condition, a series of six tests was run in which either a single vibrator or a combination of two together was used. These six tests, and the instruments of primary response, are given in Table 5.

A total of 666 transfer functions was available from these tests, but only selected results are presented in this summary. These include results from test series 1, 5, and 6. Test series 1 was selected to illustrate yaw rate gyroscope response, 5 to illustrate pitch rate gyroscope response, and 6 to illustrate roll rate gyroscope response. Only data from flight control instruments are shown because they are more significant than the data from test instrumentation. Accelerometer responses are not shown because they are not used during stage 2 flight. Finally, only data from the beginning and end of the stage 2 test condition are shown, since the midstage results are similar to the other test conditions.

Figures 12 and 13 show frequency response of flight control yaw rate gyroscopes during stage 2 vibration testing. Figure 12 shows response of the No. 1 yaw rate gyroscope mounted on the orbiter vehicle during the beginning of stage 2 test conditions, while Fig. 13 shows a similar result for the end of stage 2 test condition. The data are on a unit force basis, since only one vibrator was operating during these tests.

Figures 14 and 15 present unnormalized frequency responses of orbiter-mounted pitch rate gyroscopes. Figure 14 is for the beginning of stage 2 test condition, and Fig. 15 is for the end of stage 2 test condition. Since two normal direction vibrators operated in unison, a factor of 2 can be used to normalize these data to a unit force basis.

Figures 16 and 17 show unnormalized frequency response of orbiter-mounted roll rate gyroscopes. These data can be normalized to a unit force basis by dividing by a factor of 2, since two vibrators were operated in opposition to produce these results. Figure 16 represents the beginning of the stage 1 test condition, and Fig. 17 presents data for the end of the state 1 test condition.

Conclusion

The results show that flight control instrument response to flexible motion is well within amplification design limits. The predicted responses are generally substantiated by observed levels of responses and, in some instances, agree well in detail; however, in many instances, there is a lack of agreement in specific response between calculated and test results.

The calculated amplitudes of response could be made to match the test results by arbitrarily adjusting the value of the viscous damping factor used in the calculations at discrete frequencies. A constant value of 0.01 was used for calculation at all frequencies, whereas other related test results indicate an average value of 0.017 from stage 1 testing and 0.021 for stage 2 testing. However, it should be borne in mind that errors in mode shape and slope factors are equally significant in producing the observed differences between test results and theoretical calculations. Calculation also shows the average difference between the theoretical and observed frequencies of occurrence is 4% for the beginning of the stage 1 test condition.

Significant and essential flight control design data and knowledge of flexible motion transmission through the flight control instruments were obtained from these unique tests. This knowledge is especially important for the shuttle vehicle because of the employment of accelerometers as primary flight control instruments to provide a load alleviation control function. The imprecise agreement between theory and results is inconsequential to the control system design because planned flight control compensation functions provide more than adequate attenuation of signal content at flexible motion frequencies in either case.

Acknowledgments

The information in this paper represents the cumulative result of a large and complex test program involving extensive data processing and manipulation. Successful completion of the program depended on the knowledge and expertise of the many people who participated. The work was done by Rockwell personnel under contract to NASA. Some of the participants deserving mention for their contributions are R. Epple, R. Altenbach, T. Kaya, M. Bustamante, L. Sesto, R. Stone, S. Selkovitz, L. Selmar, M. Mazur, and R. Miller.

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