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# REQUIREMENTS AND FEASIBILITY STUDY OF FLIGHT DEMONSTRATION OF ACTIVE CONTROLS TECHNOLOGY (ACT) ON THE NASA 515 AIRPLANE

by: C. K. Gordon

PREPARED BY:

THE BOEING COMPANY
WICHITA DIVISION - WICHITA, KANSAS 67210



For Langley Research Center

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# REQUIREMENTS AND FEASIBILITY STUDY OF FLIGHT DEMONSTRATION OF ACTIVE CONTROLS TECHNOLOGY (ACT) ON THE NASA 515 AIRPLANE

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By: C. K. Gordon

Prepared Under Contract NAS1-13061 by

THE BOEING COMPANY
Wichita Division
Wichita, Kansas 37210

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OF FLIGHT DEMONSTRATION OF ACTIVE CONTROLS TECHNOLOGY (ACT)

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## SUMMARY

This technical report was prepared by Boeing under National Aeronautics and Space Administration (NASA) contract NAS1-13061. Justification of a flight validation program was developed in terms of technical requirements. A preliminary design study was conducted to evaluate the suitability of the NASA 515 airplane as a flight demonstration vehicle, and to develop plans, schedules, and budget costs for Fly-By-Wire (FBW)/Active Controls Technology (ACT) flight validation on the NASA 515 airplane. The preliminary design and planning were accomplished for two phases (or levels) of flight validation.

The technical objectives for the flight demonstration programs were developed as a result of the assessment of demonstration requirements. The purpose of a Phase I would be to start ACT flight demonstrations with the NASA 515 at the earliest possible date. The general guidelines are that Phase I must not seriously interfere with the Terminal Configured Vehicle (TCV) program functions and schedule, and must be low-cost.

It was determined that the greatest need for flight research is to develop techniques for digital implementation of FBW/ACT for large commercial transports and flight verify that the techniques are safe, reliable and cost effective.

Five flight validation program options were developed for Phase I, each designed and planned to be completely independent of all other options, and each addressing specific deficiencies in ACT technology. The objective of Phase I options is flight validation; that is:

- validation of ACT concepts in several specified areas
- validation of ACT digital system performance
- validation of ACT analytical and flight test techniques for design of commercial aircraft.

The purpose of Phase II (Option 6) is development, as opposed to the validation in Phase I. Specifically, the purpose is to develop and flight validate FBW/ACT digital implementation for large commercial aircraft. The goals are to develop hardware and software design techniques and criteria and to flight validate system design guidelines, performance (over the full

flight envelope), system compatibility, failure immunity, and implementation techniques.

Table I lists the options in order of priority, with the ACT technology deficiency satisfied by the option and the estimated cost. The priorities are based on an evaluation of cost effectiveness; i.e., the technology advancement per dollar. Option 6 not only satisfies technology deficiencies addressed by Options 1-5, but demonstrates compatibility among the ACT systems, over the full envelope. Option 6 also provides significant advancement in the development of digital/FBW hardware and software implementation techniques. The scope of Option 6 (Phase II) includes all the concepts from Options 1 through 5, plus maneuver load control.

The schedule for Phase II (Option 6) would require approximately 36 months, through flight checkout and delivery to NASA. Any of the five options for Phase I would require approximately 17 months.

The NASA 515 is an excellent test bed with which to evaluate digital system performance. The airplane has been equipped as a commercial type research vehicle for studies in digital navigation, displays and flight controls. Analyses show that the structural dynamic characteristics of the 737-100 are quite adequate for the flight demonstrations outlined in Table I.

TABLE I

SUMMARY OF NASA 515 PROGRAM OPTIONS, PURPOSE AND COST

| Priority | Option (As Numbered For Study) | ACT Concepts | ACT Deficiency Satisfied | Estimated Cost |
|---|---|---|---|---|
| 1 | 6 | All ACT/FBW | Develop FBW/ACT digital implementation for large commercial aircraft and verify:<br>- hardware and software design techniques and criteria<br>- performance over full flight envelope<br>- system compatibility<br>- failure immunity<br>- implementation techniques | $8.9M |
| 2 | 1 | Ride control via direct lift and direct lift for maneuvering | Validate:<br>- Concept of ride control via direct lift<br>- Handling qualities | $2.0M |
| 3 | 2 | Gust load alleviation (wing root) | Validate modal suppression performance with digital systems | $2.7M |
| 4 | 4 | Relaxed static stability and automatic c.g. control | Validate:<br>- Handling qualities<br>- Range improvements | $2.3M |
| 5 | 3 | Envelope limiting | Validate concept | $1.5M |
| 6 | 5 | Ride control (modal suppression) | Validate modal suppression performance with digital systems | $2.4M |

## 1.0 INTRODUCTION

This report describes a study accomplished under Contract NAS1-13061 to evaluate the suitability of the National Aeronautics and Space Administration (NASA) 515 airplane as a flight test vehicle for validation of Active Controls Technology (ACT).

The NASA 515 airplane is a 737 which is specially equipped with advanced electronic display, navigation and flight control equipment developed under NASA and Department of Transportation contracts. The airplane, with its special equipment, is called the Research Support Flight System (RSFS). The general arrangement of the 737-100 airplane is shown in Figure 1, with principal dimensions. The internal arrangements of the RSFS is illustrated in Figure 2. The RSFS features an aft flight deck (AFD), from which a two-man crew can fly the airplane through controls electrically coupled with the hydraulic control-surface actuators (i.e., fly-by-wire). The control and display systems designed for the RSFS by the Boeing Commercial Airplane Company in the Terminal Configured Vehicle (TCV) program, presently being flight tested by NASA, required extensive technology development. The TCV program addresses the problems of integrating aircraft pilot displays and controls in the advanced high density airport environment.

It is logical to utilize the commercial type navigation and controls research features of the RSFS for appropriate ACT flight demonstrations. Fly-by-wire (FBW) control techniques are required to integrate multiple active control and command augmentation modes. Digital FBW control is the basic element of all NASA 515 ACT program plans developed during this study, taking advantage of the existing RSFS digital equipment.

The word "validation" is used throughout this document, and means the establishment of the truth or reality of a concept, theory, claim or prediction, or establishment of the soundness of a design technique. "Verification" is used synonomously. "Flight demonstration" indicates a method of validation, and a <u>successful</u> demonstration is implied.

The flight validation tasks required to advance active control technology are assessed in the subject study. The NASA 515 airframe and system characteristics are analyzed to determine the feasibility of its fulfilling the ACT flight demonstration requirements. Flight validation program plans, schedules and budget planning cost estimates are developed for two phases (or levels) of flight validation.

The basic differences in the guidelines for Phase I and Phase II are that Phase I must interface with the TCV program and be low-cost. Therefore, Phase I will have minimal interference with TCV functions and schedule and the low-cost requirements for Phase I precludes major modifications to the airframe. The study guidelines for the ACT demonstration phases, then, mostly pertained to logistics,

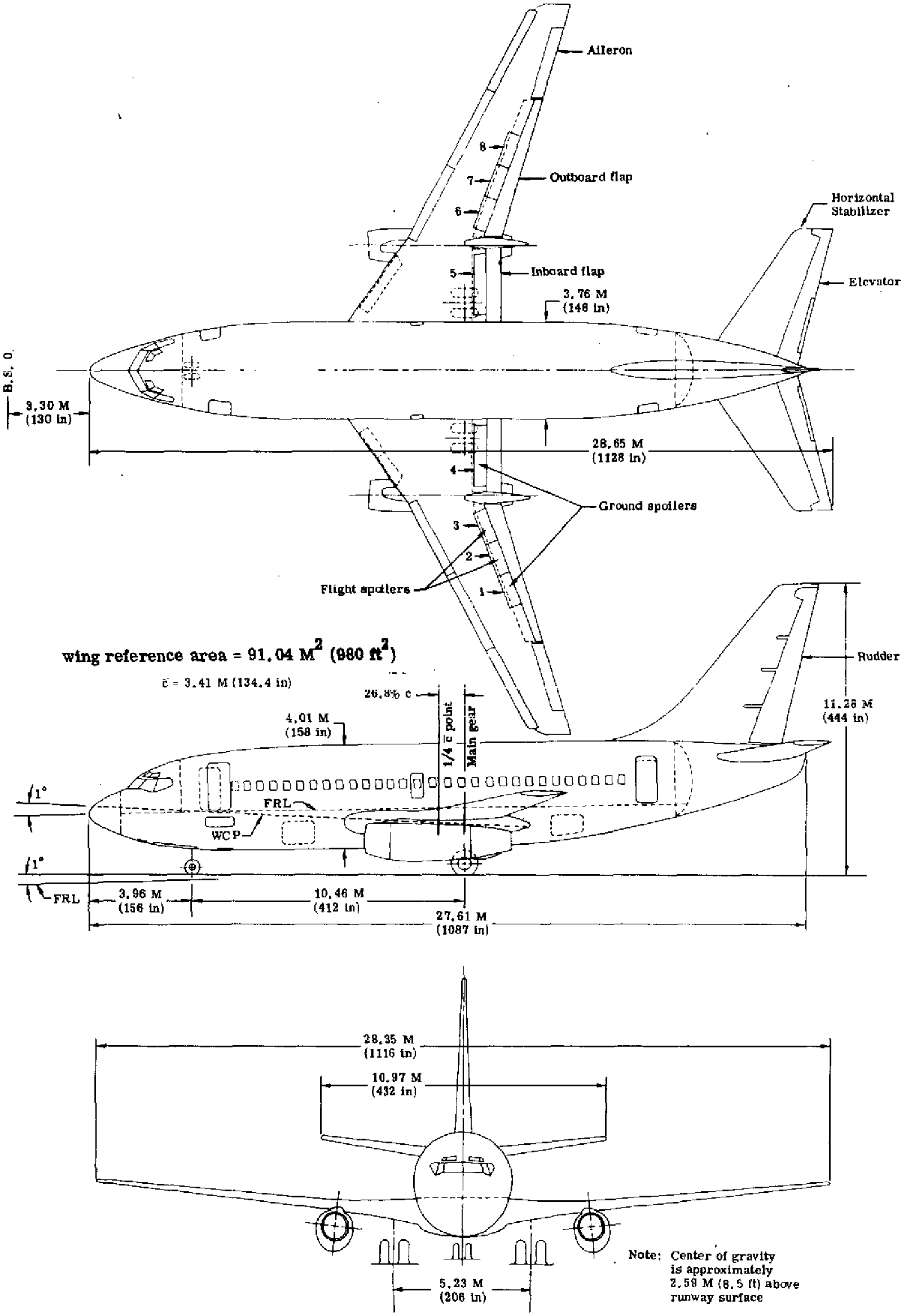


**Figure 1: General arrangement of 737-100 airplane**

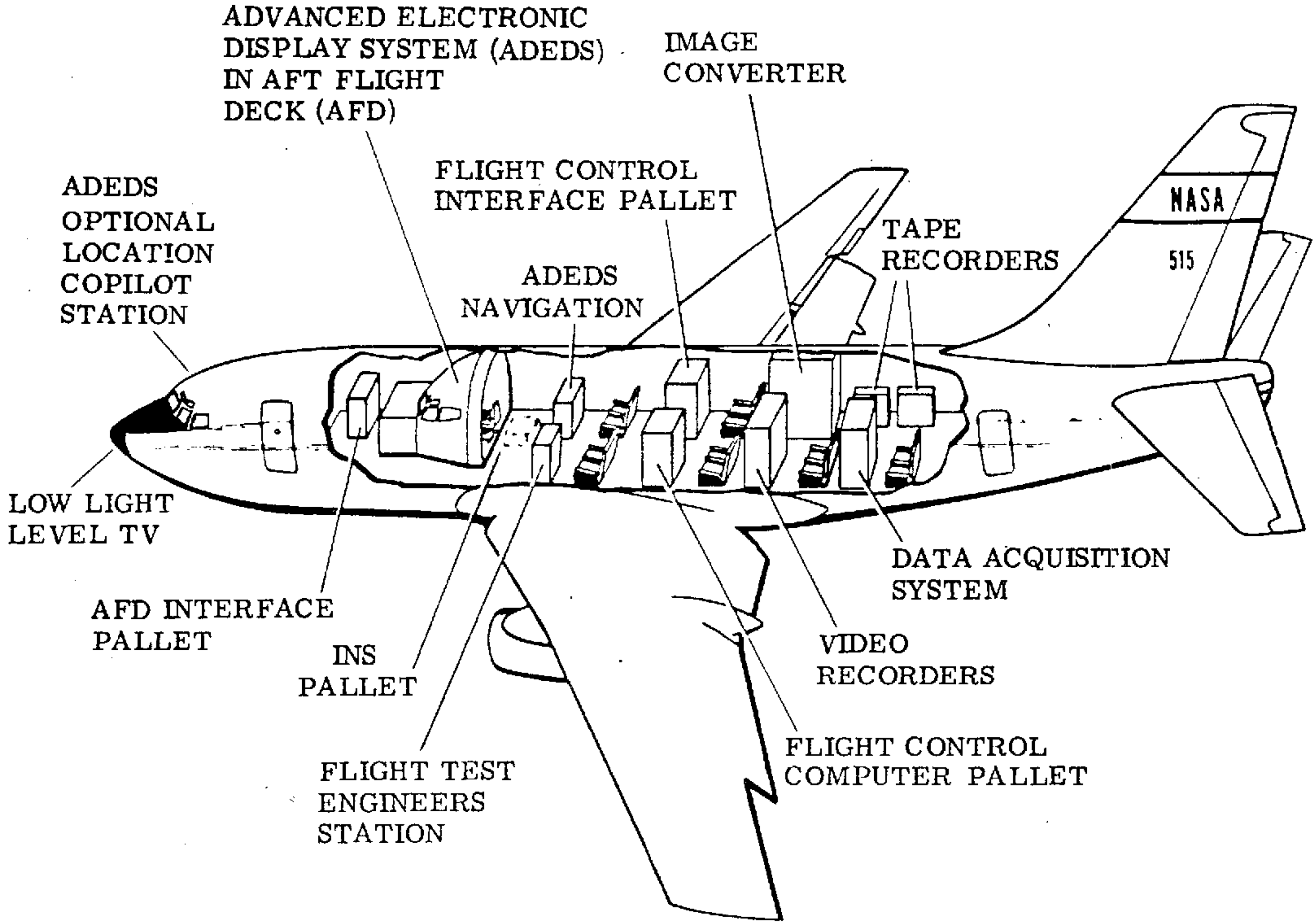


**Figure 2: Research support flight system (RSFS) internal arrangement**

schedules and costs. It was left as a task for this study to develop the demonstration phase technical objectives within these guidelines.

The following ACT concepts are considered in this study: (a) a relaxed static stability (RSS) system to reduce the size of stabilizing surfaces and the aft fuselage, (b) maneuver and gust load alleviation (MLA, GLA) systems to reduce structural strength requirements, (c) a flutter suppression (FS) system to reduce structural mass and stiffness requirements, (d) ride control (RC) systems, via both direct lift and modal control, to improve crew and passenger ride quality, (e) a center of gravity control (CGC) system to increase range and reduce fuel consumption, and (f) an envelope limiting (EL) system to limit flight operations to safe conditions.

Section 2 of this document defines the symbols, subscripts and abbreviations used in the text and figures. Section 3 discusses the technology elements involved in the validation of ACT concepts that will require flight demonstration, and assesses the flight demonstration status and requirements of each concept.

Section 4 describes continuous signal analyses performed to determine the performance potential of the RSFS for demonstration of the various concepts. This potential will be utilized to validate digital implementation of the various concepts designed as a part of the flight demonstration phases.

In Section 5, the practicability of implementation of the various concepts is considered in light of the demonstration phase guidelines, along with the analysis results of Section 4, to formulate a conclusion as to whether meaningful flight validations are feasible. Phase I and Phase II flight demonstration programs are defined in Section 5, involving six program options (five of them for Phase I).

Section 6 suggests guidelines and criteria for the NASA 515 ACT flight demonstration programs. Section 7 describes a preliminary design that was accomplished for each program option to define representative systems for cost estimating.

Section 8 presents demonstration program cost and schedule estimates, and Section 9 contains conclusions and recommendations resulting from this study.

## 2.0 SYMBOLS, SUBSCRIPTS AND ABBREVIATIONS

SYMBOLS:

| | |
|---|---|
| $\bar{c}$ | mean aerodynamic chord, M (ft) |
| f | frequency, hz |
| g | structural mode damping, equals minus two times damping ratio |
| g | normalized acceleration |
| H, h | altitude, M (ft) |
| j | $\sqrt{-1}$, unit imaginary number |
| K | unspecified gain constant |
| L | turbulence scale length, M (ft) |
| M | Mach number |
| $\bar{q}$ | dynamic pressure, $N/M^2$ ($lb/ft^2$) |
| S | Laplace operator |
| $U_o$ | steady flight velocity along X body axis, M/sec (ft/sec) |
| u | perturbation velocity along X body axis, M/sec (ft/sec) |
| V | total airplane velocity, M/sec (ft/sec) |
| w | perturbation velocity along Z body axis, M/sec (ft/sec) |
| $\ddot{Z}$ | inertial acceleration along Z body axis, g or $M/sec^2$ ($ft/sec^2$) |
| $\alpha$ | airplane angle of attack, rad |
| $\gamma$ | airplane flight path angle, rad |
| $\delta$ | control surface deflection, rad |
| $\theta$ | pitch attitude, rad |
| $\sigma$ | root mean squared |
| $\phi$ | roll attitude (Euler angle), rad |
| $\omega$ | frequency, rad/sec |

SUBSCRIPTS

| | |
|---|---|
| AIL, A | aileron |
| CAN, c | canard |
| c.g. | center of gravity |
| COL | column |
| COM | command |
| D | dive |
| DLC | direct lift control |
| e | elevator |
| g | gust |
| GLA | gust load alleviation |
| L.H. | left hand |
| M | modal |
| max | maximum |
| min | minimum |
| o | operational |
| RC | ride control |
| R.H. | right hand |
| SP | spoiler |
| SYM | symmetrical |
| $V_f$ | flutter velocity |
| W | vertical velocity |

ABBREVIATIONS

| | |
|---|---|
| AC | alternating current |
| ACT | active controls technology |
| A/D | analog to digital |

| | |
|---|---|
| ADEDS | Advanced Electronics Display System |
| AFD | aft flight deck |
| ALDCS | Active Lift Distribution Control System |
| AMP | ampere |
| APU | auxiliary power unit |
| AT | autothrottle |
| ATT | Advanced Technology Transport |
| BS | body station |
| CAS | command augmentation system |
| CCV | Controls Configured Vehicle |
| c.g. | center of gravity |
| CGC | c.g. control |
| CIU | Control Interface Unit |
| compl | complete |
| conv | conversion |
| cos | cosine |
| ctr | center |
| CWS | control wheel steering |
| D/A | digital to analog |
| db | decibels |
| D/D | digital to digital |
| deg | degree |
| DLC | direct lift control |
| EL | envelope limiting |
| elec | electrical |
| eq | equations |

| | |
|---|---|
| ext | extend |
| fab | fabrication |
| FBW | fly by wire |
| FCI | Flight Control Interface |
| flt | flight |
| FRC | Flight Research Center |
| FRL | fuselage reference line |
| FS | flutter suppression |
| ft | feet |
| fwd | forward |
| gal | gallon |
| GASDSAS | Gust Alleviation and Structural Dynamic Stability Augmentation System |
| G.E. | General Electric |
| gen | generator |
| GLA | gust load alleviation |
| GPM | gallons per minute |
| GV | gain variation |
| hyd | hydraulic |
| Hz | Hertz |
| ICPS | Incremental Control Processor Subsystem |
| in | inch |
| INS | inertial navigation system |
| instl | installation |
| integ | integration |
| I/O | input/output |
| KCAS | knots, calibrated airspeed |

| | |
|---|---|
| KEAS | knots, equivalent airspeed |
| KTAS | knots, true airspeed |
| lab | laboratory |
| LAMS | Load Alleviation and Mode Stabilization |
| lb | pound |
| L.H. | left hand |
| LRC | Langley Research Center |
| LVDT | linear variable differential transducer |
| M | meter |
| M.A.C. | mean aerodynamic chord |
| min | minute |
| MIN | minimum |
| MLA | maneuver load alleviation |
| mos | months |
| N | Newtons |
| NASA | National Aeronautics and Space Administration |
| nom | nominal |
| OWE | operating weight, empty |
| PCU | power control unit |
| PSD | power spectral density |
| PSI | pounds per square inch |
| PV | phase variation |
| QIA | quadraplex integrated actuator |
| QSA | quadraplex secondary actuator |
| QSE | quasi-static elastic |
| rad | radians |

| | |
|---|---|
| RC | ride control |
| ret | retract |
| RFP | request for proposal |
| R.H. | right hand |
| rms | root mean squared |
| rqmts | requirements |
| RSFS | Research Support Flight System |
| RSS | relaxed static stability |
| RTAC | Research and Technology Advisory Council |
| SAS | stability augmentation system |
| sec | second |
| spec | specification |
| sta | station |
| stab | stabilizer |
| STRU | Servo Transmitter/Receiver Unit |
| struct | structural |
| sys | system |
| TCV | Terminal Configured Vehicle |
| TED | trailing edge down |
| TEU | trailing edge up |
| VA | volt-Ampere |
| WBL | wing buttock line |
| WCP | wing chord plane |
| WL | water line |
| WWCS | Whole Word Computer Subsystem |

## 3.0 ACT FLIGHT VALIDATION REQUIREMENTS

A number of references were consulted in assessing ACT flight validation requirements in addition to applying Boeing experience. Three of the principal references are the NASA RTAC report[1], the NASA "Monograph" on design considerations for ACT[2], and the Air Force Flight Dynamics Laboratory paper on flight control system advances for near-future military aircraft[3]. Much of the phraseology in the following paragraphs is taken from these three sources.

Flight experience is the ultimate demonstration of concepts and design techniques. Commercial aircraft builders and airline acceptance depends upon flight verification[1]. Section 3.1 discusses the technology elements that require flight validation, and Section 3.2 discusses the flight validation status of each ACT concept. These evaluations of requirements and status were the basis for the definition of flight validation programs in Section 7.

### 3.1 Technology Elements Requiring Flight Validation

3.1.1 Analysis tools and concept performance. - The practicability of each concept must be shown by flight verification of the theory. The RTAC report, in its executive summary, recommends incorporating the ACT functions in existing aircraft and evaluating the best analytical tools for all essential disciplines by carefully correlating analyses, ground tests and flight tests. Sufficient demonstration of the accuracy and reliability of the analytical technology must be accomplished to integrate the concept into an initial design cycle.

Much of the concept validation has been accomplished by previous programs; e.g., B-52 LAMS and B-52 CCV. However, specific areas are identified in this report for which concept validation is yet required.

3.1.2 Design. The NASA ACT Monograph[2] states that a requirement exists to put all of the required ACT technology into practice in detailed designs, including signal distribution, actuation, sensors, hydraulic and electrical power supplies, and fault detection and isolation. As pointed out in the monograph mentioned above, there were numerous early problems with fully-powered controls, but virtually all of them involved design details and implementation rather than basic concepts, and the scenario could be repeated for ACT.

The ACT system design should be accomplished for a large airplane, representative of modern commercial transports. There are problems in control system design that are peculiar to large flexible aircraft because their structural mode frequencies cover a lower band than smaller aircraft, and their structural modes are more likely to couple dynamically with control loops. In addition, the possibility exists that dynamic coupling may occur between structural modes and the digital sampling frequency (or its harmonics), complicating the problem.

3.1.3 Digital implementation. - Several advantages are claimed for implementing ACT concepts with digital computation techniques. Reference 3 predicts that this will be the next major advancement in automatic flight controls. It states that digital computers are more flexible in integrating signals for the many control functions of advanced aircraft designs, and are becoming competitive with analog controls in cost, reliability, size and weight. Modification of control laws, gains, and built-in tests will be quicker and less expensive when implemented in digital software. Certain functions are more naturally implemented digitally, such as mode selection, gain scheduling (especially according to complex functions), blending of signals, built-in tests, failure isolation, and limiting.

The RTAC report asserts "Some computer hardware is unique to flight control applications and won't develop in the commercial market. The input/output sections of the flight control computer and the reliability, redundancy, and fault isolation software must be verified before the digital computer can be accepted in critical flight control applications."

The aircraft industry has not taken the position that digital flight controls are essential to incorporation of FBW/ACT. However, digital technology does have potential for advancements in automatic control capabilities, and flight research is definitely warranted to verify the claims for digital implementation. Flight demonstration of a digital FBW/ACT control system is needed on a large commercial aircraft. In a list of flight validation priorities, the RTAC report lists RSS first and digital flight controls second.

3.1.4 Interaction and compatibility of multiple concepts over the full flight envelope. - A summary of the NASA ACT Monograph panel consensus included a statement that a deficiency in the current state-of-the-art exists with respect to the integrated application of ACT functions. Numerous technology advancements, especially for large flexible vehicles, must be brought together to explore their interaction. As an example, the monograph states that reduction of structural requirements is significant only when maneuver and gust load control are practiced simultaneously. Boeing ATT studies showed that RSS plus CGC offered the largest payoff for that aircraft in terms of weight reduction.

With the exception of modal control for fatigue reduction and ride control, ACT flight validation has been conducted only at selected flight conditions. System concepts, designs and compatibility must be demonstrated throughout the flight envelope.

3.1.5 Safety and reliability. - Past programs have demonstrated performance benefits from ACT, but under carefully restricted research conditions. Further work is required to adequately demonstrate the safety and reliability of production type systems, especially on large flexible aircraft. This work must approach the full range of problems involved in producing failure immunity. This will include redundancy management of not only the sensors and computers, but of actuators, electrical power supplies, and hydraulic power supplies. Failure detection, isolation and compensation techniques must be developed for the appropriate levels of redundancy, and flight validated.

3.2 ACT Concept Flight Validation Status and Requirements

There are essentially three levels of design and flight test: proof of concept, prototype, and production. Figure 3 shows the status of flight validation for digital FBW and for the various ACT concepts. The most significant programs in the verification of each concept are indicated.

There have been many proof-of-concept programs for analog FBW. The Air Force B-52 LAMS (Load Alleviation and Mode Stabilization) and B-52 CCV (Control Configured Vehicles) research programs exemplified analog FBW with mechanical backup, as does the B-1 prototype. The Air Force F-4 Survivable Flight Control System research program successfully demonstrated a quadruply-redundant analog FBW system with no mechanical backup ("all FBW"). Analog FBW technology has developed to an analog all FBW control system in the Convair YF-16 prototype lightweight fighter.

Digital FBW technology is still in the proof-of-concept phase. The NASA-FRC research program on the F-8 is presently the most significant digital FBW flight validation program. The F-8 system presently has essentially single-thread digital electronics with triply-redundant analog electronics as backup. Later phases are planned, incorporating quadruply redundant digital all FBW. For the reasons stated in paragraphs 3.1.2, 3.1.3, and 3.1.5, proof-of-concept flight validation is still needed on a current state-of-the-art, operational, large commercial aircraft.

The B-52 CCV program demonstrated that good short period pitch responses and stick maneuvering forces can be attained at neutral longitudinal relaxed static stability. The Convair YF-16 prototype will incorporate RSS to reduce drag and gross weight. Additional research is required in the area of handling qualities, especially the speed stability, and the effect of actuator deadbands and hysteresis. The NASA ACT Monograph[2] also declares that flight experience is required to confirm predicted drag and weight reduction. The NASA RTAC report[1] places the highest flight validation priority among the FBW/ACT concepts on RSS.

Maneuver and Gust Load Alleviation concepts have been well demonstrated,

| ACT CONCEPT | PROOF OF CONCEPT DESIGN & TEST | PROTOTYPE DESIGN & TEST | PRODUCTION COMMITMENT |
|---|---|---|---|
| DIGITAL FBW | F-8 | | |
| RELAXED STATIC STABILITY | CCV B-52 | YF-16 | |
| GUST LOAD ALLEVIATION | LAMS | | B-52 ECP 1195<br>C-5A ALDCS<br>L-1011 |
| MANEUVER LOAD ALLEVIATION | CCV B-52 | | C-5A ALDCS |
| RIDE CONTROL | CCV B-52<br>XB-70 GASDSAS | B-1 | |
| FLUTTER SUPPRESSION | CCV B-52 | | |
| AUTOMATIC C.G. CONTROL | | B-1 | B-58 |
| ENVELOPE CONTROL | | YF-16 | |

**Figure 3: ACT flight validation status**

and commitments have been made to incorporate these concepts on existing airplanes, as indicated in Figure 3. In fact, GLA was incorporated in the L-1011 airplane design phase to reduce the design limit loads on the structure. No structural mode control systems have been implemented digitally, however. Modal suppression performance with digital implementation has yet to be validated. New problems are introducted by digital implementation, such as transport lag caused by the computing interval, lag associated with analog prefiltering that might be required, and dynamic coupling of the structural modes and the digital sampling frequencies.

Ride Control (RC) via modal suppression has also been well validated, as indicated by Figure 3, and is incorporated in the B-1 strategic bomber prototype to improve crew ride qualities during terrain following missions. Similar to GLA, the element yet to be accomplished in validation of RC via modal suppression is digital implementation.

Several studies have been conducted regarding the feasibility of the RC concept via direct lift, but no system design or flight test has been accomplished. The power of the random turbulence forcing function is concentrated at low frequencies, and it may be difficult to effectively control the c.g. translational acceleration response to gust without significantly affecting the airplane handling qualities. In addition, analyses show that exceptionally high frequency response and control surface rates are required, relative to other automatic rigid body control systems.

The validity of the flutter suppression concept was demonstrated in the B-52 CCV program when the airplane was flown to 10 KCAS above the flutter speed with the FS system on (more than 50 KCAS above placard speed). The mode being controlled was relatively low-frequency and destabilized gradually as speed was increased. Further research should treat multiple modes, higher frequency modes, and perhaps more violent modes. The RTAC report[1] suggests the use of remotely piloted vehicles for initial evaluation of such high risk problems.

The B-58 design included an automatic c.g. control (CGC) system, utilizing an analog computer, and the B-1 prototype has a quite complex CGC.

The flight envelope of many airplanes has been limited by scheduling certain parameters according to altitude and air data, such as pilot control forces, control surface authority, and normal acceleration. The Convair YF-16 prototype limits both airplane normal acceleration and angle of attack throughout the flight envelope, utilizing analog computation. The power of the digital computer to perform logic, to impose limits, and to schedule variables according to complex functions should be flight demonstrated for a system that would delimit angle-of attack, air speed, Mach number, and normal acceleration envelopes.

Specific gaps in FBW/ACT validation experience are cited in the foregoing paragraphs. A general deficiency exists in that, although analog implementation of most of the concepts has been demonstrated to some extent, and digital FBW flight research has been accomplished, there has been no flight validation of digital implementation of any of the ACT concepts.

Further, the safety and at least a qualitative indication of the reliability of FBW and critical ACT concepts have yet to be demonstrated in a way that will generate the confidence in the commercial aircraft industry required for acceptance of these concepts in new designs.

# 4.0 SYSTEM SYNTHESIS AND ANALYSIS

Syntheses and analyses were conducted in sufficient depth to determine concept feasibility, system configurations, and preliminary system performance. Airplane flexible and quasi-elastic mathematical models, developed at four representative flight conditions, were used in determining feedback variables, sensor locations, filters and gains for each ACT concept. Performance analyses were conducted to ensure that each system would provide measurable experimental performance improvements. An analog simulation of ACT systems and TCV control systems with potential interaction was conducted to study compatibility and verify that the proposed ACT systems did not degrade TCV system performance.

## 4.1 Flight Conditions

A single weight condition was analyzed --- fuel tanks approximately half full of fuel, with typical RSFS equipment, crew and passengers (13 total). The airplane weight at this condition is 346 960 N (78 000 lbs).

The flight conditions modeled were chosen from typical mission segments to provide analysis of the feasibility of effective flight demonstration of the respective concepts. The flight conditions are defined below.

| FLIGHT PHASE | ALTITUDE (M) | AIRSPEED KEAS |
|---|---|---|
| Climb | 3048 (10 000 Ft) | 340 (M = 0.62) |
| Cruise | 6096 (20 000 Ft) | 350 (M = 0.78) |
| Holding | 1524 (5000 Ft) | 230 (M = 0.38) |
| Landing | 0 | 118 (M = 0.18) |

The design flight conditions for the respective ACT concepts are indicated below:

| CONCEPT | SYNTHESIS CONDITION |
|---|---|
| Ride Control | Climb |
| Gust Load Alleviation | Climb |
| Envelope Limiting | (Only system concepts developed) |
| Flutter Suppression | Cruise |
| Relaxed Static Stability | Cruise |
| Maneuver Load Control | Holding |
| C.G. Control | (Synthesis not required) |

## 4.2 Airplane Mathematical Models

### 4.2.1 Elastic equations.

- The elastic equations of motion employed a beam element model of the elastic wing, body, tail and engine struts.

Twenty-two free-vibration modes were included in each axis (symmetric and antisymmetric), plus the gust and control surface forcing functions.

Doublet lattice models of the coupled wing, body, tail, and control surface aerodynamics were used in the equations and included lift growth and gust penetration. Rational polynomials were generalized and made a part of the mathematical model to interpolate between frequencies.

Systems synthesis and PSD analyses were accomplished with the elastic equations on digital computers.

### 4.2.2 Quasi-static elastic (QSE) equations.

- Small perturbation, linearized equations of motion for each axis (symmetric and antisymmetric) included three rigid body degrees of freedom, for which stability derivatives had been computed that approximated the effect of flexibility. These equations were formulated for the 737-100 in 1969 and have been used by The Boeing Commercial Airplane Company since then.

The QSE equations were used in a hybrid simulation to show the compatibility between ACT and TCV program systems.

## 4.3 Atmospheric Turbulence Mathematical Models

### 4.3.1 Random turbulence.

- Random atmospheric turbulence was modeled with a von Karman power spectrum having the following spectral density for digital computation of airplane response power spectra.

$$\phi_{w_g} = \frac{\sigma_{w_g}^2 L}{\pi U_o} \frac{1 + \frac{8}{3}\left(1.339 \frac{L}{U_o} \omega\right)^2}{\left[1 + \left(1.339 \frac{L}{U_o} \omega\right)^2\right]^{11/6}} \quad \frac{(\mathrm{M/sec})^2}{\mathrm{rad/sec}} \tag{1}$$

where $\sigma_{w_g}$ = rms gust velocity, M/sec.

$U_o$ = airplane forward velocity, M/sec.

$L$ = turbulence scale length, M.

$\omega$ = frequency, rad/sec.

A scale length (L) of 762 M (2500 ft) was used for the climb and cruise conditions, and 152 M (500 ft) for the landing condition.[4]

The analog simulation of random turbulence velocity was generated by filtering the output of a random white noise source so that the power spectral density of the filter output approximated the von Karman spectrum. The Laplace transform of the von Karman gust filter is:

$$F(S) = \frac{\sigma_{w_g} \sqrt{U_o}\; 2.229 \;(S + .317\frac{U_o}{L})(S + 11.54\frac{U_o}{L})(S + 166.3\frac{U_o}{L})}{\sqrt{L}\;(S + .372\frac{U_o}{L})(S + 1.372\frac{U_o}{L})(S + 17.79\frac{U_o}{L})(S + 264.8\frac{U_o}{L})} \quad (2)$$

4.3.2 Discrete gusts. - Discrete vertical gust forcing functions were used in the analog simulation, with a 1 - cos shape as specified by Federal Aviation Regulations, Part 25.341.

The gust shape was defined by:

$$V_g = 7.62 \left[1 - \cos \frac{2\pi Vt}{25\bar{c}}\right]$$

where

$V_g$ = vertical gust velocity, M/sec.

$V$ = true airspeed, M/sec.

$\bar{c}$ = mean aerodynamic wing chord, M.

This resulted in a 15.24 M/sec (50 ft/sec) discrete gust with a frequency of 0.7 Hz in the landing condition and 2.4 Hz in the climb condition.

## 4.4 Free Airplane Responses to Random Turbulence

Gust response analyses were conducted for the NASA 515 free airplane to identify potential areas of improvement with ACT. Power spectral densities and root-mean-squared data were computed for acceleration and bending moment responses of the free airplane to random vertical and lateral turbulence, at a number of body and wing stations. Summary data and conclusions are presented in the following paragraphs.

Figure 4 shows the rms vertical acceleration along the fuselage. The rms was computed for a frequency band of 0-25 rad/sec, which contains only the rigid body modes, and a frequency band of 25-80 rad/sec, which contains the structural modes. The structural modes have essentially no contribution to the total rms except at the pilot's station,

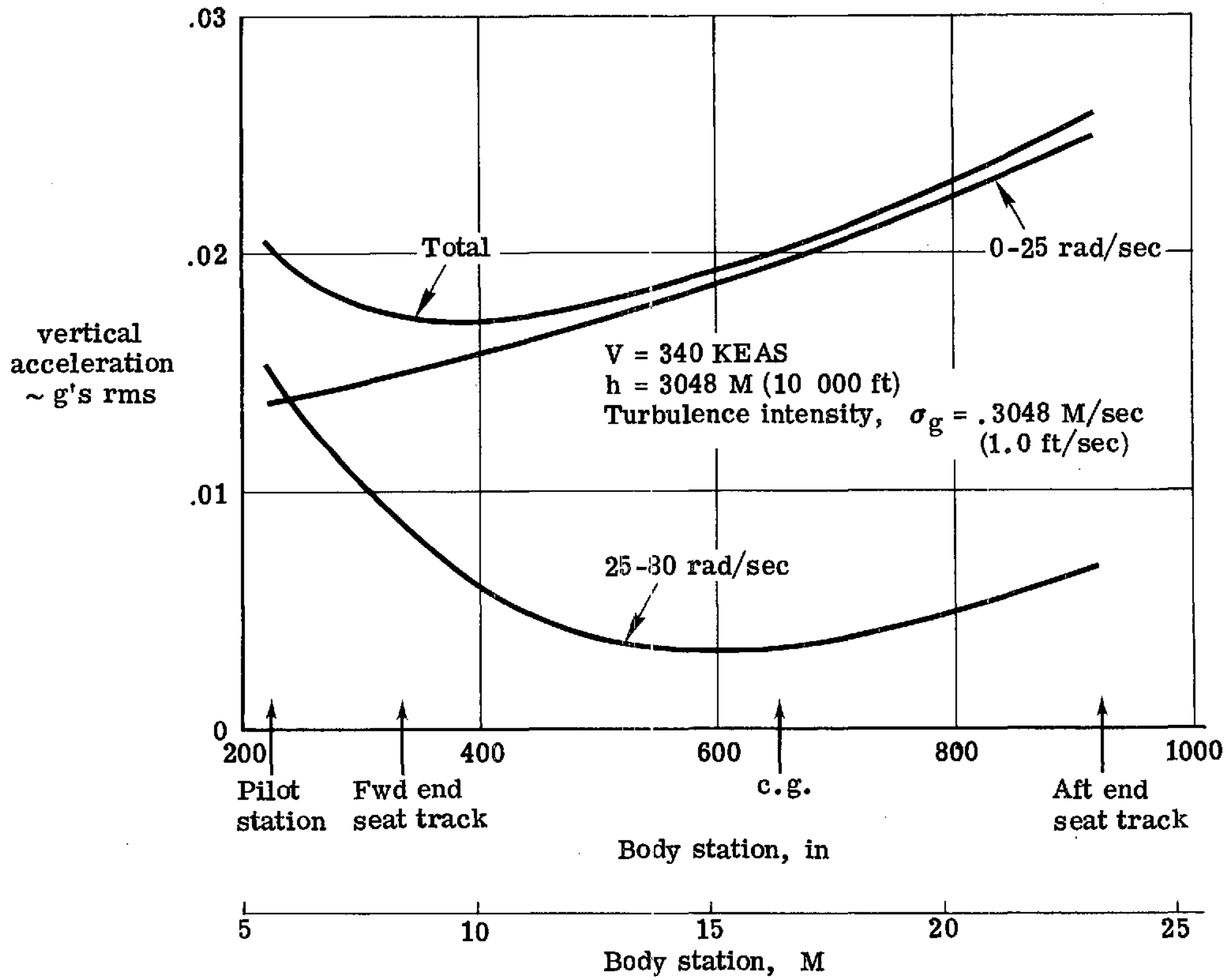


**Figure 4: Baseline airplane random vertical acceleration response**

where the contributions are about equal. Notice that the rms between 0-25 rad/sec (rigid body) is almost equal to the total, farther aft. At any frequency, $\sigma_{tot} = \sqrt{\sigma^2_{(0-25)} + \sigma^2_{(25-80)}}$.
A ride control system via structural mode control would have significant payoff only at the pilot's station.

A similar graph is presented for lateral accelerations in Figure 5. Data is shown in Figure 5 with and without the yaw damper, which is a part of the production 737 airplane. Again, structural mode suppression would not improve ride quality for the passengers, but would for the pilot. The existing yaw damper does reduce the aft body lateral accelerations quite well, but does not control the structural modes.

Figure 6 is a graph of the PSD and cumulative rms of the random vertical bending moment response at approximately the aft body cantilever point, BS 17.02 M (670 in). There is strong contribution to bending moment from the structural modes at this station, suggesting an appreciable payoff for an aft body fatigue reduction stability augmentation system (SAS). An exceptionally high-response elevator actuator would be required, with a bandpass of approximately sixty rad/sec.

The rms random lateral bending moment responses along the aft body are shown in Figure 7, based on straight line approximation between computations for BS 17.02 M (670 in) and BS 22.68 M (893 in). The rms bending moment for rigid body and structural mode frequency bands are presented. The results are quite similar to those for lateral accelerations along the aft body, shown previously. The greatest part of the free airplane total rms results from rigid body dynamics. The existing yaw damper does an adequate job of reducing the rigid body contribution, but does not affect the structural modes.

Figure 8 shows PSD/rms data for the random vertical bending moment at the wing root (approximate wing/body intersection). The rms component resulting from short period dynamics is dominant. However, there is sufficient contribution from a structural mode at approximately 22 rad/sec (3.5 Hz) to validate the digital implementation of a wing GLA system and demonstrate predicted performance.

## 4.5 ACT System Synthesis and Performance Analysis

Preliminary synthesis and performance analyses were performed for the various concepts to contribute to the decisions regarding flight validation feasibility, and to provide a prediction of performance that can be expected. Even better performance is likely from a more complete synthesis study phase of a flight validation program.

4.5.1 Ride control/direct lift. - A ride control (RC) system using direct lift control surfaces near the c.g. was synthesized. The direct lift surface ideally produce purely translational vertical acceleration of

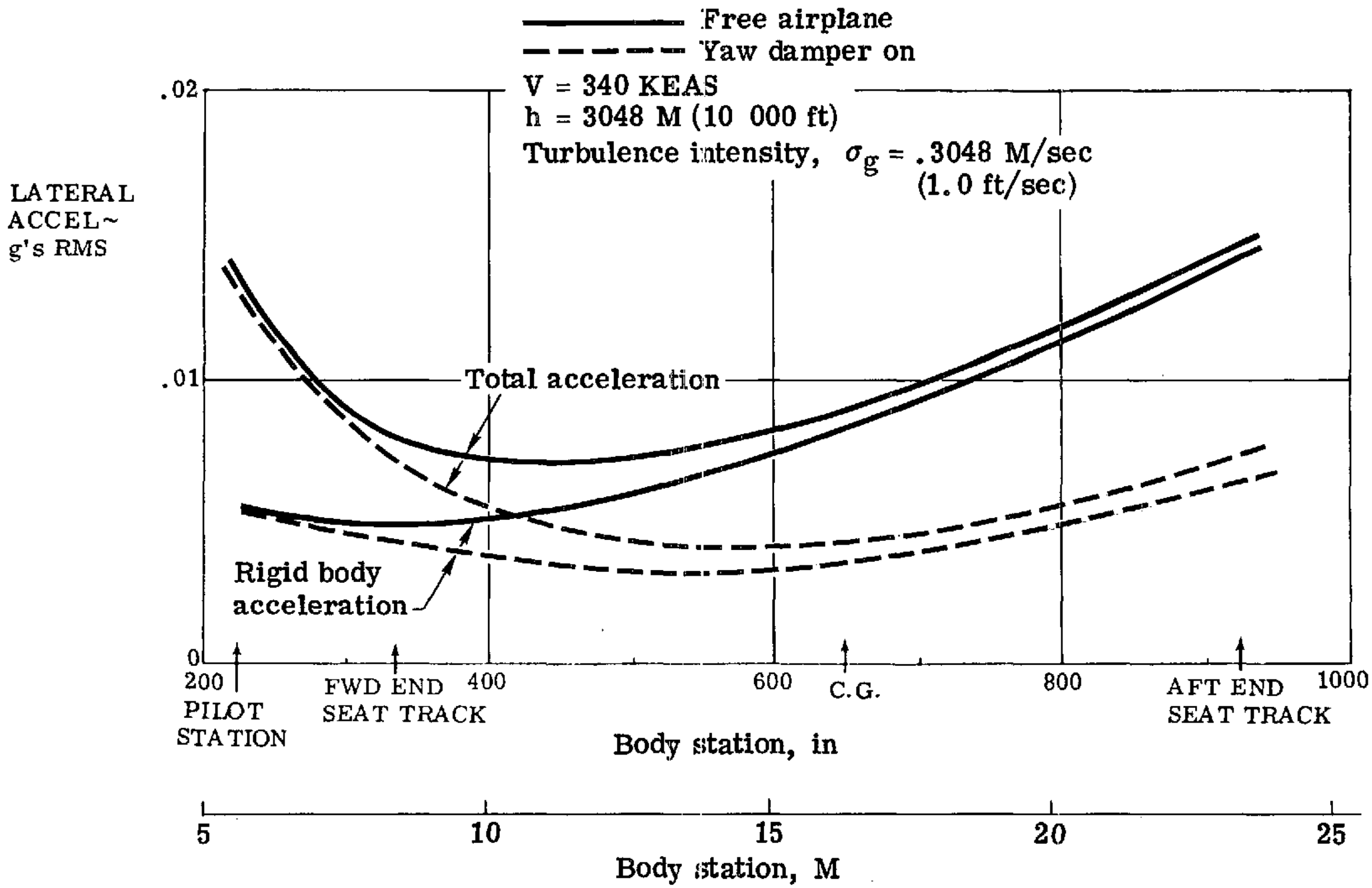


Figure 5: Baseline airplane random lateral accelerations

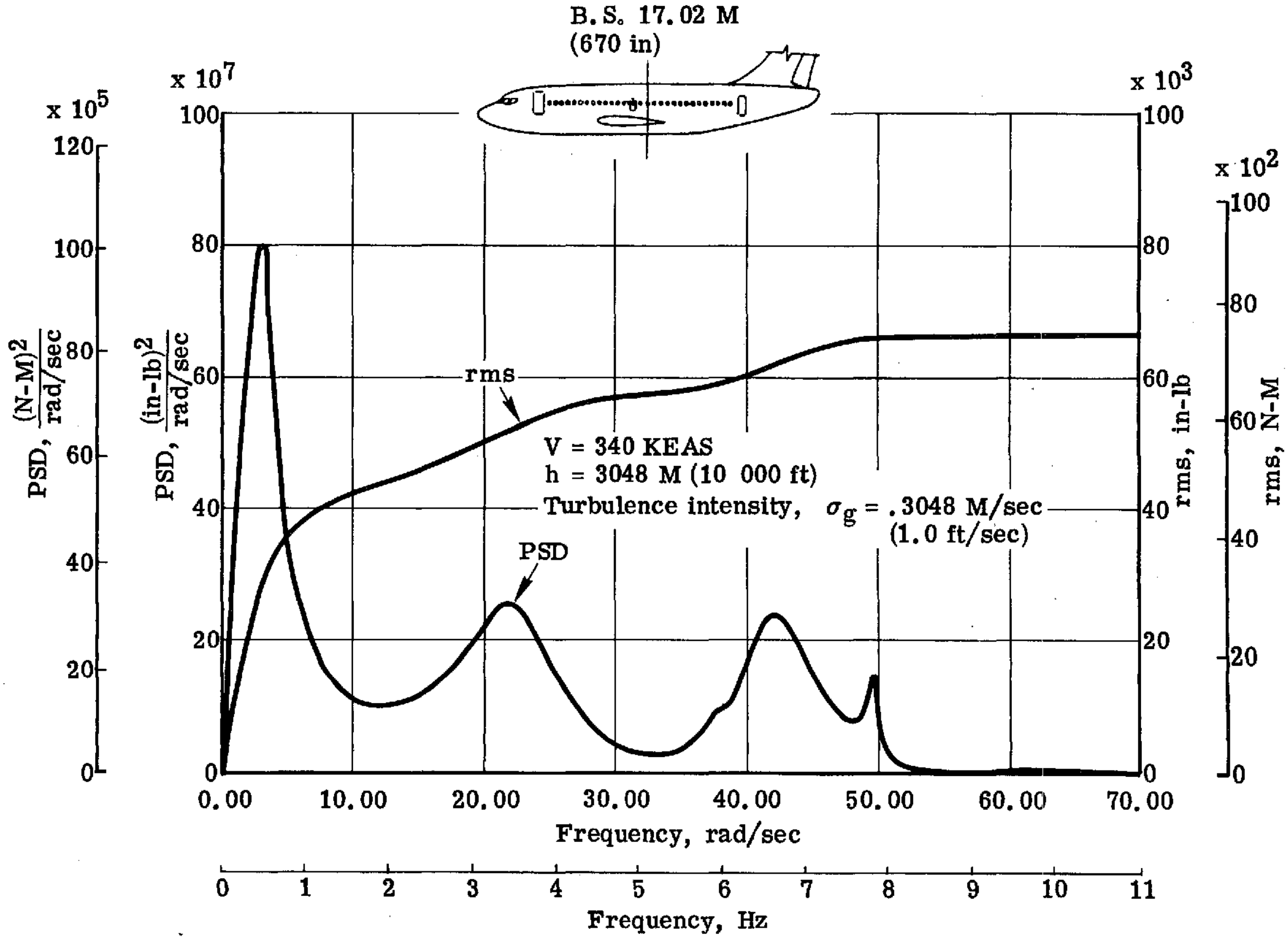


Figure 6: Mid-body random vertical bending moment response, baseline airplane

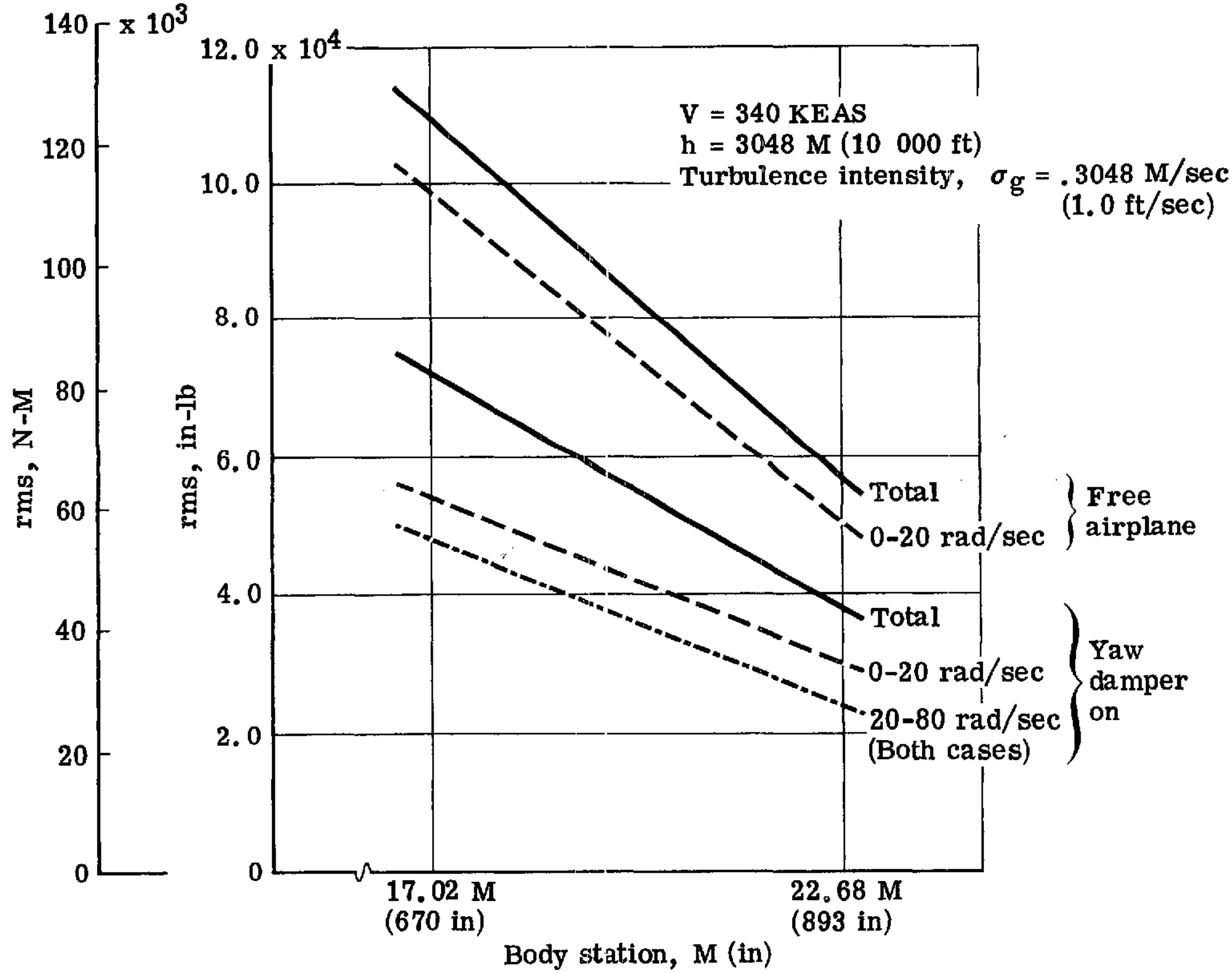


Figure 7: Aft body random lateral bending moment response, baseline airplane

NASA CR-132555

# REQUIREMENTS AND FEASIBILITY STUDY OF FLIGHT DEMONSTRATION OF ACTIVE CONTROLS TECHNOLOGY (ACT) ON THE NASA 515 AIRPLANE

TECHNICAL REPORT

By: C. K. Gordon

Prepared Under Contract NAS1-13061 by

THE BOEING COMPANY
Wichita Division
Wichita, Kansas 37210

For

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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REQUIREMENTS AND FEASIBILITY STUDY

OF FLIGHT DEMONSTRATION OF ACTIVE CONTROLS TECHNOLOGY (ACT)

By C. K. GORDON

THE BOEING COMPANY, WICHITA DIVISION

## SUMMARY

This technical report was prepared by Boeing under National Aeronautics and Space Administration (NASA) contract NAS1-13061. Justification of a flight validation program was developed in terms of technical requirements. A preliminary design study was conducted to evaluate the suitability of the NASA 515 airplane as a flight demonstration vehicle, and to develop plans, schedules, and budget costs for Fly-By-Wire (FBW)/Active Controls Technology (ACT) flight validation on the NASA 515 airplane. The preliminary design and planning were accomplished for two phases (or levels) of flight validation.

The technical objectives for the flight demonstration programs were developed as a result of the assessment of demonstration requirements. The purpose of a Phase I would be to start ACT flight demonstrations with the NASA 515 at the earliest possible date. The general guidelines are that Phase I must not seriously interfere with the Terminal Configured Vehicle (TCV) program functions and schedule, and must be low-cost.

It was determined that the greatest need for flight research is to develop techniques for digital implementation of FBW/ACT for large commercial transports and flight verify that the techniques are safe, reliable and cost effective.

Five flight validation program options were developed for Phase I, each designed and planned to be completely independent of all other options, and each addressing specific deficiencies in ACT technology. The objective of Phase I options is flight validation; that is:

- validation of ACT concepts in several specified areas
- validation of ACT digital system performance
- validation of ACT analytical and flight test techniques for design of commercial aircraft.

The purpose of Phase II (Option 6) is development, as opposed to the validation in Phase I. Specifically, the purpose is to develop and flight validate FBW/ACT digital implementation for large commercial aircraft. The goals are to develop hardware and software design techniques and criteria and to flight validate system design guidelines, performance (over the full

flight envelope), system compatibility, failure immunity, and implementation techniques.

Table I lists the options in order of priority, with the ACT technology deficiency satisfied by the option and the estimated cost. The priorities are based on an evaluation of cost effectiveness; i.e., the technology advancement per dollar. Option 6 not only satisfies technology deficiencies addressed by Options 1-5, but demonstrates compatibility among the ACT systems, over the full envelope. Option 6 also provides significant advancement in the development of digital/FBW hardware and software implementation techniques. The scope of Option 6 (Phase II) includes all the concepts from Options 1 through 5, plus maneuver load control.

The schedule for Phase II (Option 6) would require approximately 36 months, through flight checkout and delivery to NASA. Any of the five options for Phase I would require approximately 17 months.

The NASA 515 is an excellent test bed with which to evaluate digital system performance. The airplane has been equipped as a commercial type research vehicle for studies in digital navigation, displays and flight controls. Analyses show that the structural dynamic characteristics of the 737-100 are quite adequate for the flight demonstrations outlined in Table I.

TABLE I

SUMMARY OF NASA 515 PROGRAM OPTIONS, PURPOSE AND COST

| Priority | Option (As Numbered For Study) | ACT Concepts | ACT Deficiency Satisfied | Estimated Cost |
|---|---|---|---|---|
| 1 | 6 | All ACT/FBW | Develop FBW/ACT digital implementation for large commercial aircraft and verify:<br>- hardware and software design techniques and criteria<br>- performance over full flight envelope<br>- system compatibility<br>- failure immunity<br>- implementation techniques | $8.9M |
| 2 | 1 | Ride control via direct lift and direct lift for maneuvering | Validate:<br>- Concept of ride control via direct lift<br>- Handling qualities | $2.0M |
| 3 | 2 | Gust load alleviation (wing root) | Validate modal suppression performance with digital systems | $2.7M |
| 4 | 4 | Relaxed static stability and automatic c.g. control | Validate:<br>- Handling qualities<br>- Range improvements | $2.3M |
| 5 | 3 | Envelope limiting | Validate concept | $1.5M |
| 6 | 5 | Ride control (modal suppression) | Validate modal suppression performance with digital systems | $2.4M |

## 1.0 INTRODUCTION

This report describes a study accomplished under Contract NAS1-13061 to evaluate the suitability of the National Aeronautics and Space Administration (NASA) 515 airplane as a flight test vehicle for validation of Active Controls Technology (ACT).

The NASA 515 airplane is a 737 which is specially equipped with advanced electronic display, navigation and flight control equipment developed under NASA and Department of Transportation contracts. The airplane, with its special equipment, is called the Research Support Flight System (RSFS). The general arrangement of the 737-100 airplane is shown in Figure 1, with principal dimensions. The internal arrangements of the RSFS is illustrated in Figure 2. The RSFS features an aft flight deck (AFD), from which a two-man crew can fly the airplane through controls electrically coupled with the hydraulic control-surface actuators (i.e., fly-by-wire). The control and display systems designed for the RSFS by the Boeing Commercial Airplane Company in the Terminal Configured Vehicle (TCV) program, presently being flight tested by NASA, required extensive technology development. The TCV program addresses the problems of integrating aircraft pilot displays and controls in the advanced high density airport environment.

It is logical to utilize the commercial type navigation and controls research features of the RSFS for appropriate ACT flight demonstrations. Fly-by-wire (FBW) control techniques are required to integrate multiple active control and command augmentation modes. Digital FBW control is the basic element of all NASA 515 ACT program plans developed during this study, taking advantage of the existing RSFS digital equipment.

The word "validation" is used throughout this document, and means the establishment of the truth or reality of a concept, theory, claim or prediction, or establishment of the soundness of a design technique. "Verification" is used synonomously. "Flight demonstration" indicates a method of validation, and a successful demonstration is implied.

The flight validation tasks required to advance active control technology are assessed in the subject study. The NASA 515 airframe and system characteristics are analyzed to determine the feasibility of its fulfilling the ACT flight demonstration requirements. Flight validation program plans, schedules and budget planning cost estimates are developed for two phases (or levels) of flight validation.

The basic differences in the guidelines for Phase I and Phase II are that Phase I must interface with the TCV program and be low-cost. Therefore, Phase I will have minimal interference with TCV functions and schedule and the low-cost requirements for Phase I precludes major modifications to the airframe. The study guidelines for the ACT demonstration phases, then, mostly pertained to logistics,

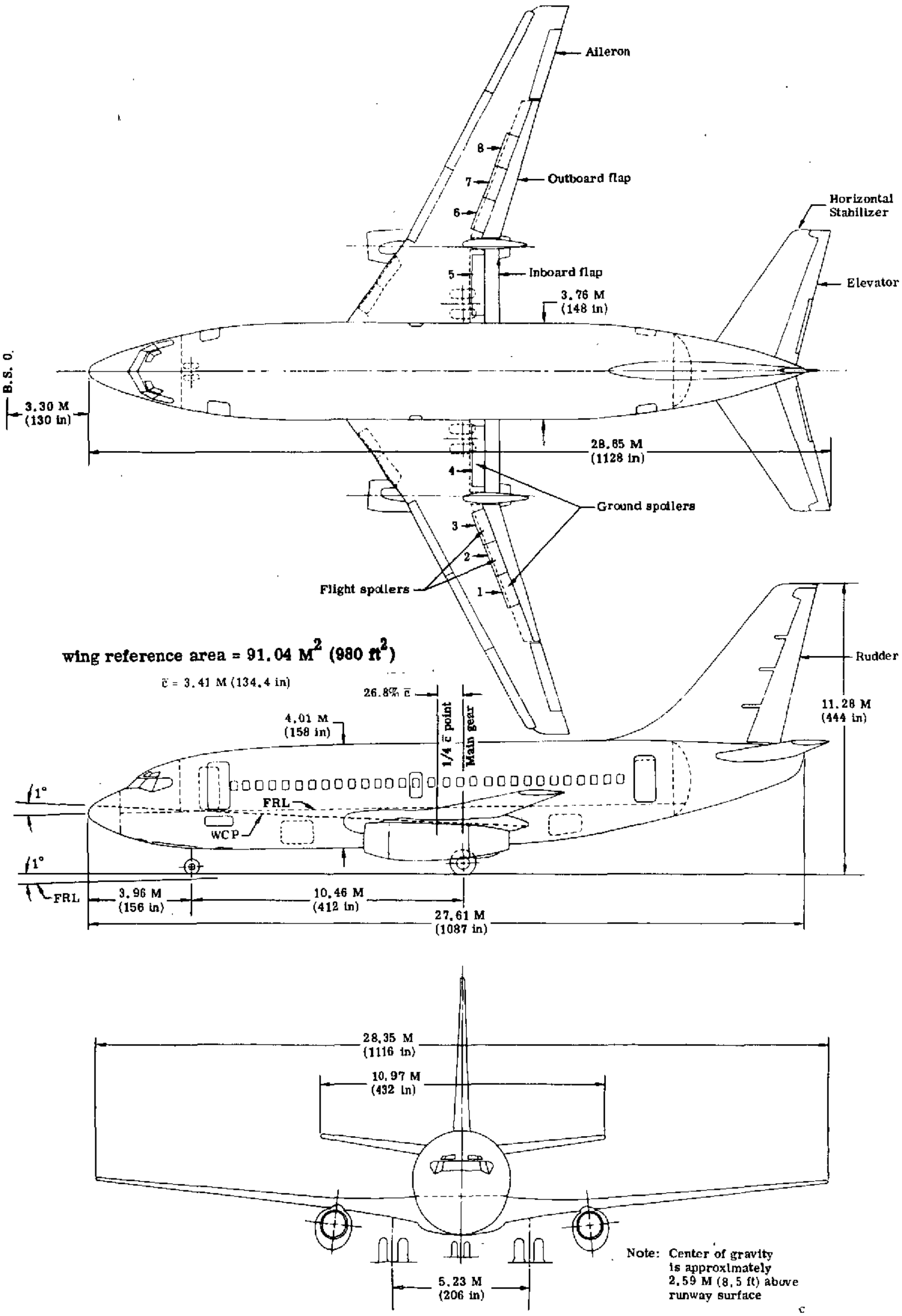



Figure 1: General arrangement of 737-100 airplane

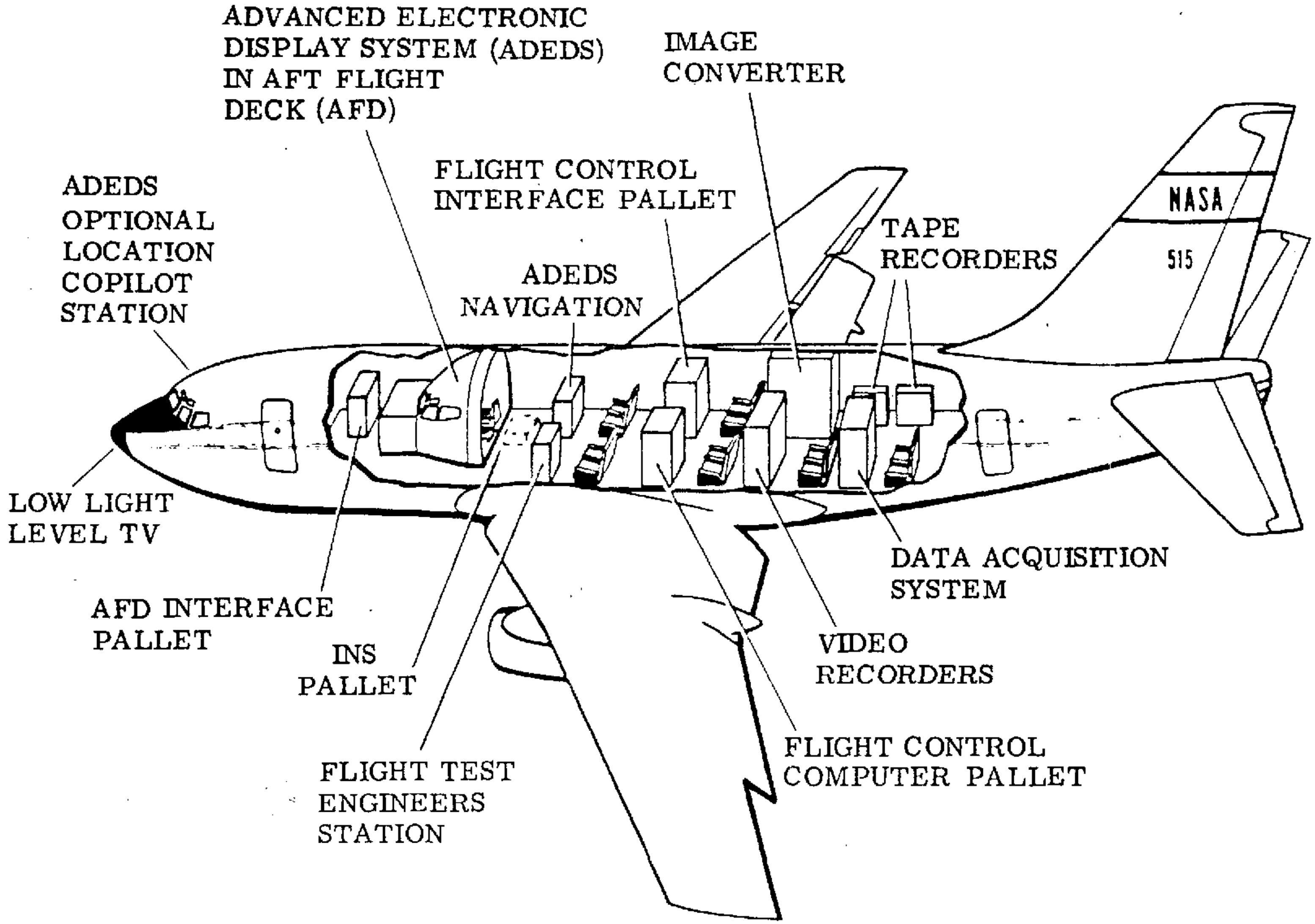



**Figure 2:** Research support flight system (RSFS) internal arrangement

schedules and costs. It was left as a task for this study to develop the demonstration phase technical objectives within these guidelines.

The following ACT concepts are considered in this study: (a) a relaxed static stability (RSS) system to reduce the size of stabilizing surfaces and the aft fuselage, (b) maneuver and gust load alleviation (MLA, GLA) systems to reduce structural strength requirements, (c) a flutter suppression (FS) system to reduce structural mass and stiffness requirements, (d) ride control (RC) systems, via both direct lift and modal control, to improve crew and passenger ride quality, (e) a center of gravity control (CGC) system to increase range and reduce fuel consumption, and (f) an envelope limiting (EL) system to limit flight operations to safe conditions.

Section 2 of this document defines the symbols, subscripts and abbreviations used in the text and figures. Section 3 discusses the technology elements involved in the validation of ACT concepts that will require flight demonstration, and assesses the flight demonstration status and requirements of each concept.

Section 4 describes continuous signal analyses performed to determine the performance potential of the RSFS for demonstration of the various concepts. This potential will be utilized to validate digital implementation of the various concepts designed as a part of the flight demonstration phases.

In Section 5, the practicability of implementation of the various concepts is considered in light of the demonstration phase guidelines, along with the analysis results of Section 4, to formulate a conclusion as to whether meaningful flight validations are feasible. Phase I and Phase II flight demonstration programs are defined in Section 5, involving six program options (five of them for Phase I).

Section 6 suggests guidelines and criteria for the NASA 515 ACT flight demonstration programs. Section 7 describes a preliminary design that was accomplished for each program option to define representative systems for cost estimating.

Section 8 presents demonstration program cost and schedule estimates, and Section 9 contains conclusions and recommendations resulting from this study.

## 2.0 SYMBOLS, SUBSCRIPTS AND ABBREVIATIONS

SYMBOLS:

| | |
|---|---|
| $\bar{c}$ | mean aerodynamic chord, M (ft) |
| f | frequency, hz |
| g | structural mode damping, equals minus two times damping ratio |
| g | normalized acceleration |
| H, h | altitude, M (ft) |
| j | $\sqrt{-1}$, unit imaginary number |
| K | unspecified gain constant |
| L | turbulence scale length, M (ft) |
| M | Mach number |
| $\bar{q}$ | dynamic pressure, $N/M^2$ $(lb/ft^2)$ |
| S | Laplace operator |
| $U_o$ | steady flight velocity along X body axis, M/sec (ft/sec) |
| u | perturbation velocity along X body axis, M/sec (ft/sec) |
| V | total airplane velocity, M/sec (ft/sec) |
| w | perturbation velocity along Z body axis, M/sec (ft/sec) |
| $\ddot{Z}$ | inertial acceleration along Z body axis, g or $M/sec^2$ $(ft/sec^2)$ |
| $\alpha$ | airplane angle of attack, rad |
| $\gamma$ | airplane flight path angle, rad |
| $\delta$ | control surface deflection, rad |
| $\theta$ | pitch attitude, rad |
| $\sigma$ | root mean squared |
| $\phi$ | roll attitude (Euler angle), rad |
| $\omega$ | frequency, rad/sec |

SUBSCRIPTS

| | |
|---|---|
| AIL, A | aileron |
| CAN, c | canard |
| c.g. | center of gravity |
| COL | column |
| COM | command |
| D | dive |
| DLC | direct lift control |
| e | elevator |
| g | gust |
| GLA | gust load alleviation |
| L.H. | left hand |
| M | modal |
| max | maximum |
| min | minimum |
| o | operational |
| RC | ride control |
| R.H. | right hand |
| SP | spoiler |
| SYM | symmetrical |
| $V_f$ | flutter velocity |
| W | vertical velocity |

ABBREVIATIONS

| | |
|---|---|
| AC | alternating current |
| ACT | active controls technology |
| A/D | analog to digital |

| | |
|---|---|
| ADEDS | Advanced Electronics Display System |
| AFD | aft flight deck |
| ALDCS | Active Lift Distribution Control System |
| AMP | ampere |
| APU | auxiliary power unit |
| AT | autothrottle |
| ATT | Advanced Technology Transport |
| BS | body station |
| CAS | command augmentation system |
| CCV | Controls Configured Vehicle |
| c.g. | center of gravity |
| CGC | c.g. control |
| CIU | Control Interface Unit |
| compl | complete |
| conv | conversion |
| cos | cosine |
| ctr | center |
| CWS | control wheel steering |
| D/A | digital to analog |
| db | decibels |
| D/D | digital to digital |
| deg | degree |
| DLC | direct lift control |
| EL | envelope limiting |
| elec | electrical |
| eq | equations |

| | |
|---|---|
| ext | extend |
| fab | fabrication |
| FBW | fly by wire |
| FCI | Flight Control Interface |
| flt | flight |
| FRC | Flight Research Center |
| FRL | fuselage reference line |
| FS | flutter suppression |
| ft | feet |
| fwd | forward |
| gal | gallon |
| GASDSAS | Gust Alleviation and Structural Dynamic Stability Augmentation System |
| G.E. | General Electric |
| gen | generator |
| GLA | gust load alleviation |
| GPM | gallons per minute |
| GV | gain variation |
| hyd | hydraulic |
| Hz | Hertz |
| ICPS | Incremental Control Processor Subsystem |
| in | inch |
| INS | inertial navigation system |
| instl | installation |
| integ | integration |
| I/O | input/output |
| KCAS | knots, calibrated airspeed |

| | |
|---|---|
| KEAS | knots, equivalent airspeed |
| KTAS | knots, true airspeed |
| lab | laboratory |
| LAMS | Load Alleviation and Mode Stabilization |
| lb | pound |
| L.H. | left hand |
| LRC | Langley Research Center |
| LVDT | linear variable differential transducer |
| M | meter |
| M.A.C. | mean aerodynamic chord |
| min | minute |
| MIN | minimum |
| MLA | maneuver load alleviation |
| mos | months |
| N | Newtons |
| NASA | National Aeronautics and Space Administration |
| nom | nominal |
| OWE | operating weight, empty |
| PCU | power control unit |
| PSD | power spectral density |
| PSI | pounds per square inch |
| PV | phase variation |
| QIA | quadraplex integrated actuator |
| QSA | quadraplex secondary actuator |
| QSE | quasi-static elastic |
| rad | radians |

| | |
|---|---|
| RC | ride control |
| ret | retract |
| RFP | request for proposal |
| R.H. | right hand |
| rms | root mean squared |
| rqmts | requirements |
| RSFS | Research Support Flight System |
| RSS | relaxed static stability |
| RTAC | Research and Technology Advisory Council |
| SAS | stability augmentation system |
| sec | second |
| spec | specification |
| sta | station |
| stab | stabilizer |
| STRU | Servo Transmitter/Receiver Unit |
| struct | structural |
| sys | system |
| TCV | Terminal Configured Vehicle |
| TED | trailing edge down |
| TEU | trailing edge up |
| VA | volt-Ampere |
| WBL | wing buttock line |
| WCP | wing chord plane |
| WL | water line |
| WWCS | Whole Word Computer Subsystem |

## 3.0 ACT FLIGHT VALIDATION REQUIREMENTS

A number of references were consulted in assessing ACT flight validation requirements in addition to applying Boeing experience. Three of the principal references are the NASA RTAC report[1], the NASA "Monograph" on design considerations for ACT[2], and the Air Force Flight Dynamics Laboratory paper on flight control system advances for near-future military aircraft[3]. Much of the phraseology in the following paragraphs is taken from these three sources.

Flight experience is the ultimate demonstration of concepts and design techniques. Commercial aircraft builders and airline acceptance depends upon flight verification[1]. Section 3.1 discusses the technology elements that require flight validation, and Section 3.2 discusses the flight validation status of each ACT concept. These evaluations of requirements and status were the basis for the definition of flight validation programs in Section 7.

### 3.1 Technology Elements Requiring Flight Validation

3.1.1 Analysis tools and concept performance. - The practicability of each concept must be shown by flight verification of the theory. The RTAC report, in its executive summary, recommends incorporating the ACT functions in existing aircraft and evaluating the best analytical tools for all essential disciplines by carefully correlating analyses, ground tests and flight tests. Sufficient demonstration of the accuracy and reliability of the analytical technology must be accomplished to integrate the concept into an initial design cycle.

Much of the concept validation has been accomplished by previous programs; e.g., B-52 LAMS and B-52 CCV. However, specific areas are identified in this report for which concept validation is yet required.

3.1.2 Design. The NASA ACT Monograph[2] states that a requirement exists to put all of the required ACT technology into practice in detailed designs, including signal distribution, actuation, sensors, hydraulic and electrical power supplies, and fault detection and isolation. As pointed out in the monograph mentioned above, there were numerous early problems with fully-powered controls, but virtually all of them involved design details and implementation rather than basic concepts, and the scenario could be repeated for ACT.

The ACT system design should be accomplished for a large airplane, representative of modern commercial transports. There are problems in control system design that are peculiar to large flexible aircraft because their structural mode frequencies cover a lower band than smaller aircraft, and their structural modes are more likely to couple dynamically with control loops. In addition, the possibility exists that dynamic coupling may occur between structural modes and the digital sampling frequency (or its harmonics), complicating the problem.

3.1.3 Digital implementation. - Several advantages are claimed for implementing ACT concepts with digital computation techniques. Reference 3 predicts that this will be the next major advancement in automatic flight controls. It states that digital computers are more flexible in integrating signals for the many control functions of advanced aircraft designs, and are becoming competitive with analog controls in cost, reliability, size and weight. Modification of control laws, gains, and built-in tests will be quicker and less expensive when implemented in digital software. Certain functions are more naturally implemented digitally, such as mode selection, gain scheduling (especially according to complex functions), blending of signals, built-in tests, failure isolation, and limiting.

The RTAC report asserts "Some computer hardware is unique to flight control applications and won't develop in the commercial market. The input/output sections of the flight control computer and the reliability, redundancy, and fault isolation software must be verified before the digital computer can be accepted in critical flight control applications."

The aircraft industry has not taken the position that digital flight controls are essential to incorporation of FBW/ACT. However, digital technology does have potential for advancements in automatic control capabilities, and flight research is definitely warranted to verify the claims for digital implementation. Flight demonstration of a digital FBW/ACT control system is needed on a large commercial aircraft. In a list of flight validation priorities, the RTAC report lists RSS first and digital flight controls second.

3.1.4 Interaction and compatibility of multiple concepts over the full flight envelope. - A summary of the NASA ACT Monograph panel consensus included a statement that a deficiency in the current state-of-the-art exists with respect to the integrated application of ACT functions. Numerous technology advancements, especially for large flexible vehicles, must be brought together to explore their interaction. As an example, the monograph states that reduction of structural requirements is significant only when maneuver and gust load control are practiced simultaneously. Boeing ATT studies showed that RSS plus CGC offered the largest payoff for that aircraft in terms of weight reduction.

With the exception of modal control for fatigue reduction and ride control, ACT flight validation has been conducted only at selected flight conditions. System concepts, designs and compatibility must be demonstrated throughout the flight envelope.

3.1.5 Safety and reliability. - Past programs have demonstrated performance benefits from ACT, but under carefully restricted research conditions. Further work is required to adequately demonstrate the safety and reliability of production type systems, especially on large flexible aircraft. This work must approach the full range of problems involved in producing failure immunity. This will include redundancy management of not only the sensors and computers, but of actuators, electrical power supplies, and hydraulic power supplies. Failure detection, isolation and compensation techniques must be developed for the appropriate levels of redundancy, and flight validated.

## 3.2 ACT Concept Flight Validation Status and Requirements

There are essentially three levels of design and flight test: proof of concept, prototype, and production. Figure 3 shows the status of flight validation for digital FBW and for the various ACT concepts. The most significant programs in the verification of each concept are indicated.

There have been many proof-of-concept programs for analog FBW. The Air Force B-52 LAMS (Load Alleviation and Mode Stabilization) and B-52 CCV (Control Configured Vehicles) research programs exemplified analog FBW with mechanical backup, as does the B-1 prototype. The Air Force F-4 Survivable Flight Control System research program successfully demonstrated a quadruply-redundant analog FBW system with no mechanical backup ("all FBW"). Analog FBW technology has developed to an analog all FBW control system in the Convair YF-16 prototype lightweight fighter.

Digital FBW technology is still in the proof-of-concept phase. The NASA-FRC research program on the F-8 is presently the most significant digital FBW flight validation program. The F-8 system presently has essentially single-thread digital electronics with triply-redundant analog electronics as backup. Later phases are planned, incorporating quadruply redundant digital all FBW. For the reasons stated in paragraphs 3.1.2, 3.1.3, and 3.1.5, proof-of-concept flight validation is still needed on a current state-of-the-art, operational, large commercial aircraft.

The B-52 CCV program demonstrated that good short period pitch responses and stick maneuvering forces can be attained at neutral longitudinal relaxed static stability. The Convair YF-16 prototype will incorporate RSS to reduce drag and gross weight. Additional research is required in the area of handling qualities, especially the speed stability, and the effect of actuator deadbands and hysteresis. The NASA ACT Monograph[2] also declares that flight experience is required to confirm predicted drag and weight reduction. The NASA RTAC report[1] places the highest flight validation priority among the FBW/ACT concepts on RSS.

Maneuver and Gust Load Alleviation concepts have been well demonstrated,

| ACT CONCEPT | PROOF OF CONCEPT DESIGN & TEST | PROTOTYPE DESIGN & TEST | PRODUCTION COMMITMENT |
|---|---|---|---|
| DIGITAL FBW | F-8 | | |
| RELAXED STATIC STABILITY | CCV B-52 | YF-16 | |
| GUST LOAD ALLEVIATION | LAMS | | B-52 ECP 1195<br>C-5A ALDCS<br>L-1011 |
| MANEUVER LOAD ALLEVIATION | CCV B-52 | | C-5A ALDCS |
| RIDE CONTROL | CCV B-52<br>XB-70 GASDSAS | B-1 | |
| FLUTTER SUPPRESSION | CCV B-52 | | |
| AUTOMATIC C. G. CONTROL | | B-1 | B-58 |
| ENVELOPE CONTROL | | YF-16 | |

**Figure 3: ACT flight validation status**

and commitments have been made to incorporate these concepts on existing airplanes, as indicated in Figure 3. In fact, GLA was incorporated in the L-1011 airplane design phase to reduce the design limit loads on the structure. No structural mode control systems have been implemented digitally, however. Modal suppression performance with digital implementation has yet to be validated. New problems are introducted by digital implementation, such as transport lag caused by the computing interval, lag associated with analog prefiltering that might be required, and dynamic coupling of the structural modes and the digital sampling frequencies.

Ride Control (RC) via modal suppression has also been well validated, as indicated by Figure 3, and is incorporated in the B-1 strategic bomber prototype to improve crew ride qualities during terrain following missions. Similar to GLA, the element yet to be accomplished in validation of RC via modal suppression is digital implementation.

Several studies have been conducted regarding the feasibility of the RC concept via direct lift, but no system design or flight test has been accomplished. The power of the random turbulence forcing function is concentrated at low frequencies, and it may be difficult to effectively control the c.g. translational acceleration response to gust without significantly affecting the airplane handling qualities. In addition, analyses show that exceptionally high frequency response and control surface rates are required, relative to other automatic rigid body control systems.

The validity of the flutter suppression concept was demonstrated in the B-52 CCV program when the airplane was flown to 10 KCAS above the flutter speed with the FS system on (more than 50 KCAS above placard speed). The mode being controlled was relatively low-frequency and destabilized gradually as speed was increased. Further research should treat multiple modes, higher frequency modes, and perhaps more violent modes. The RTAC report[1] suggests the use of remotely piloted vehicles for initial evaluation of such high risk problems.

The B-58 design included an automatic c.g. control (CGC) system, utilizing an analog computer, and the B-1 prototype has a quite complex CGC.

The flight envelope of many airplanes has been limited by scheduling certain parameters according to altitude and air data, such as pilot control forces, control surface authority, and normal acceleration. The Convair YF-16 prototype limits both airplane normal acceleration and angle of attack throughout the flight envelope, utilizing analog computation. The power of the digital computer to perform logic, to impose limits, and to schedule variables according to complex functions should be flight demonstrated for a system that would delimit angle-of attack, air speed, Mach number, and normal acceleration envelopes.

Specific gaps in FBW/ACT validation experience are cited in the foregoing paragraphs. A general deficiency exists in that, although analog implementation of most of the concepts has been demonstrated to some extent, and digital FBW flight research has been accomplished, there has been no flight validation of digital implementation of any of the ACT concepts.

Further, the safety and at least a qualitative indication of the reliability of FBW and critical ACT concepts have yet to be demonstrated in a way that will generate the confidence in the commercial aircraft industry required for acceptance of these concepts in new designs.

## 4.0 SYSTEM SYNTHESIS AND ANALYSIS

Syntheses and analyses were conducted in sufficient depth to determine concept feasibility, system configurations, and preliminary system performance. Airplane flexible and quasi-elastic mathematical models, developed at four representative flight conditions, were used in determining feedback variables, sensor locations, filters and gains for each ACT concept. Performance analyses were conducted to ensure that each system would provide measurable experimental performance improvements. An analog simulation of ACT systems and TCV control systems with potential interaction was conducted to study compatibility and verify that the proposed ACT systems did not degrade TCV system performance.

### 4.1 Flight Conditions

A single weight condition was analyzed --- fuel tanks approximately half full of fuel, with typical RSFS equipment, crew and passengers (13 total). The airplane weight at this condition is 346 960 N (78 000 lbs).

The flight conditions modeled were chosen from typical mission segments to provide analysis of the feasibility of effective flight demonstration of the respective concepts. The flight conditions are defined below.

| FLIGHT PHASE | ALTITUDE (M) | AIRSPEED KEAS |
|---|---|---|
| Climb | 3048 (10 000 Ft) | 340 (M = 0.62) |
| Cruise | 6096 (20 000 Ft) | 350 (M = 0.78) |
| Holding | 1524 (5000 Ft) | 230 (M = 0.38) |
| Landing | 0 | 118 (M = 0.18) |

The design flight conditions for the respective ACT concepts are indicated below:

| CONCEPT | SYNTHESIS CONDITION |
|---|---|
| Ride Control | Climb |
| Gust Load Alleviation | Climb |
| Envelope Limiting | (Only system concepts developed) |
| Flutter Suppression | Cruise |
| Relaxed Static Stability | Cruise |
| Maneuver Load Control | Holding |
| C.G. Control | (Synthesis not required) |

## 4.2 Airplane Mathematical Models

4.2.1 Elastic equations. - The elastic equations of motion employed a beam element model of the elastic wing, body, tail and engine struts.

Twenty-two free-vibration modes were included in each axis (symmetric and antisymmetric), plus the gust and control surface forcing functions.

Doublet lattice models of the coupled wing, body, tail, and control surface aerodynamics were used in the equations and included lift growth and gust penetration. Rational polynomials were generalized and made a part of the mathematical model to interpolate between frequencies.

Systems synthesis and PSD analyses were accomplished with the elastic equations on digital computers.

4.2.2 Quasi-static elastic (QSE) equations. - Small perturbation, linearized equations of motion for each axis (symmetric and antisymmetric) included three rigid body degrees of freedom, for which stability derivatives had been computed that approximated the effect of flexibility. These equations were formulated for the 737-100 in 1969 and have been used by The Boeing Commercial Airplane Company since then.

The QSE equations were used in a hybrid simulation to show the compatibility between ACT and TCV program systems.

## 4.3 Atmospheric Turbulence Mathematical Models

4.3.1 Random turbulence. - Random atmospheric turbulence was modeled with a von Karman power spectrum having the following spectral density for digital computation of airplane response power spectra.

$$\phi_{w_g} = \frac{\sigma_{w_g}^2 L}{\pi U_o} \frac{1+\frac{8}{3}\left(1.339 \frac{L}{U_o} \omega\right)^2}{\left[1+\left(1.339 \frac{L}{U_o} \omega\right)^2\right]^{11/6}} \quad \frac{(M/sec)^2}{rad/sec} \qquad (1)$$

where $\sigma_{w_g}$ = rms gust velocity, M/sec.

$U_o$ = airplane forward velocity, M/sec.

$L$ = turbulence scale length, M.

$\omega$ = frequency, rad/sec.

A scale length (L) of 762 M (2500 ft) was used for the climb and cruise conditions, and 152 M (500 ft) for the landing condition.[4]

The analog simulation of random turbulence velocity was generated by filtering the output of a random white noise source so that the power spectral density of the filter output approximated the von Karman spectrum. The Laplace transform of the von Karman gust filter is:

$$F(S) = \frac{\sigma_{w_g} \sqrt{U_o}\ 2.229\ (S + .317\frac{U_o}{L})(S + 11.54\frac{U_o}{L})(S + 166.3\frac{U_o}{L})}{\sqrt{L}\ (S + .372\frac{U_o}{L})(S + 1.372\frac{U_o}{L})(S + 17.79\frac{U_o}{L})(S + 264.8\frac{U_o}{L})} \qquad (2)$$

4.3.2 Discrete gusts. - Discrete vertical gust forcing functions were used in the analog simulation, with a 1 - cos shape as specified by Federal Aviation Regulations, Part 25.341.

The gust shape was defined by:

$$V_g = 7.62 \left[1 - \cos \frac{2\pi V t}{25\bar{c}}\right]$$

where

$V_g$ = vertical gust velocity, M/sec.

$V$ = true airspeed, M/sec.

$\bar{c}$ = mean aerodynamic wing chord, M.

This resulted in a 15.24 M/sec (50 ft/sec) discrete gust with a frequency of 0.7 Hz in the landing condition and 2.4 Hz in the climb condition.

4.4 Free Airplane Responses to Random Turbulence

Gust response analyses were conducted for the NASA 515 free airplane to identify potential areas of improvement with ACT. Power spectral densities and root-mean-squared data were computed for acceleration and bending moment responses of the free airplane to random vertical and lateral turbulence, at a number of body and wing stations. Summary data and conclusions are presented in the following paragraphs.

Figure 4 shows the rms vertical acceleration along the fuselage. The rms was computed for a frequency band of 0-25 rad/sec, which contains only the rigid body modes, and a frequency band of 25-80 rad/sec, which contains the structural modes. The structural modes have essentially no contribution to the total rms except at the pilot's station,

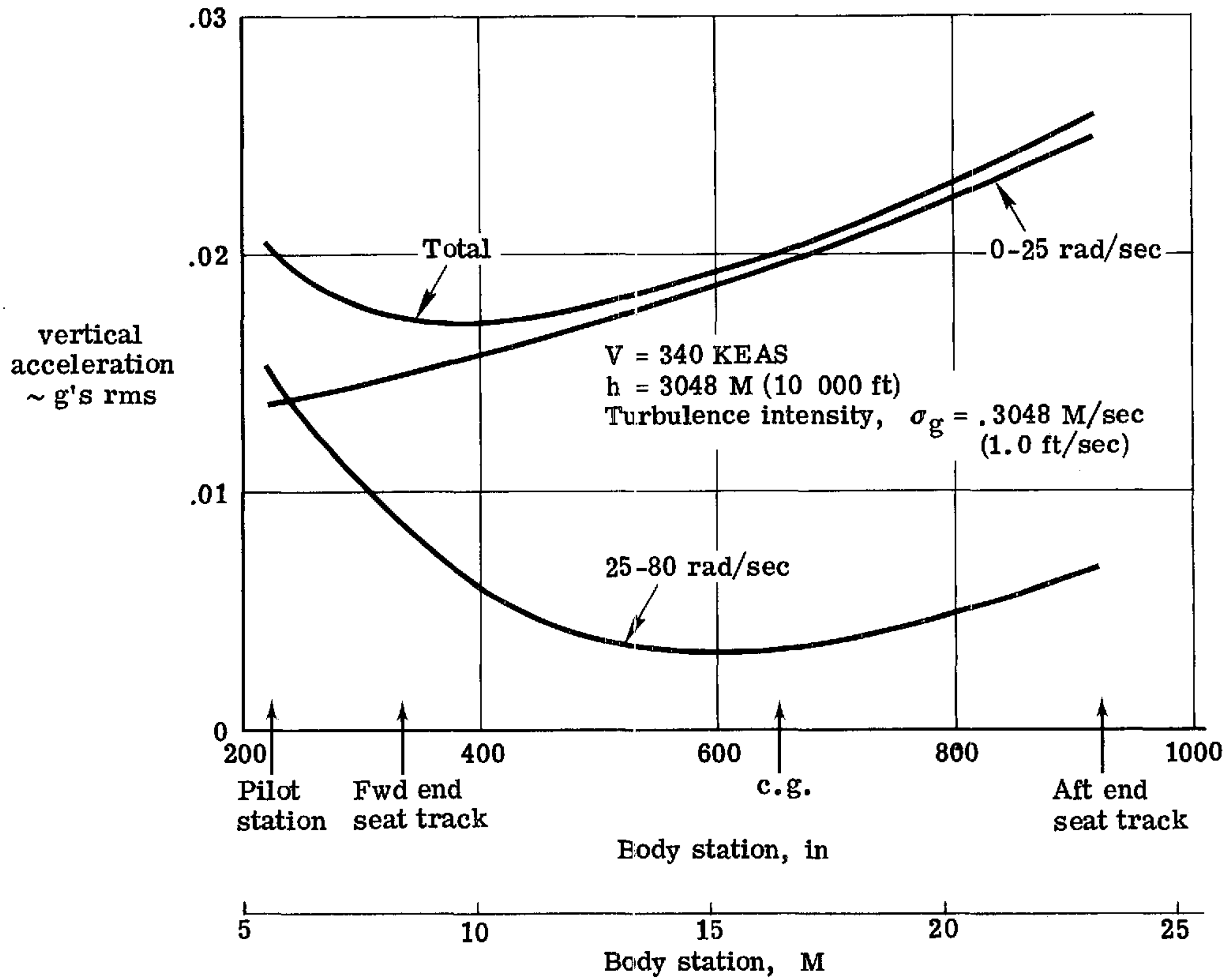



**Figure 4: Baseline airplane random vertical acceleration response**

where the contributions are about equal. Notice that the rms between 0-25 rad/sec (rigid body) is almost equal to the total, farther aft. At any frequency, $\sigma_{tot} = \sqrt{\sigma^2_{(0-25)} + \sigma^2_{(25-80)}}$.
A ride control system via structural mode control would have significant payoff only at the pilot's station.

A similar graph is presented for lateral accelerations in Figure 5. Data is shown in Figure 5 with and without the yaw damper, which is a part of the production 737 airplane. Again, structural mode suppression would not improve ride quality for the passengers, but would for the pilot. The existing yaw damper does reduce the aft body lateral accelerations quite well, but does not control the structural modes.

Figure 6 is a graph of the PSD and cumulative rms of the random vertical bending moment response at approximately the aft body cantilever point, BS 17.02 M (670 in). There is strong contribution to bending moment from the structural modes at this station, suggesting an appreciable payoff for an aft body fatigue reduction stability augmentation system (SAS). An exceptionally high-response elevator actuator would be required, with a bandpass of approximately sixty rad/sec.

The rms random lateral bending moment responses along the aft body are shown in Figure 7, based on straight line approximation between computations for BS 17.02 M (670 in) and BS 22.68 M (893 in). The rms bending moment for rigid body and structural mode frequency bands are presented. The results are quite similar to those for lateral accelerations along the aft body, shown previously. The greatest part of the free airplane total rms results from rigid body dynamics. The existing yaw damper does an adequate job of reducing the rigid body contribution, but does not affect the structural modes.

Figure 8 shows PSD/rms data for the random vertical bending moment at the wing root (approximate wing/body intersection). The rms component resulting from short period dynamics is dominant. However, there is sufficient contribution from a structural mode at approximately 22 rad/sec (3.5 Hz) to validate the digital implementation of a wing GLA system and demonstrate predicted performance.

## 4.5 ACT System Synthesis and Performance Analysis

Preliminary synthesis and performance analyses were performed for the various concepts to contribute to the decisions regarding flight validation feasibility, and to provide a prediction of performance that can be expected. Even better performance is likely from a more complete synthesis study phase of a flight validation program.

4.5.1 Ride control/direct lift. - A ride control (RC) system using direct lift control surfaces near the c.g. was synthesized. The direct lift surface ideally produce purely translational vertical acceleration of

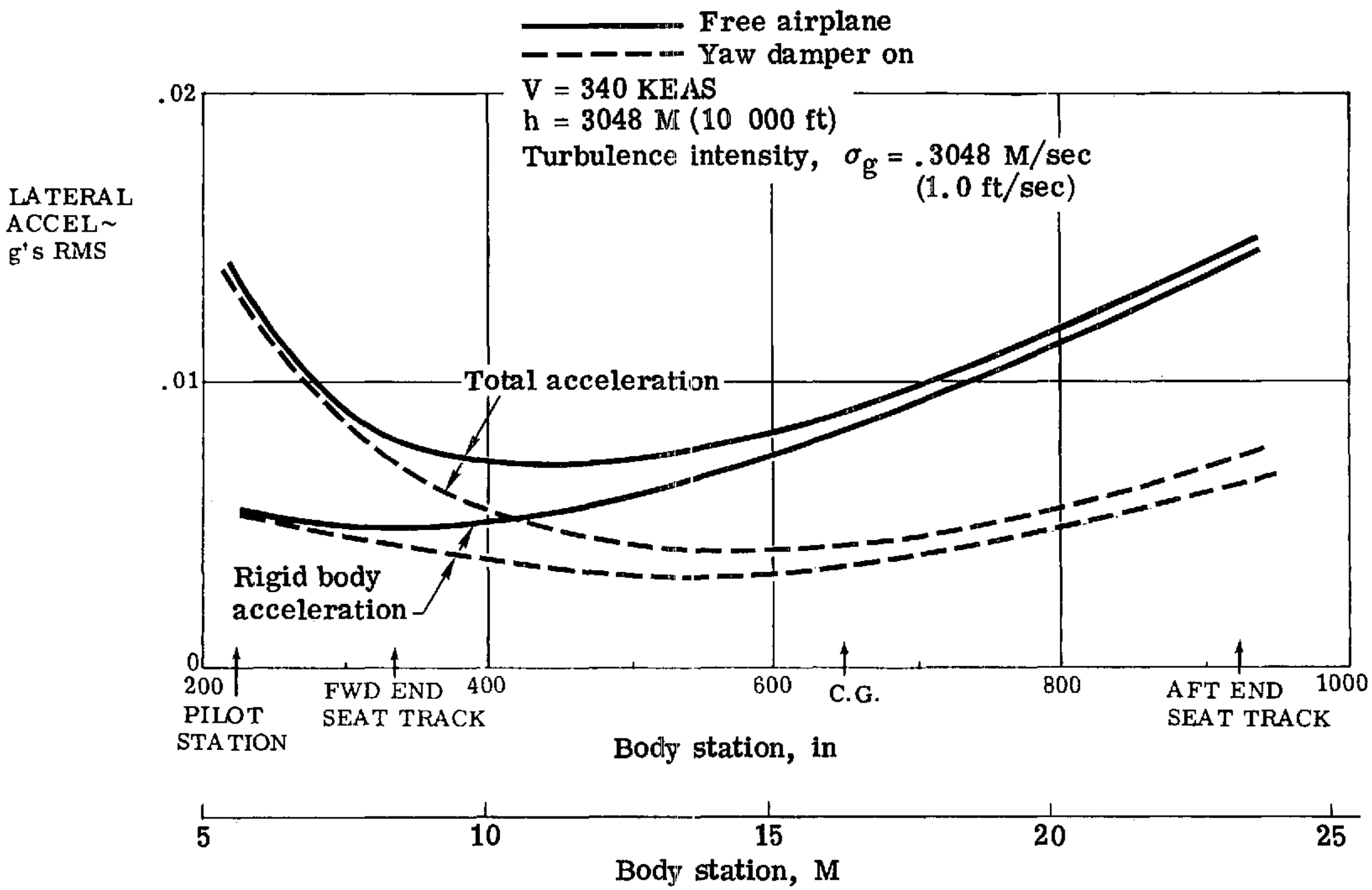



**Figure 5: Baseline airplane random lateral accelerations**

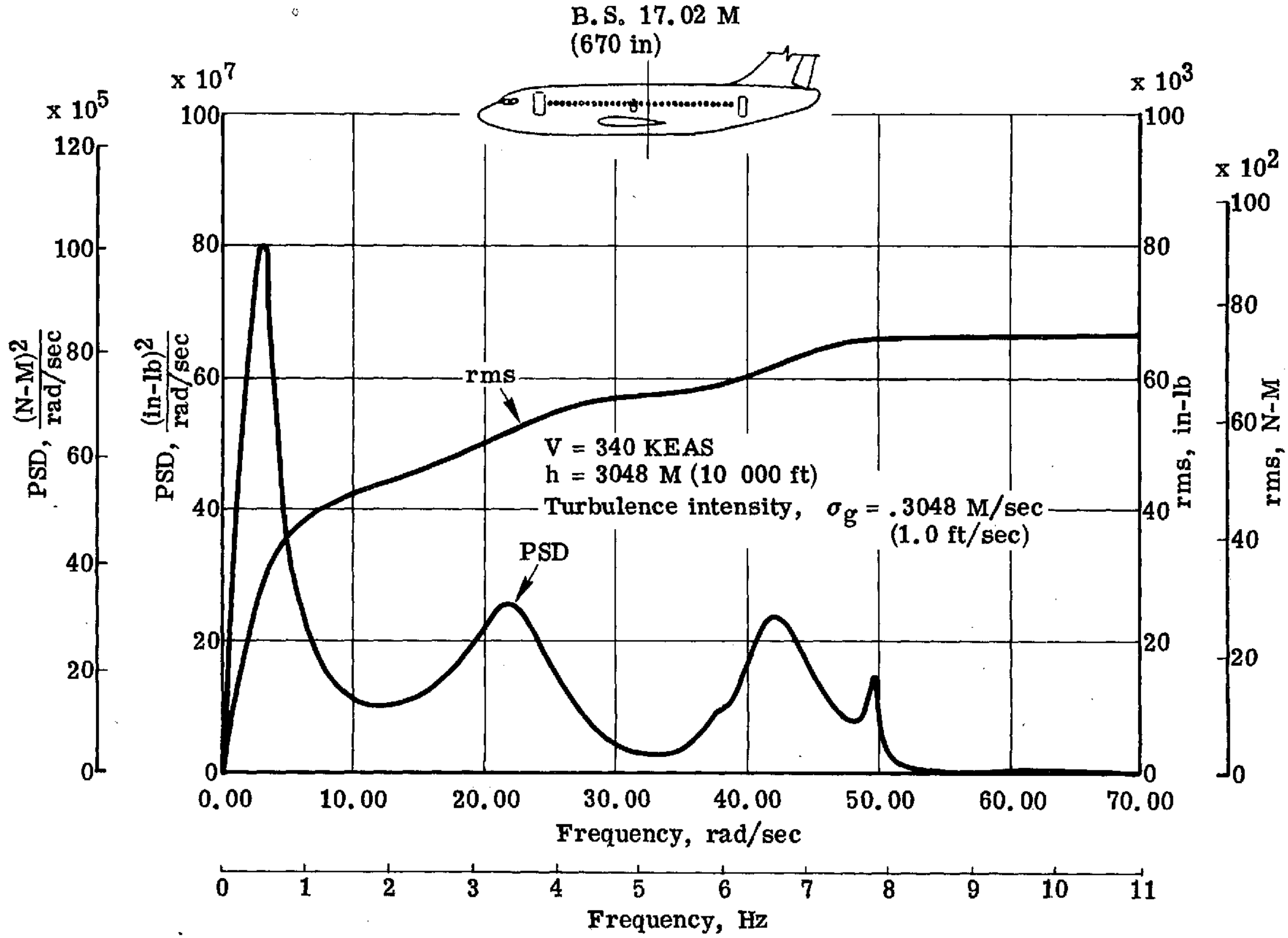



Figure 6: Mid-body random vertical bending moment response, baseline airplane

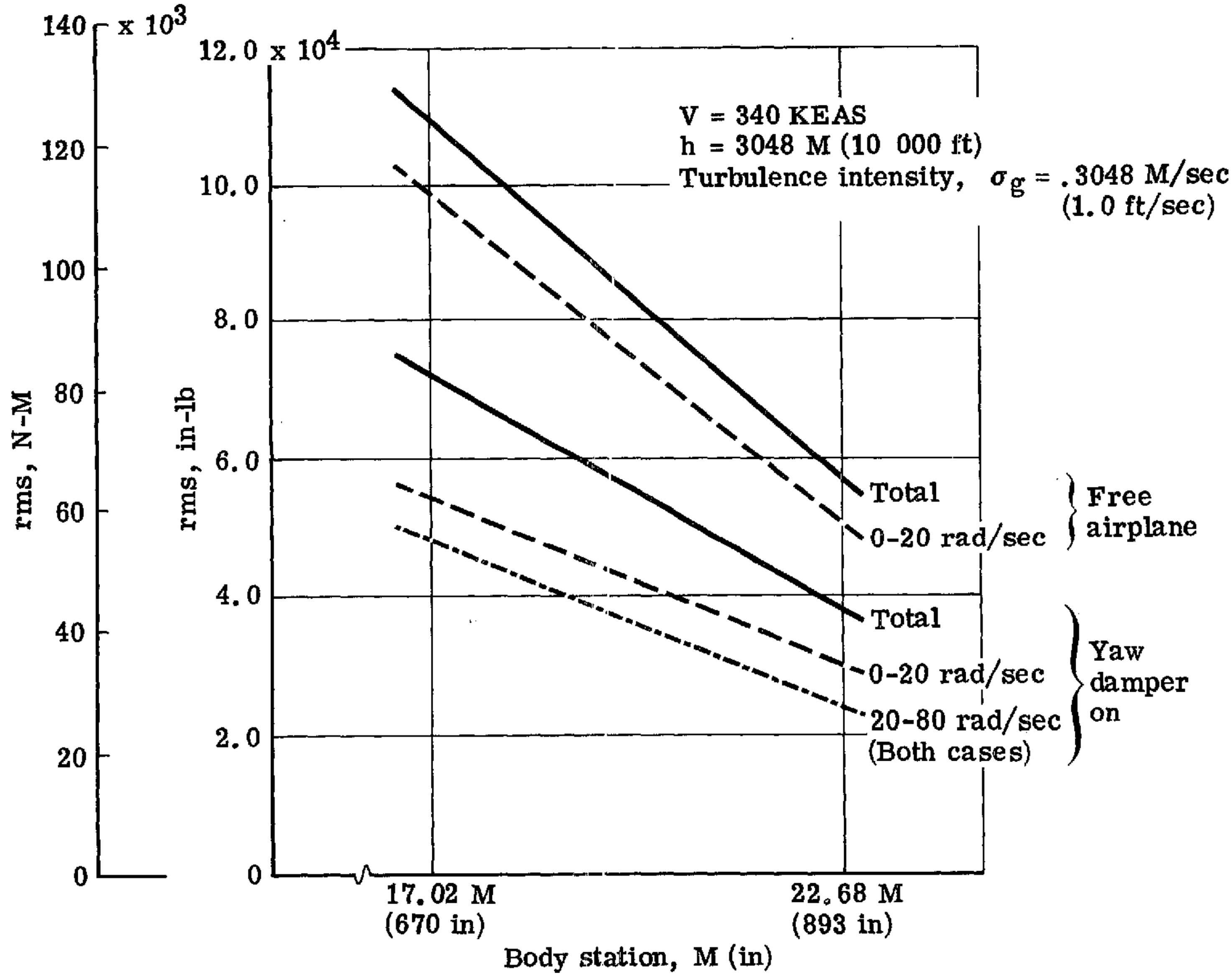



Figure 7: Aft body random lateral bending moment response, baseline airplane

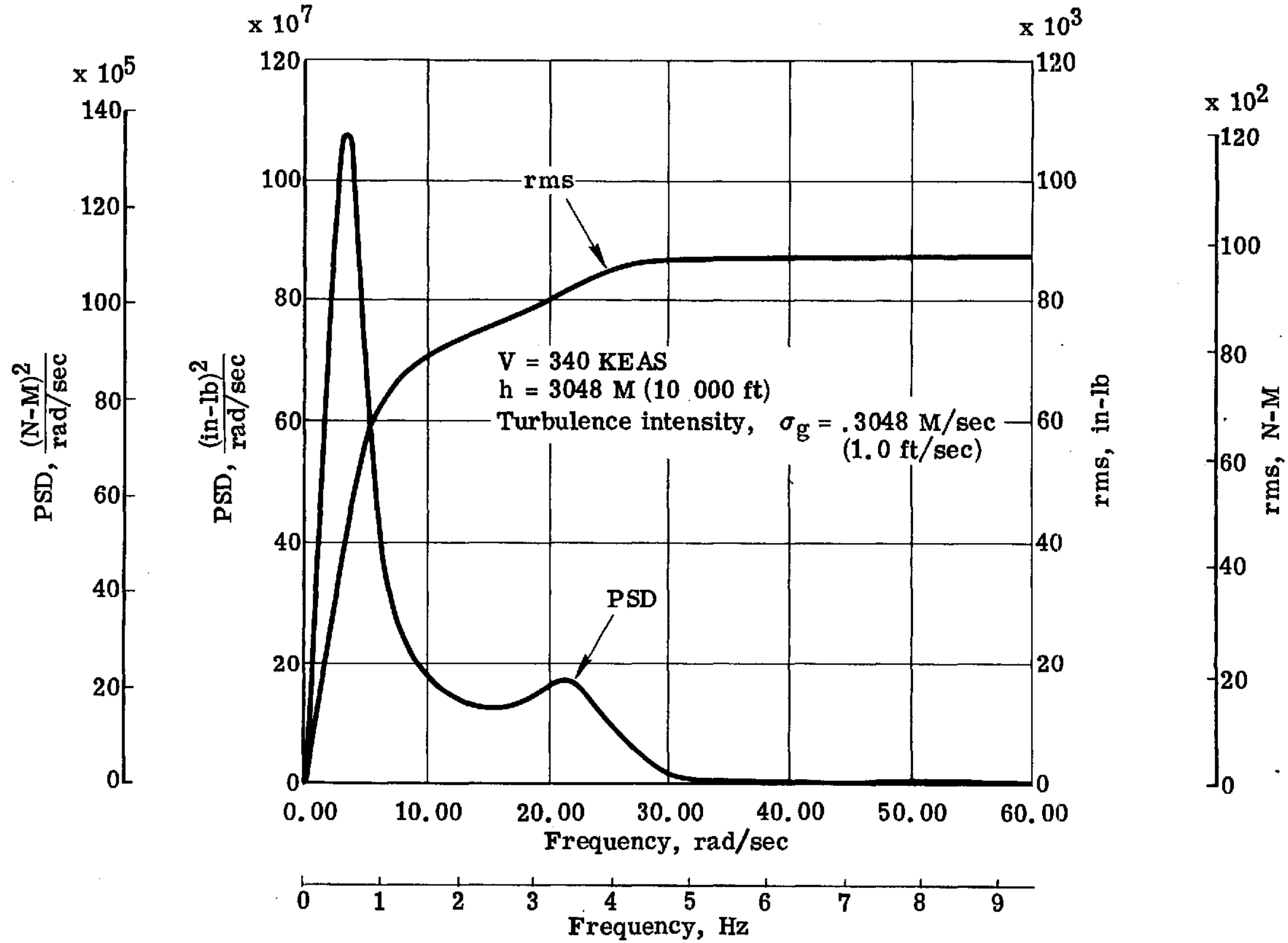



Figure 8: Wing root random vertical bending moment, baseline airplane

the c.g. The purpose of such a RC system is to reduce short period translational response, reducing vertical acceleration all along the fuselage.

The block diagram in Figure 9 shows the RC system synthesized utilizing biased existing flight spoilers, operated symmetrically. Vertical acceleration at the c.g. is used for feedback to the spoilers. A pitch damper (pitch rate to elevator) was added to reduce rotational accelerations, including pitching motions induced by the spoilers.

Reductions in rms accelerations accomplished by the RCS are shown in Figure 10. The reductions range from 30 percent at the forward passenger station to 39 percent at the aft passenger station. The system was designed to operate without performance degradation at turbulence intensities up to 2.1 M/sec (7.0 ft/sec) rms, which has less than a .01 probability of exceedance in the climb condition.

Since direct lift surfaces would be available for ride control, direct lift control (DLC) for pilot maneuvers could be implemented at little additional cost. The DLC would provide improved glide path capture and precision glide path tracking and landing flare. The DLC surfaces would also have considerable potential for implementing TCV control laws. The block diagram (Figure 9) includes a concept for DLC, in which the pilot command signals go directly to the symmetrical spoilers, with a crossfeed to the elevators to balance out spoiler-induced pitching moments. A low-gain integration of the column signal combined with a pitch attitude feedback to the elevator accomplishes a trim function.

4.5.2 Gust load alleviation (wing root). - A gust load alleviation (GLA) system was synthesized to reduce vertical bending moment response to random turbulence at the wing root (approximate wing/body intersection). The block diagram of the system is shown in Figure 11. Rate gyros are used to sense local roll rates at left and right Wing Buttock Lines 10.03 M (395 in). The difference is taken between left and right wing roll rates to cancel out the rigid body roll component, leaving essentially wing vertical bending rate. The phasing requirements are sharply different between the short period frequency and the first structural mode in the response. Two filters were used to satisfy the requirements of the two modes. The feedback gains shown are for full existing ailerons.

Figure 12 shows the comparison between the vertical bending moment PSD/rms of the free airplane and with the GLA system on. A goal of 20 percent reduction in rms had been set, and 22 percent reduction was achieved. This was sufficient to show the feasibility of demonstrating the system to validate digital implementation and predicted performance. It is felt that an additional increment of rms reduction could be achieved with a pitch damper, which is included in general terms in the block diagram.

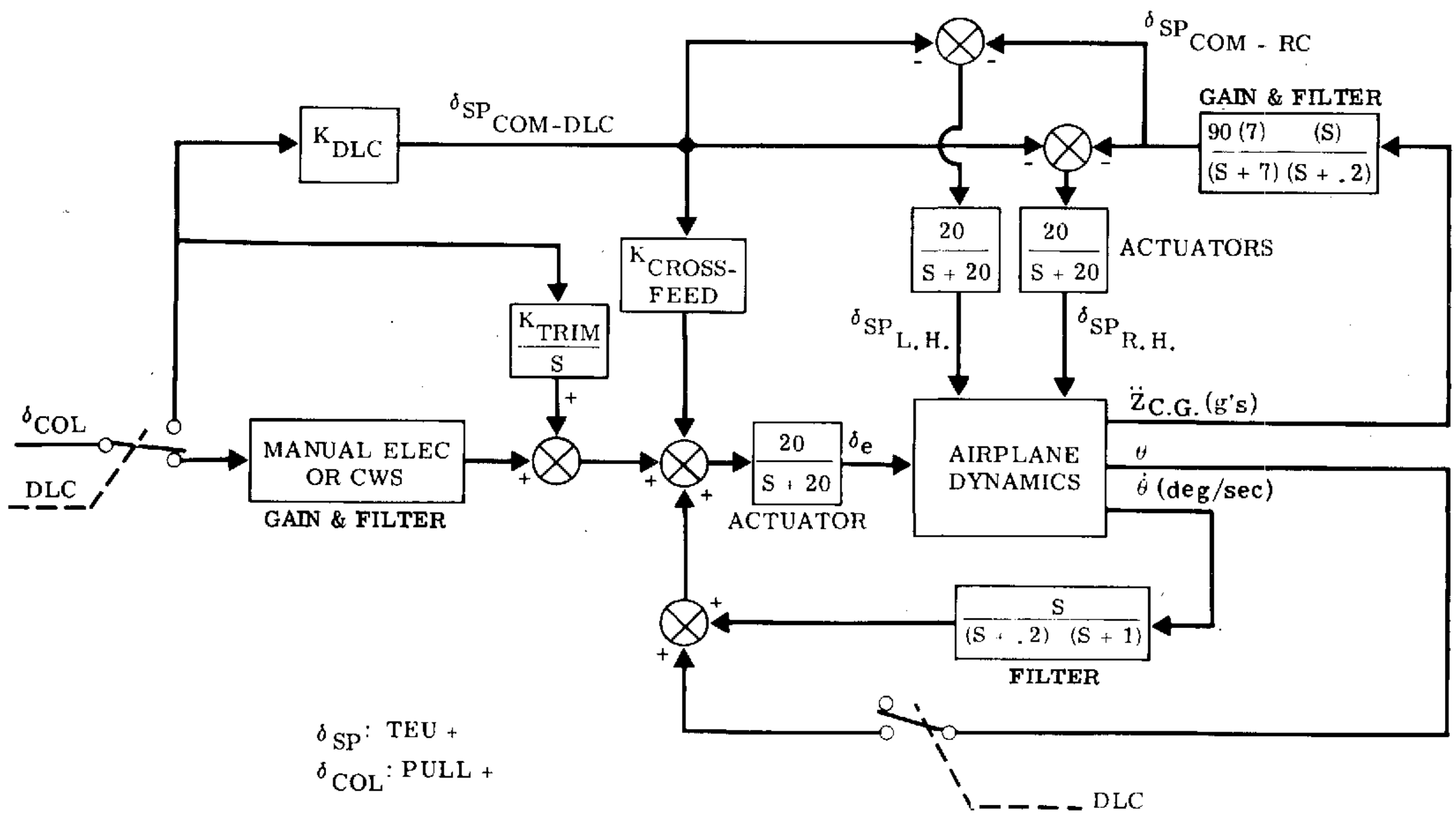



**Figure 9: Ride control/direct lift control system concept**

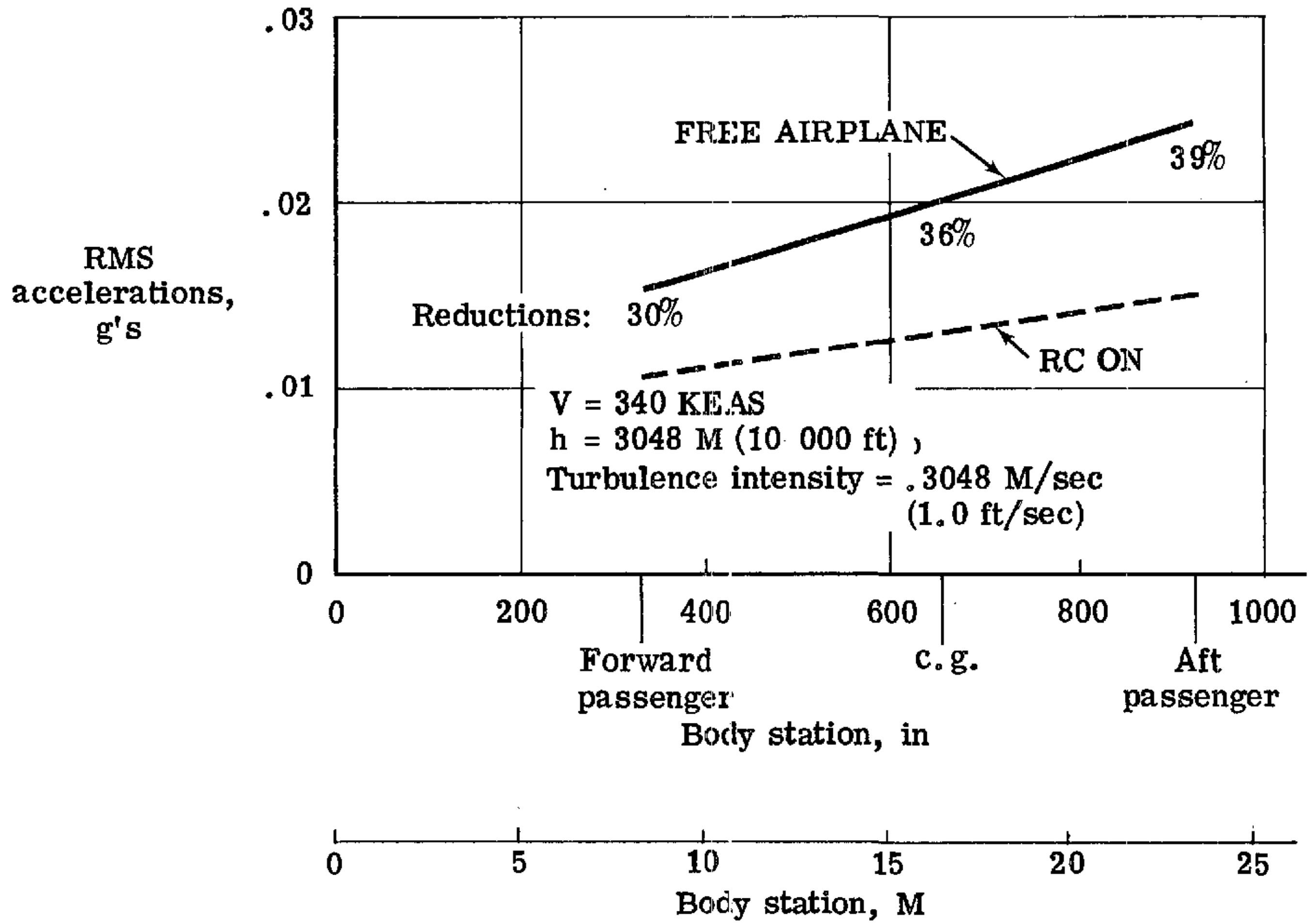



Figure 10: Effect of ride control (via direct lift) on rms vertical acceleration

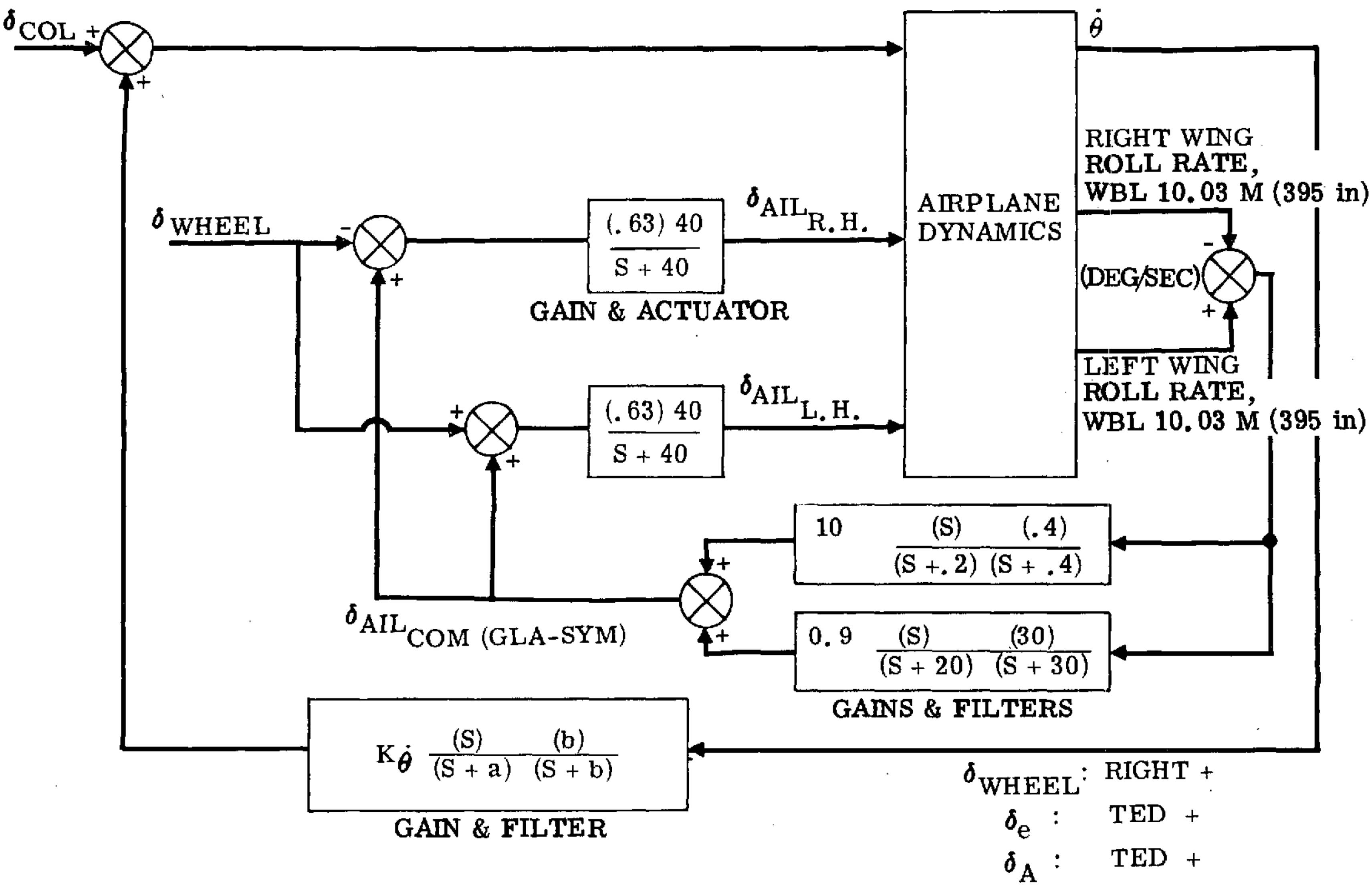



Figure 11: Wing gust load alleviation system concept

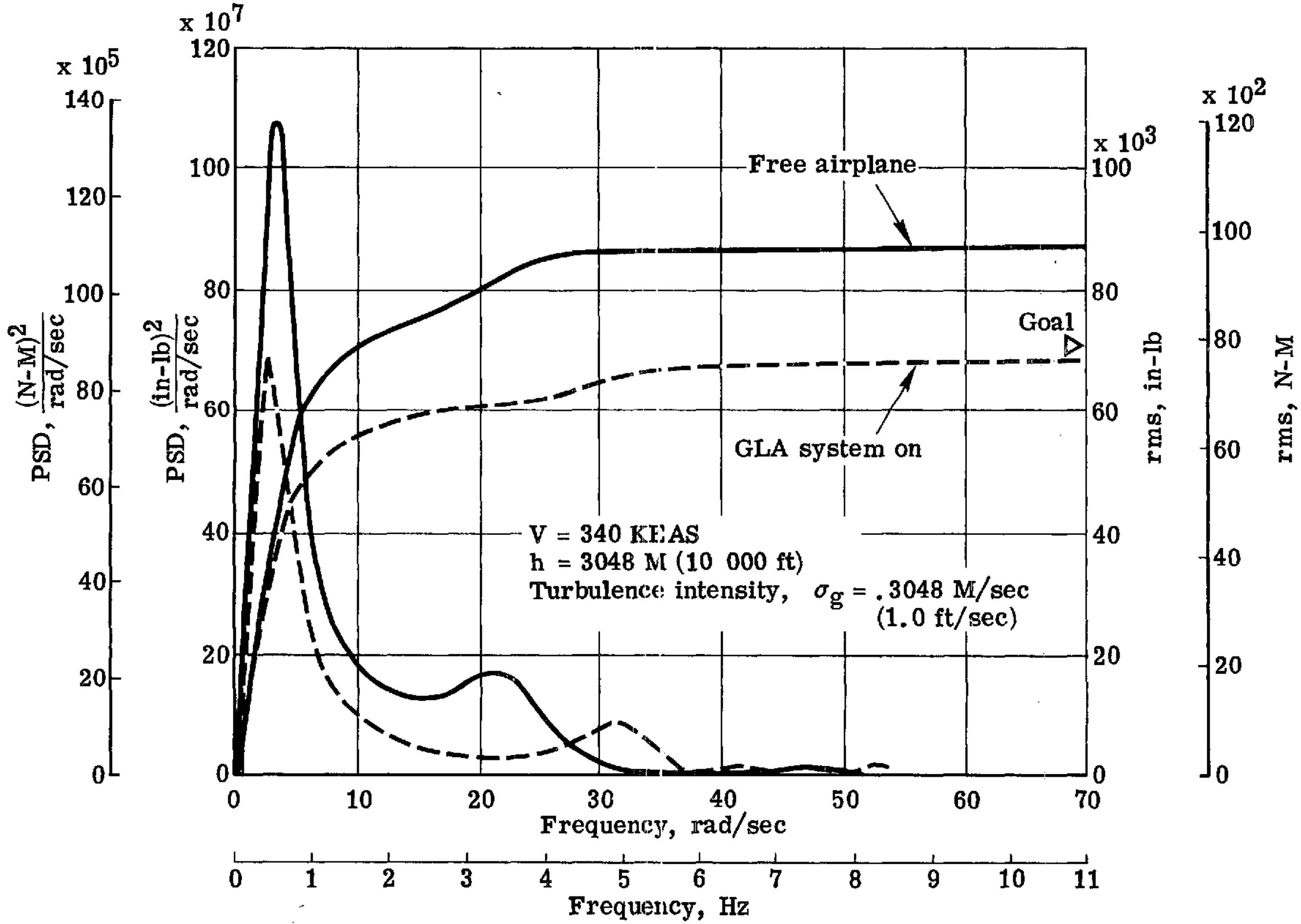



**Figure 12: Effect of gust load alleviation system on wing root random vertical bending moment response**

The effect of the GLA system on rms vertical acceleration response was computed to assure that it does not degrade the ride quality. The computed data is shown in Figure 13 over the length of the fuselage. Ride quality is not significantly affected, and for the most part, is slightly improved by the GLA system.

The system was designed for undegraded performance at turbulence intensities up to 2.1 M/sec (7.0 ft/sec) rms, which has less than a .01 probability of exceedance in the climb condition.

Because of the simplicity in synthesis approach, gyros were used to develop the GLA concept. However, accelerometers are more reliable and should be investigated in the analysis phase of a flight validation program.

4.5.3 Relaxed static stability/c.g. control. - The effect of moving the c.g. aft, in terms of the phugoid mode root locus is illustrated in Figure 14. As the c.g. is moved aft on the free airplane, the roots move down to the real axis and separate, one going unstable, as illustrated in the left-hand pannel. Static neutral stability occurs at 31 percent M.A.C.

In the center panel of Figure 14, the c.g. is held at 42 percent M.A.C. and the production Mach trim system feedback (Mach number to elevator) is closed and the gain is increased from zero to the nominal Mach trim system gain. The root is restabilized, and becomes oscillatory, and the oscillatory roots are neutrally stable at the nominal gain. In other words, the airplane is neutrally stable at 31 percent M.A.C. with the Mach trim system off and at 42 percent with the Mach trim system on.

If the synthesized speed feedback is added around the existing Mach trim system, with the c.g. at 42 percent the low frequency oscillatory roots are again stabilized, as shown in the right-hand panel of Figure 14.

Figure 15 shows the effect of moving the c.g. aft in terms of the short period root locus, in the left-hand panel. The right-hand panel shows the variation of root locations for the same range of c.g. positions after the synthesized pitch rate feedback to the elevator is added. Essentially constant short period pitch response to column displacement would result for the full range of c.g. locations.

The velocity and pitch rate feedbacks synthesized are shown in the block diagram of Figure 16. A command augmentation filter is also indicated, which would achieve required column forces for maneuvers.

In addition to studying the handling qualities problems of a relaxed stability airplane, it would be desirable to empirically determine the fuel savings (or range improvement) achievable on the 737 from aft c.g. locations by flight test. Figure 17 illustrates a typical graph of fuel savings as a function of c.g. position estimated for the B-52 airplane. The automatic CGC used to set up RSS conditions could be used to measure this kind of fuel savings data for the 737,

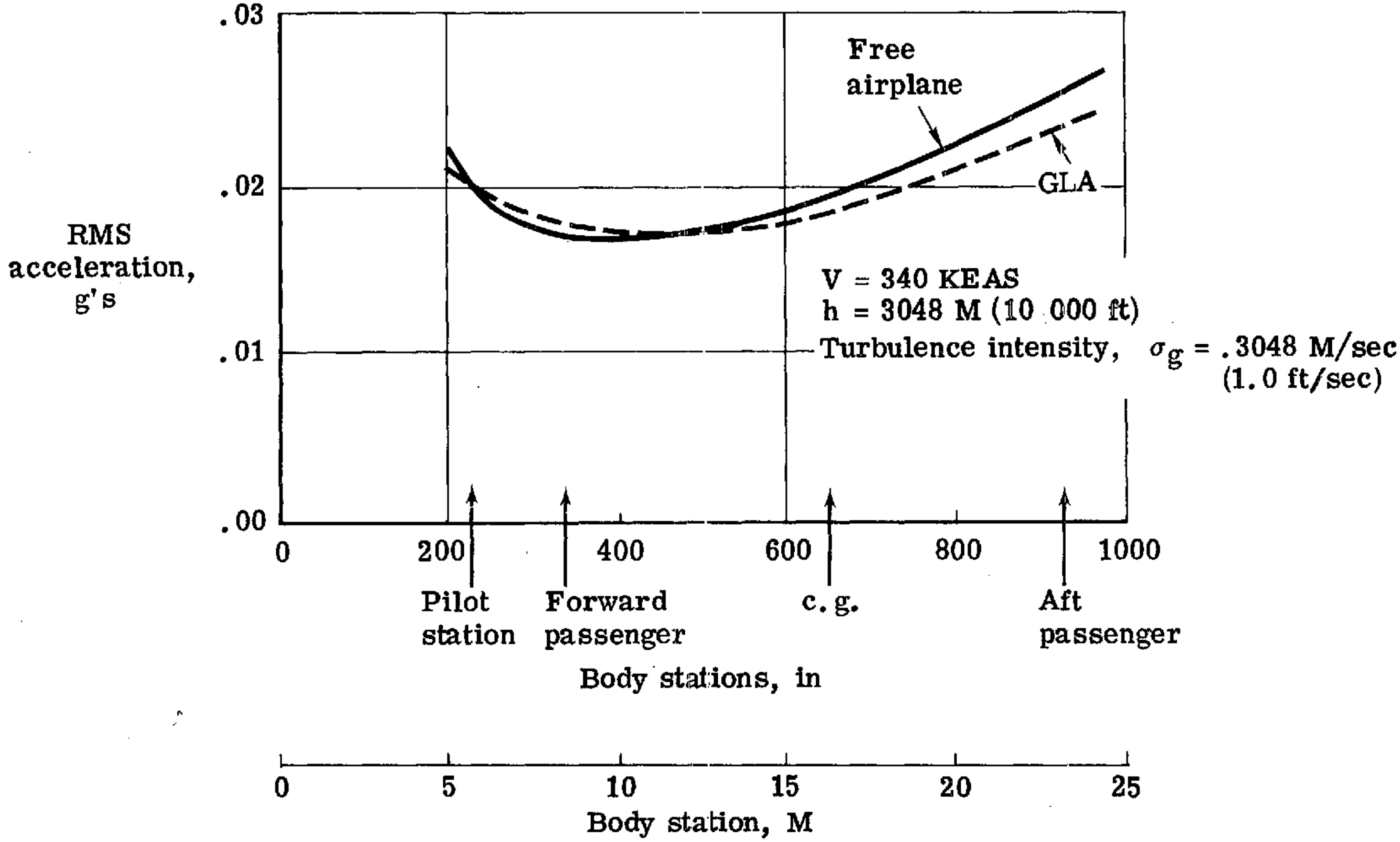



Figure 13: Effect of gust load alleviation system on rms vertical accelerations

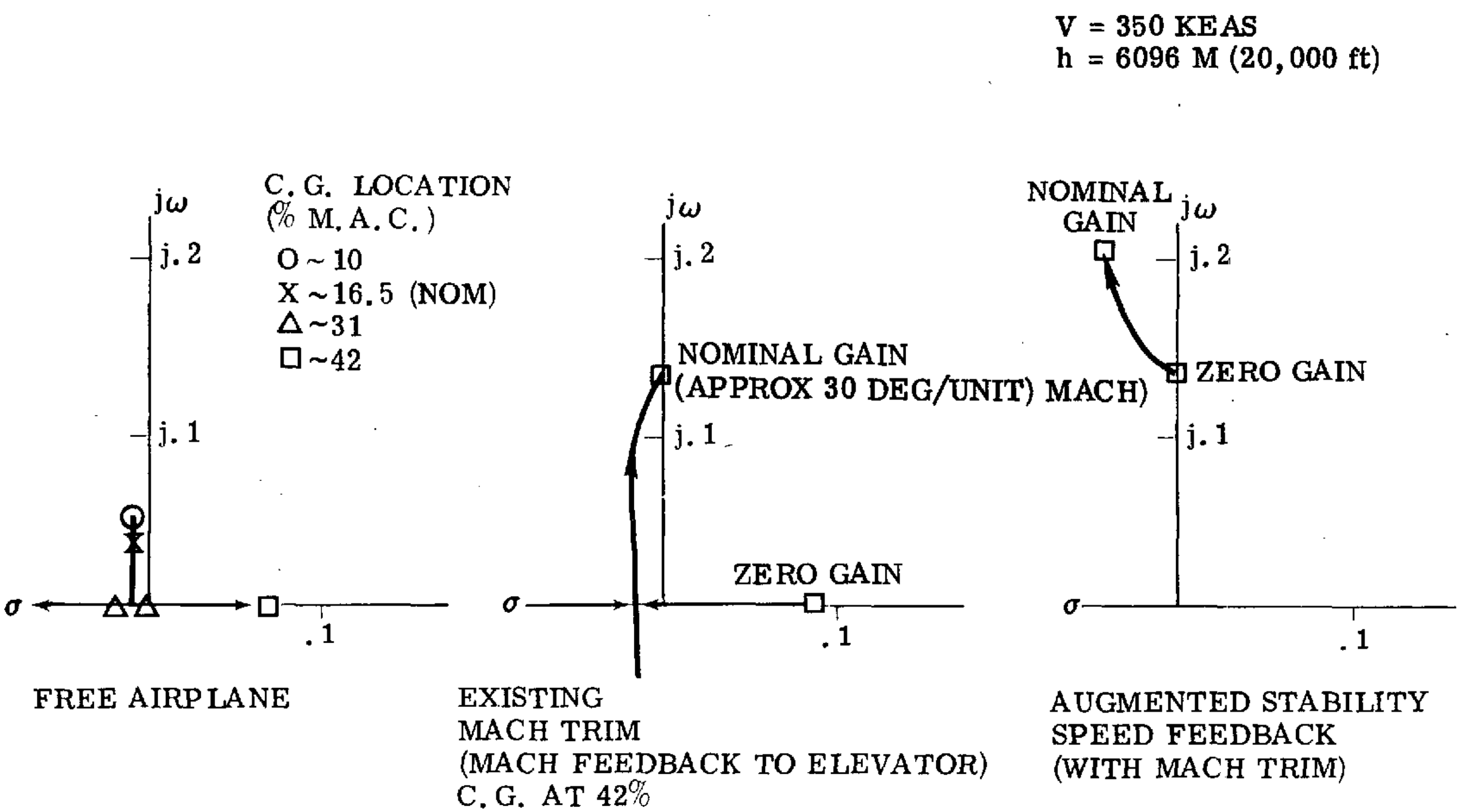



**Figure 14:** Effect of pitch stability augmentation system on phugoid roots

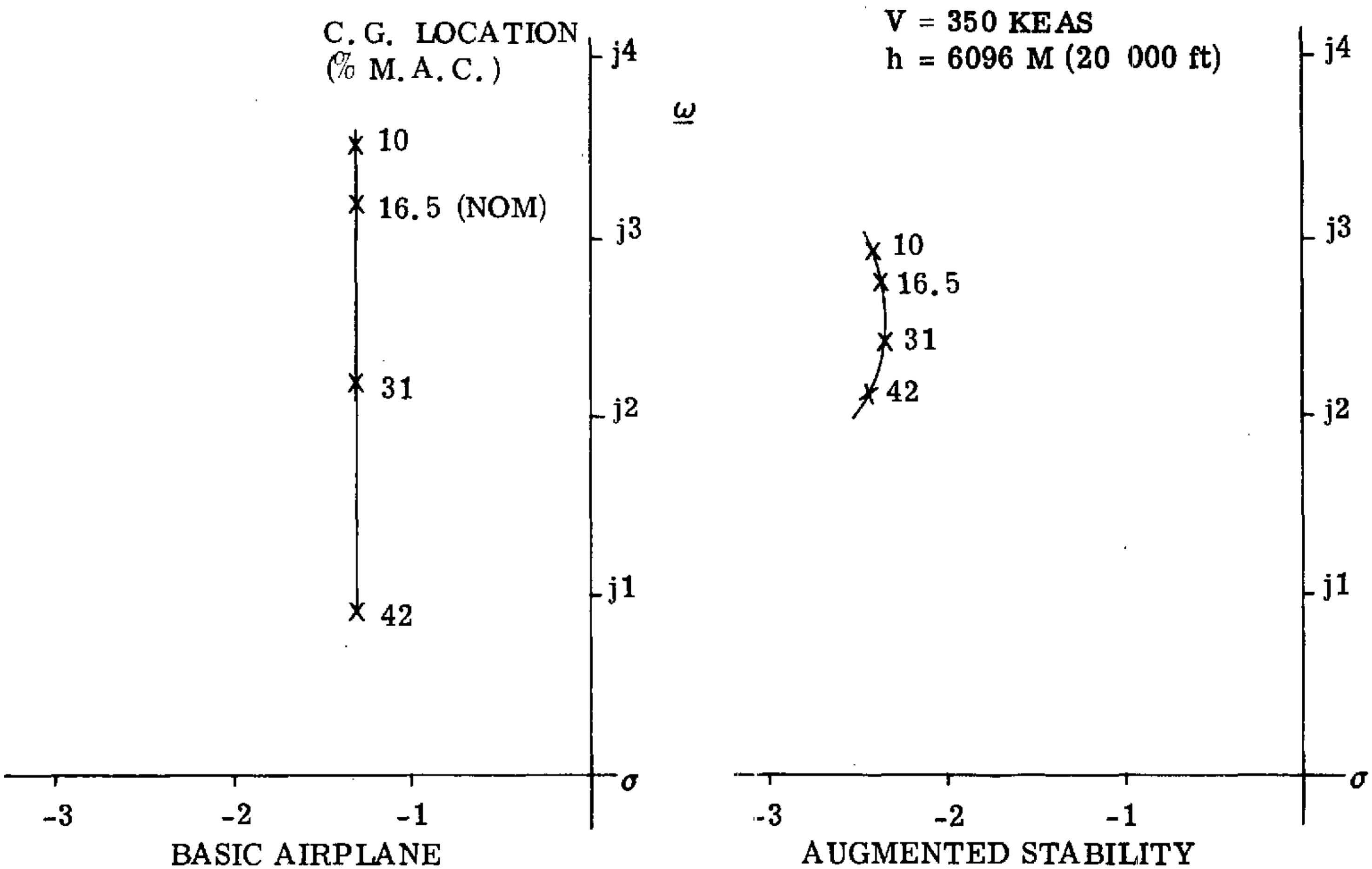



**Figure 15: Effect of pitch stability augmentation system on short period roots**

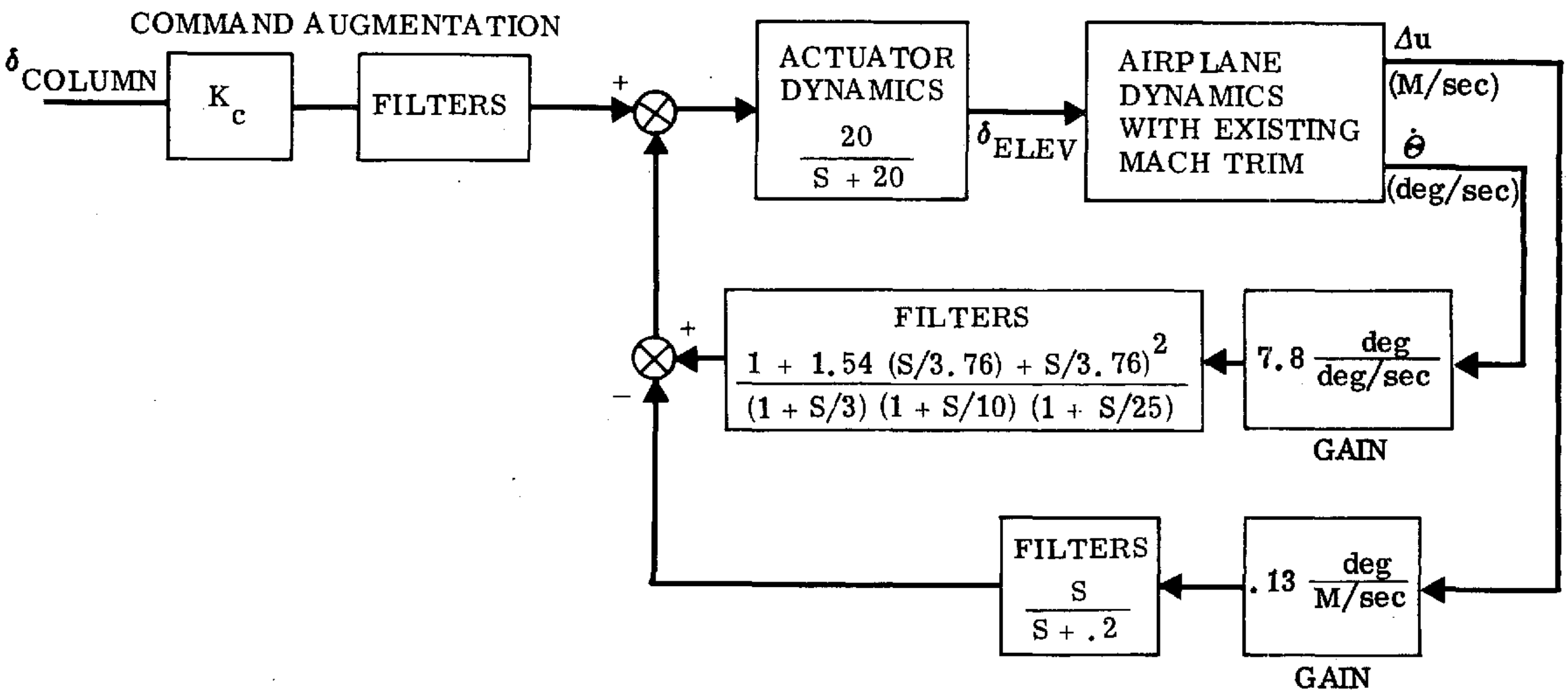



**Figure 16: Pitch stability augmentation system**

NOTE: B-737 – 0.5% RANGE IMPROVEMENT FOR
4% AFT SHIFT IN C.G

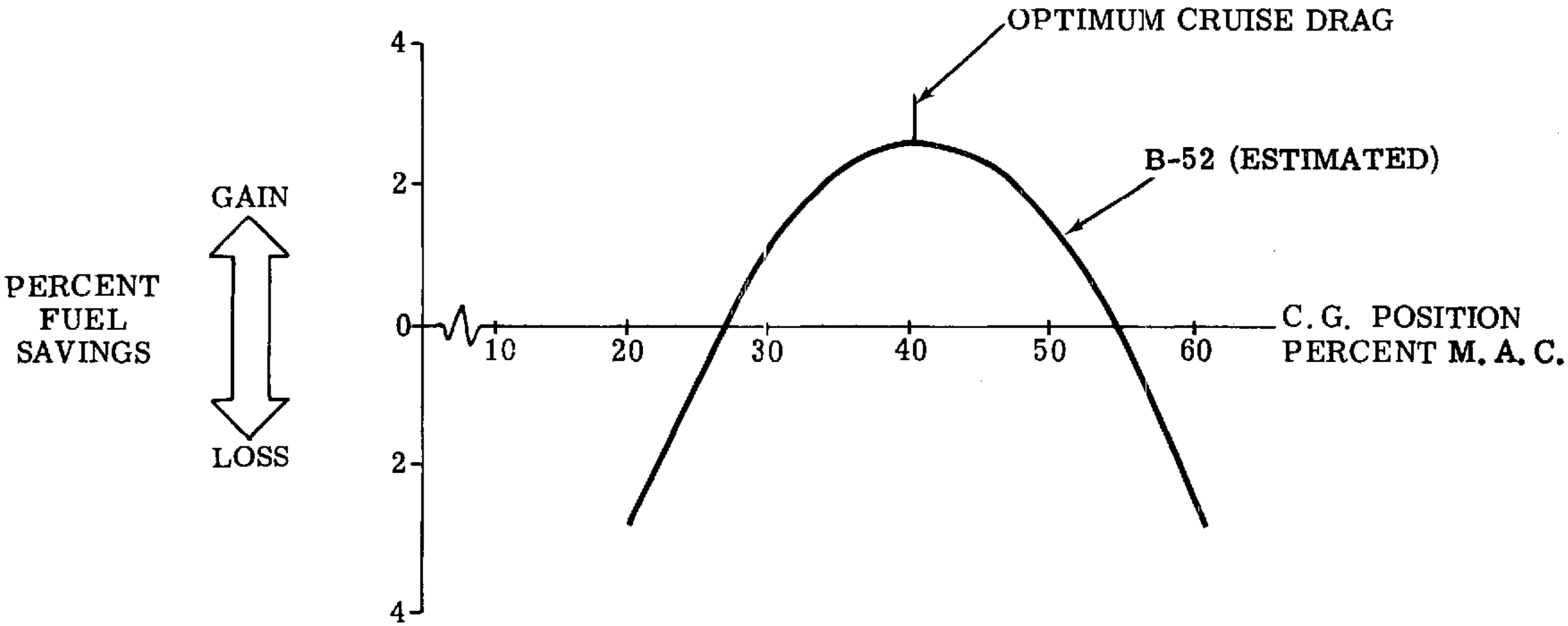



**Figure 17: Typical effect of c.g. location on fuel savings**

with c.g. locations run back to back under consistent flight and test equipment conditions. Boeing has advised their airline customers via the Boeing Airliner Magazine[5] that analysis shows a 0.5 percent fuel savings per 4 percent M.A.C. aft shift in c.g. These tests would validate this slope, determine where it crosses the axis, and at what c.g. location the peak savings occur.

4.5.4 Ride control via structural mode suppression. - Figures 4 and 5, paragraph 4.4, showed that structural flexibility contributes significantly to fuselage accelerations only at the very forward end. A forward body modal suppression ride control system was synthesized with a pair of horizontal canards added at BS 5.69 M (224 in) and WL 5.08 M (200 in). The system is shown in the block diagram of Figure 18. The feedback sensor is a vertical accelerometer at the pilot's station. A bandpass filter is used in the region of the structural mode frequencies.

The system should have practically no effect on handling characteristics with a high-pass filter (washout) in the feedback at 30 rad/sec. The PSD/rms computations of pilot station vertical acceleration responses to random turbulence, with and without the RC system, are shown in Figure 19. The acceleration response of the structural modes is almost totally suppressed by the RC system. In addition, the RC system reduces the rigid body response somewhat, probably in the rotational acceleration. The rms of total vertical acceleration at the pilot's station is reduced by 43 percent. Figure 20 shows the effect of the modal suppression RC system on random vertical accelerations along the rest of the fuselage. The rms is reduced at all stations, ranging from 3 to 43 percent. The system was designed to operate without performance degradation at turbulence intensities up to 2.1 M/sec (7.0 ft/sec) rms, which has less than a .01 probability of exceedance in the climb condition.

A pitch damper might provide an additional increment of vertical acceleration reduction, and is shown in general terms in the block diagram (Figure 18). The pitch damper is not included in the analysis results.

4.5.5 Maneuver load alleviation. - The effectiveness of the existing ailerons and added flaperons for maneuver load alleviation (MLA) is illustrated in Figure 21. The flaperons used in the analysis were the same span as the existing inboard flaps and had a chord varying from .50 M (19.5 in) to .76 M (30 in). The graph shows wing root bending moment reduction, in terms of percent of design limit load, as a function of symmetrical aileron deflection. The reductions shown are relative to pullups with elevator control only. The ailerons dump lift at the wing tip and the flaperons increase the lift on the inboard portion. The elevators are deflected to maintain the specified constant maneuver.

Additional flaperons are not within the scope of the Phase I type

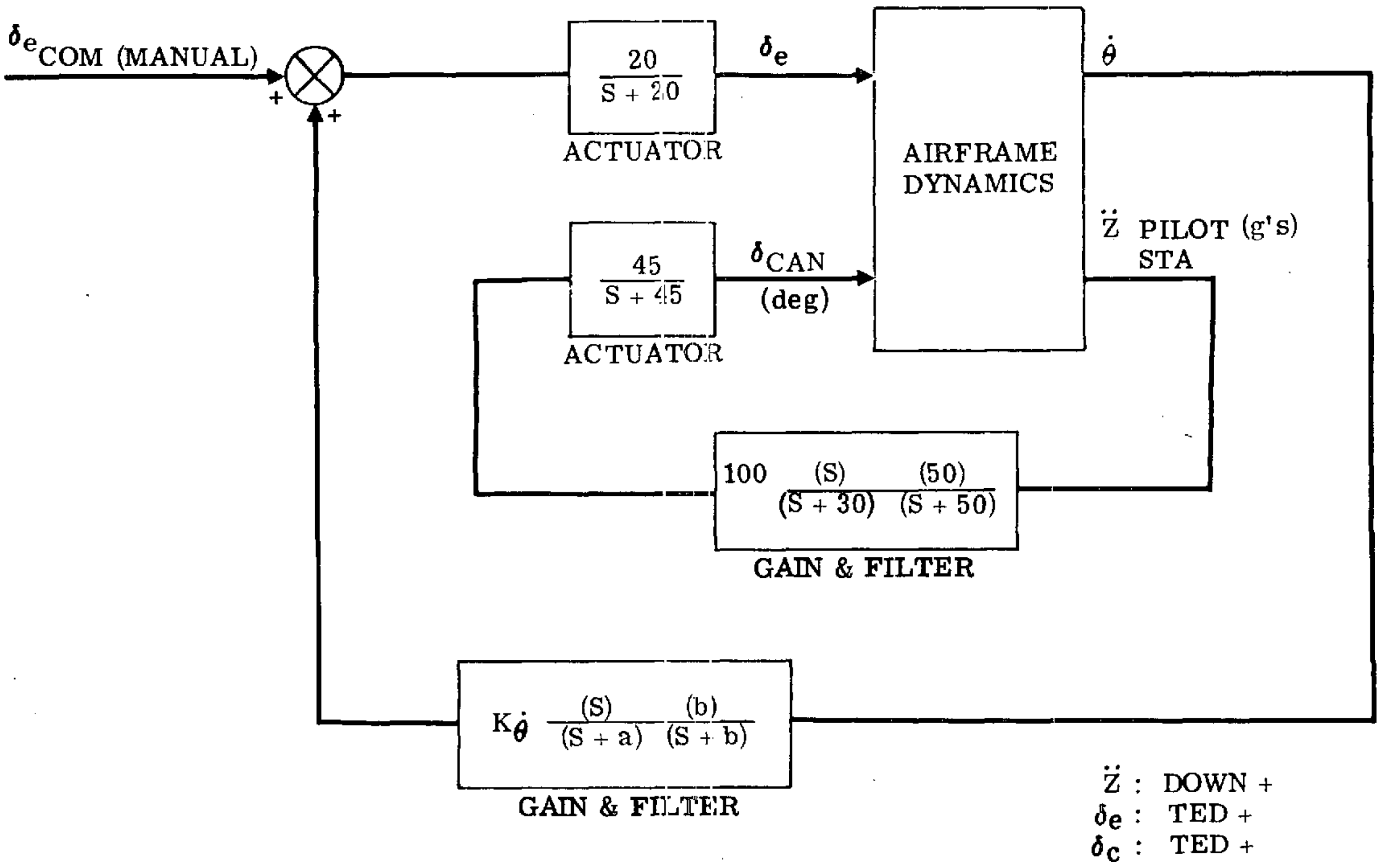



**Figure 18: Forward body mode ride control system concept**

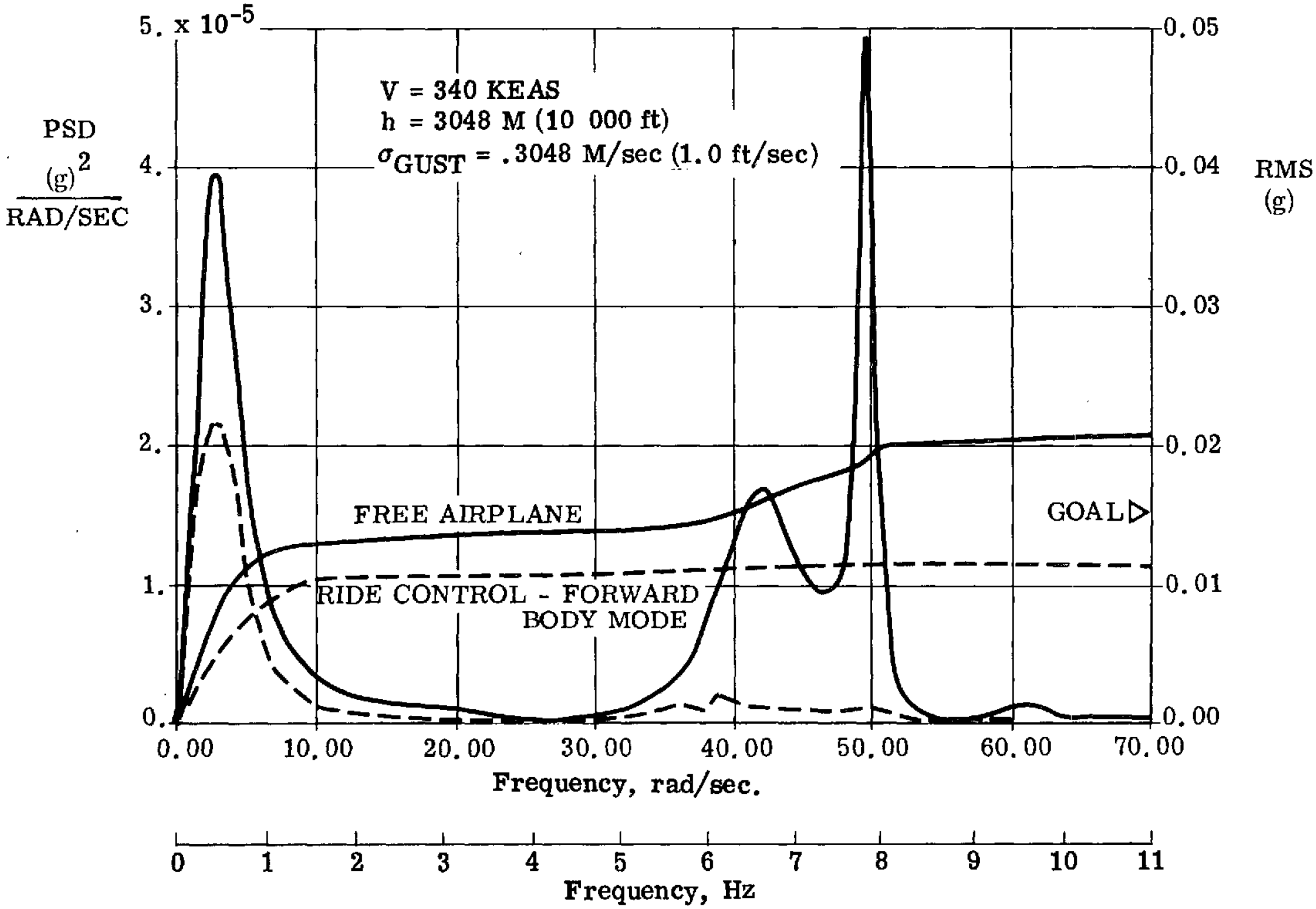



**Figure 19: Effect of forward body mode ride control on random pilot station vertical acceleration response**

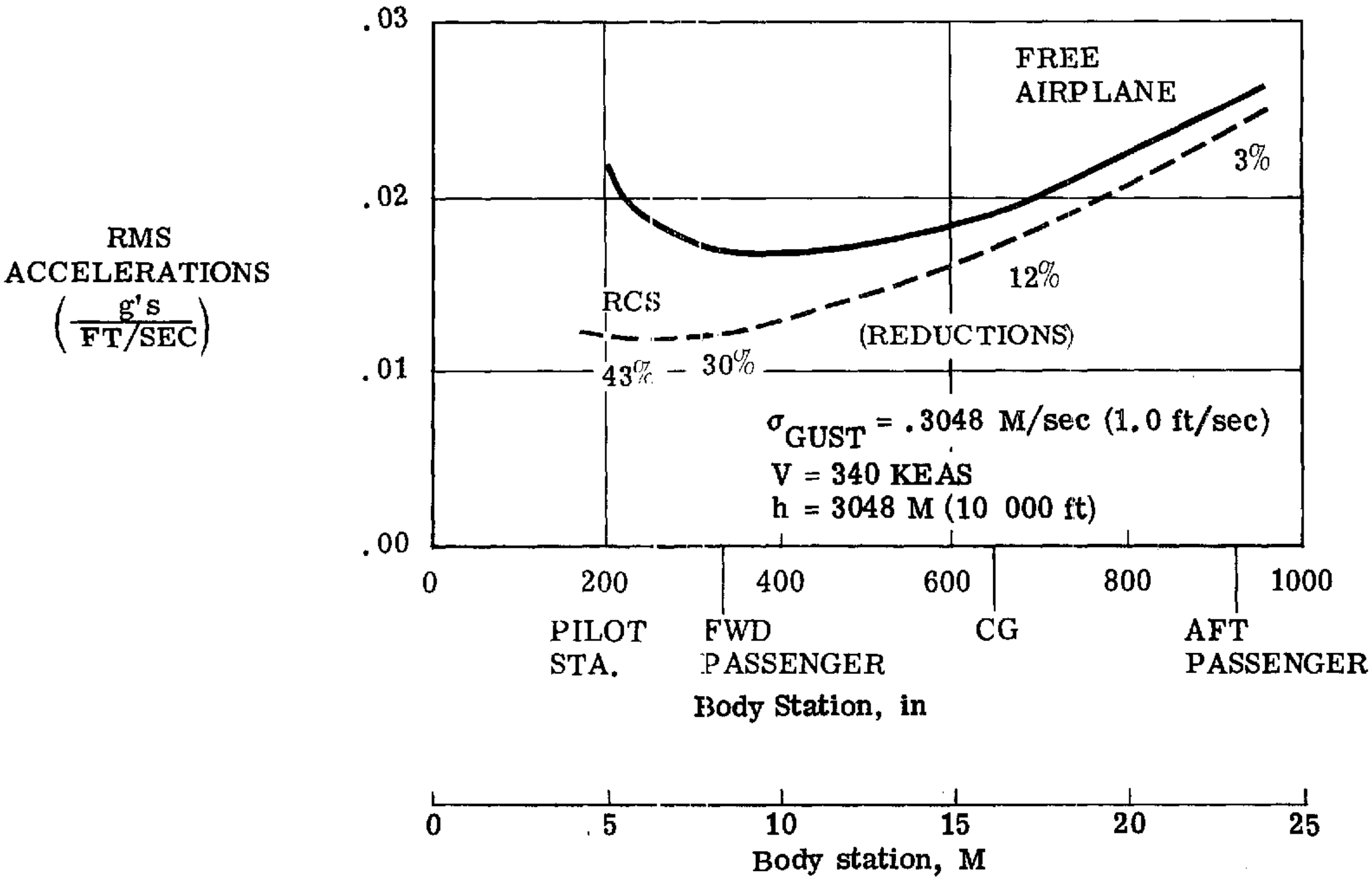
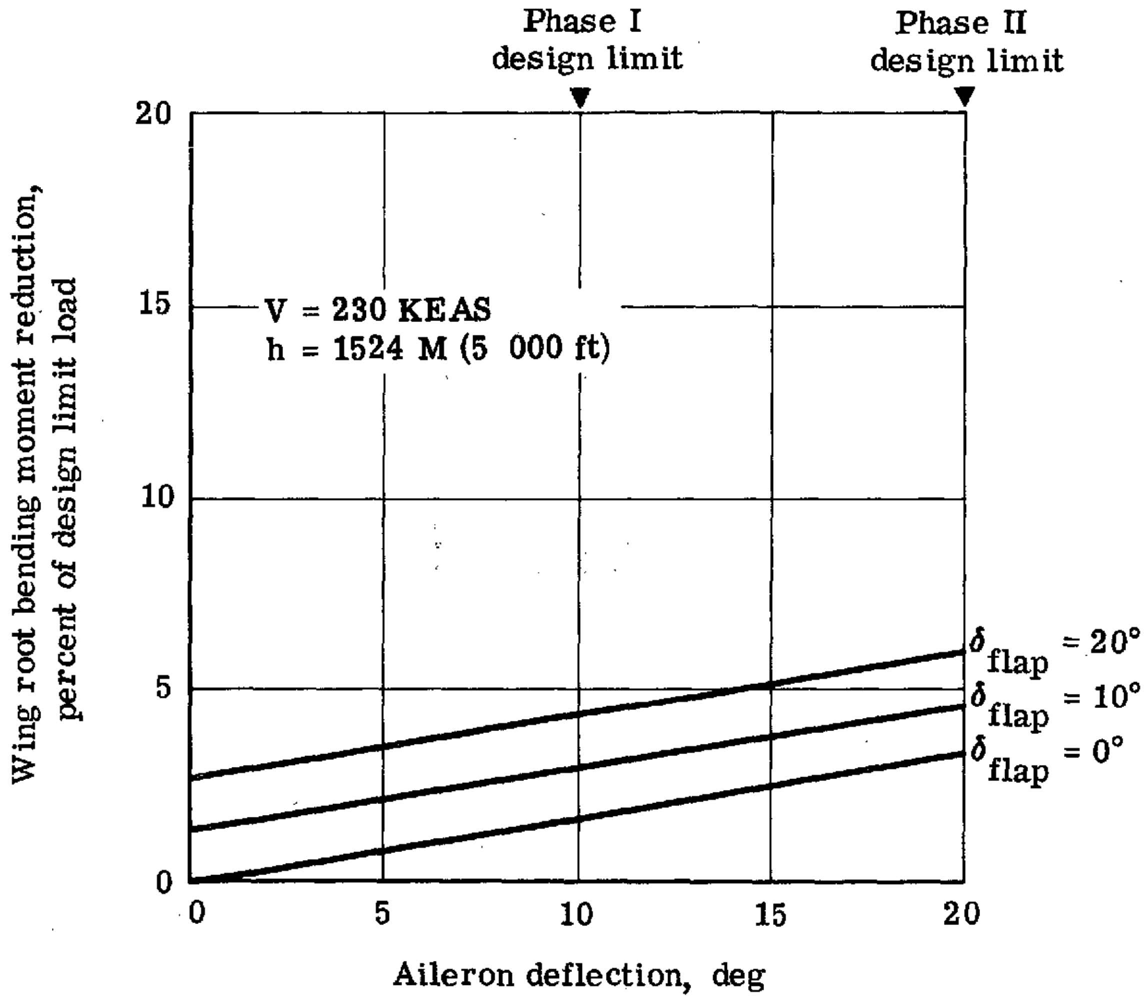



**Figure 20:** Effect of forward body mode ride control on rms vertical acceleration along fuselage




Note; Reductions are calculated relative to elevator-only maneuver

Figure 21: Effectiveness of flaperon and aileron for maneuver load control

programs (low-cost, minimal effect on the TCV program). Also in Phase I, the safety of ACT controls will depend largely upon limited control authority rather than on a high degree of redundancy. A maximum of ten degrees of aileron would be available for ACT in Phase I. Without flaperons ten degrees of symmetrical ailerons would accomplish a 1.65 percent reduction in the design limit load at the wing root. This would not be sufficient benefit to validate the concept. Even with twenty degrees of flaperon and twenty degrees of aileron the reduction is only 6 percent, a marginal payoff for the modifications involved.

4.5.6 Flutter suppression. - Prior analyses and flutter model testing accomplished by Boeing Commercial Airplane Company on the 737-100 airplane have shown that the basic airplane is flutter free within the airplane flight envelope. However, a symmetric flutter mode does exist in excess of 1.2 $V_D$, as noted in Documents D6-17751[6] and D6-17752[7], which summarize analytical results and flutter model/flight flutter test results, respectively.

The flutter mode frequency is shown to be approximately seven to eight Hz. Test results also indicate that the flutter mode is slightly sensitive to wing fuel loading and also to nacelle strut flexibility parameters. The flutter mode is comprised primarily of second wing vertical bending coupled with first wing torsion.

The mathematical representation of the RSFS airplane uses elastic axis lumped mass idealization with linear superposition of component cantilever modes. Unsteady aerodynamic influences are generated utilizing three-dimensional pressure distributions based upon a lattice of doublets in the plane of the primary lifting surfaces.

The basic RSFS symmetric aeroelastic idealization noted in the above paragraph was used to explore the feasibility of reconfiguring the basic RSFS to degrade the flutter boundary to within the flight envelope for the purpose of synthesizing and demonstrating an active flutter suppression system. The goal was to obtain a moderate flutter mode; i.e., one that destabilizes at a moderate rate as speed is increased.

The most feasible modification would be to add mass, thereby minimizing structural-elastic modification. In view of this, several mass variations were made with the following results (shown typically in Figure 22):

1. A wing tip mass forward of the elastic axis softens the flutter mode but increases the flutter velocity.

2. A wing tip mass aft of the elastic axis decreases the flutter speed to within the flight envelope (at the stated altitude), but causes the flutter mode to be more violent.

3. Nacelle cowl mass lowers the flutter speed somewhat and the

destabilization is moderate; however, the additional mass requirements may be excessive.

The term $f_{V_f}$ in Figure 22 is used for "frequency at flutter velocity." Within the flight envelop, the frequencies of flutter modes identified vary over the approximate range of 3.5 to 7.5 Hz.

Recapitulating, the results show a very mild mode beyond the design dive speed ($V_D$), an exceptionally violent mode well within the envelope, and a moderate mode very near $V_D$. It appears feasible, then, to identify a moderate mode within the flight envelope at a frequency of about six Hz. However, there is considerable risk in defining airplane modifications from this data because the mathematical model may be sensitive to structural/elastic parameters that are not representative of the actual system, particularly for item 3. Further analysis would be required including finite element modeling. Before a system is implemented, ground vibration tests would be required to verify analytical frequencies and mode shapes.

## 4.6 Analog Simulation

An analog computer simulation study was conducted to evaluate the compatibility of ACT control laws with TCV program control laws, and to assess the effects of actuator rate, position and bandpass limits on the Direct Lift Ride Control system performance. The QSE equations described in paragraph 4.2.2 were used in the analog simulation.

4.6.1 Compatibility of ACT control laws with TCV control laws. - Autothrottle Command Augmentation System (CAS) and pitch attitude (θ) and velocity (or flight path - γ) Control Wheel Steering (CWS) modes of the TCV program control laws were selected as representative for simulation. It is felt that ACT systems compatible with the CAS and CWS modes for the pilot will be compatible with TCV systems receiving commands from guidance systems. The two ACT systems that might significantly affect the path of the airplane, Ride Control (RC) via direct lift and the Stability Augmentation System (SAS) for RSS, were also simulated.

Compatibility was investigated at the landing approach condition, for which the CAS and CWS were designed. RC and SAS were simulated with gains developed for their respective design conditions (viz., high speed climb and cruise), and were not optimized for the landing condition.

Table II presents rms altitude, c.g. vertical acceleration, and pitch rate responses to random turbulence for eighteen combinations of ACT and TCV control laws, in the landing condition. All data are presented as decimal multiples of the respective rms responses of the free airplane. The responses are perturbations from straight and level flight. This is indicative of the perturbations from a glide path that would be induced by turbulence.

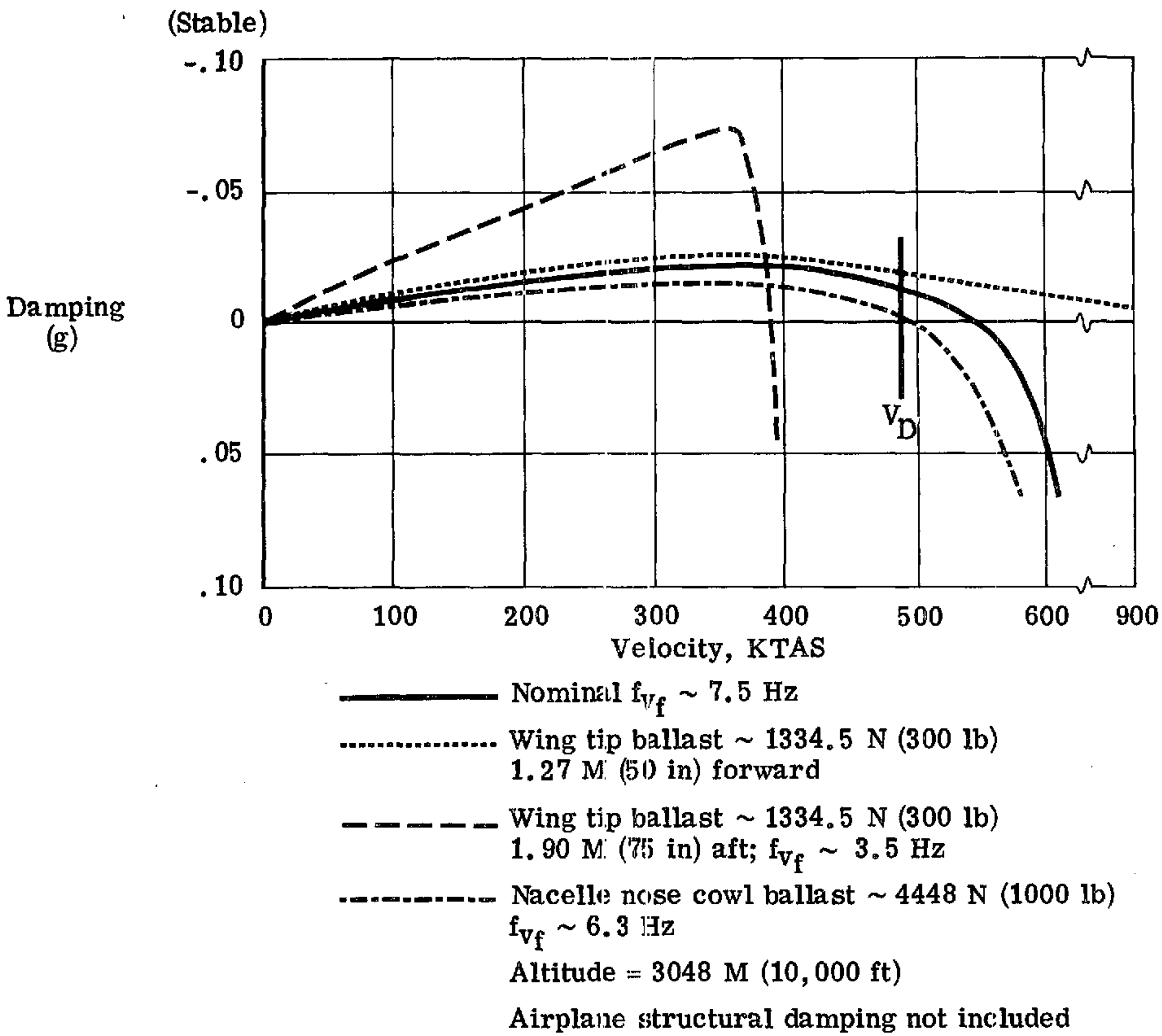



Figure 22: Typical effects of ballast on wing flutter mode

TABLE II

TCV/ACT CONTROL LAW COMPATIBILITY, RMS RESPONSES TO RANDOM TURBULENCE[a,b,c]

| Configuration | Altitude, $h$ | Vertical Acceleration, $\ddot{z}$ | Pitch Rate, $\dot{\theta}$ |
|---|---|---|---|
| Free Airplane | 1.000 | 1.000 | 1.000 |
| RCS | 1.077 | .926 | .945 |
| SAS | .991 | .958 | .927 |
| AT[d] | .976 | .947 | .927 |
| AT + RCS | .976 | .737 | .791 |
| AT + SAS | .973 | .926 | .855 |
| Attitude CWS, $\theta$ | .971 | .916 | .345 |
| RCS | .978 | .916 | .336 |
| SAS | .969 | .789 | .345 |
| AT | .925 | .905 | .345 |
| AT + RCS | .932 | .768 | .336 |
| AT + SAS | .925 | .905 | .336 |
| Velocity CWS, $\gamma$ | 2.287 | 1.179 | 1.336 |
| RCS | 2.304 | .905 | 1.055 |
| SAS | 2.287 | 1.158 | 1.300 |
| AT | 1.061 | 1.200 | 1.391 |
| AT + RCS | 1.063 | .937 | 1.100 |
| AT + SAS | 1.060 | 1.179 | 1.345 |

[a]Landing approach flight condition

[b]All parameters indicated in decimal multiple of free airplane value.

[c]Turbulence intensity, $\sigma_g$ = 2.1 M/sec (7.0 ft/sec)

[d]Autothrottle - TCV

Rate of change of altitude was computed by the equation $\dot{h} = w \cos\theta - u \sin\theta$, where u and w are velocities along the airplane X and Z axes, respectively. Difficulty with drift was experienced in integrating $\dot{h}$ to obtain altitude, and a pseudo integration was performed instead, the transfer function being $1/(S + .04)$. The Laplace transform of $\dot{h}$ being Sh, the output is then $h[S/(S + .04)]$, or altitude through a washout (high-pass) filter with a break frequency of .04 rad/sec (.006 Hz).

Free airplane rms altitude perturbations are increased slightly by the RC system, but it has negligible effect on altitude responses of the autothrottle (AT), attitude CWS, velocity CWS, or combinations of these. When the RC system is added in any of these cases, the acceleration and pitch rate responses are equal to or less than the initial case. This is true for the SAS in all cases for altitude, acceleration and pitch rate.

The performance of the primary ACT functions (e.g., acceleration reduction by RC) is not impressive in this data, for two reasons. As expressed before, the gains used in the RC system and SAS were developed for their respective design flight conditions, and the gains were not optimized for the landing condition. The data does show that the ACT systems as designed are compatible with TCV systems. Secondly, after the RC system analysis was accomplished as described in Paragraph 4.5.1 using the flight spoilers, a design decision was made to use the ground spoilers instead for ride control (see Section 7). Lift, drag, and pitching moment coefficients for the ground spoilers were used in the analog simulation, and the acceleration feedback gain was adjusted to accomplish the same direct force feedback as synthesized. However, the pitch damper (pitch rate to elevator) gain was not compensated for the different spoiler pitching moment.

Compatibility of ACT and TCV control laws was also demonstrated with 1-cos discrete gust forcing functions (reference Paragraph 4.3.2). The peak value of responses are presented in Table III in the same manner the rms results were presented. In every case the RC and SAS responses to gusts are equal to or less than the configuration to which the RC or SAS was added.

The effect of ACT systems on airplane response to pilot commands was demonstrated by applying a six-degree step to the elevator actuator. With control wheel steering engaged, a 40-lb force pulse of 0.5 second duration was applied to the column, which commanded a constant elevator.

Strip chart recordings of responses were made for all combinations of ACT and TCV systems in Table IV. The inputs were applied with and without the TCV autothrottle for all combinations listed.

Again, the data was computed for the landing condition, without optimizing RC and SAS gains for that condition. The SAS has practically no effect on TCV systems. The RC system reduces the initial overshoot in both pitch rate and vertical acceleration. This is exemplified

TABLE III

TCV/ACT CONTROL LAW COMPATIBILITY, RESPONSE PEAKS TO DISCRETE GUSTS [a, b]

| Configuration | Altitude $h$ | Vertical Acceleration, $\ddot{z}$ | Pitch Rate, $\dot{\theta}$ | Pitch Attitude, $\theta$ |
|---|---|---|---|---|
| Free Airplane | 1.000 | 1.000 | 1.000 | 1.000 |
| RCS | .644 | .789 | .968 | .925 |
| SAS | .956 | 1.000 | .935 | .925 |
| AT | .956 | 1.000 | 1.000 | 1.000 |
| AT + RCS | .867 | .775 | 1.000 | .925 |
| AT + SAS | .933 | 1.000 | .952 | .943 |
| Attitude CWS, $\theta$ | .711 | .972 | .484 | .340 |
| RCS | .711 | .746 | .484 | .340 |
| SAS | .711 | .972 | .468 | .340 |
| AT | .644 | .972 | .484 | .340 |
| AT + RCS | .644 | .746 | .484 | .340 |
| AT + SAS | .644 | .972 | .468 | .321 |
| Velocity CWS, $\gamma$ | 1.289 | 1.014 | 1.129 | 1.377 |
| RCS | 1.244 | .789 | 1.000 | 1.075 |
| SAS | 1.267 | 1.014 | 1.113 | 1.340 |
| AT | 1.244 | 1.014 | 1.129 | 1.358 |
| AT + RCS | 1.156 | .755 | 1.000 | 1.075 |
| AT + SAS | 1.222 | 1.104 | 1.113 | 1.321 |

[a] Landing approach flight condition

[b] All parameters indicated in decimal multiple of free airplane value.

TABLE IV

ACT/TCV COMBINATIONS TO SHOW PILOT COMMAND COMPATIBILITY

| CWS \ ACT | NONE | RCS | SAS |
|---|---|---|---|
| None | $\delta_e$ step | $\delta_e$ step | $\delta_e$ step |
| $\theta$ CWS | $\delta_{col}$ pulse | $\delta_{col}$ pulse | $\delta_{col}$ pulse |
| $\gamma$ CWS | $\delta_{col}$ pulse | $\delta_{col}$ pulse | $\delta_{col}$ pulse |

relative to the TCV autothrottle CAS in Figure 23. Without control wheel steering (CWS), an elevator step results in steady state pitch rate and vertical acceleration. The steady state responses are not significantly changed by the RC system, only the initial overshoot. The responses beyond the initial peak are masked by the phugoid mode without autothrottle or CWS, which suppress the phugoid mode.

With CWS, the column force pulse integrates into a constant elevator command. However, the CWS feedbacks essentially null out this command, resulting in pitch rate and vertical acceleration pulses. Figure 24 shows these responses for $\theta$ CWS, with a 29 percent reduction in the peak acceleration.

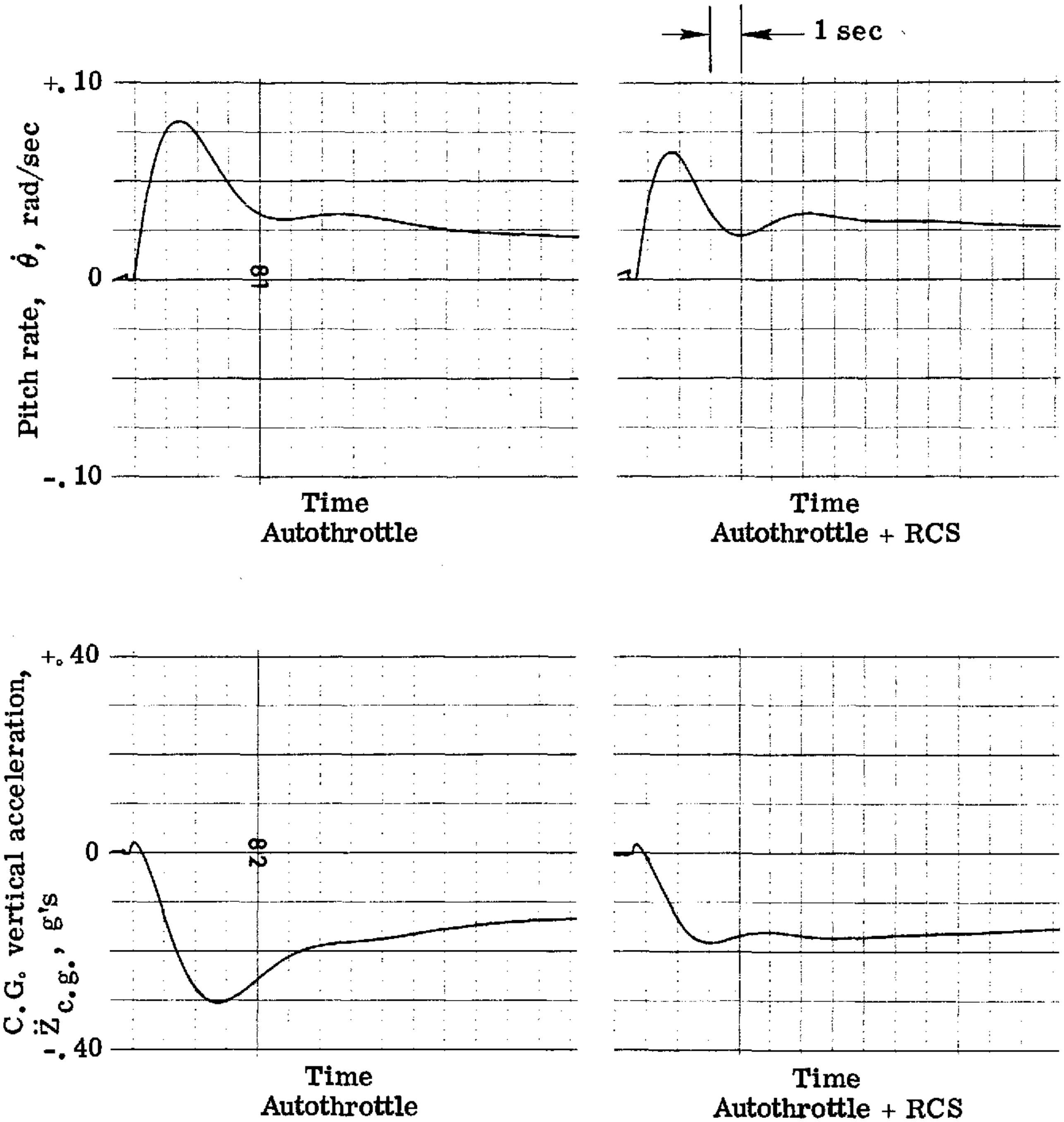
4.6.2 Ride control system actuator performance requirements. - The Direct Lift RC system was evaluated on the analog computer to assess the actuator requirements in terms of rate limit, position limit and bandpass for the symmetrical spoilers. The climb flight condition, 340 KEAS and 3049 M (10 000 ft) altitude, was simulated for this data. A random turbulence forcing function was used with an intensity of 2.1 M (7.0 ft/sec),

On the basis of previous studies, it was decided that break frequencies of 20 rad/sec in the main power stage of the elevator and spoiler actuators would provide completely satisfactory performance. This is confirmed by the data in Figure 25. Performance levels off at twenty rad/sec, and has little sensitivity to errors in break frequency. The RC system acceleration performance is not very dependent upon the elevator, and the elevator actuator break frequency could be reduced, but the bandpass of 20 rad/sec is required for the spoiler actuator.

A required spoiler rate capability of 60 deg/sec was estimated from digital computation of rms spoiler rate. Figure 26 shows the effect of varying spoiler rate limits on RC system performance, while holding deflection limit at the design nominal of ±7.5 degrees. The maximum rate capability is safely beyond the knee of the performance curve.

The RC system was designed for fifteen degrees maximum deflection of the spoilers. That is, plus and minus 7.5 degrees from a 7.5-degree upward bias. It was estimated from spoiler lift coefficients and digitally computed rms deflections that this would be adequate. Figure 27 shows the effect of varying the deflection limit while holding the rate limit constant at the design nominal of 60 deg/sec. Figure 27 confirms that the deflection limit was satisfactory. The limit will not degrade performance, and is not unnecessarily high.

The actuator limits and bandpass will have no effect on the digitally computed rms acceleration performance in Paragraph 4.5.1. Figure 28 shows the pitch rate and acceleration responses to a 1-cos discrete gust, with and without the RC system. The RC system accomplishes a 22 percent reduction in peak acceleration, with the nominal bandpass and limits discussed in the foregoing paragraphs.




V = 118 KEAS
h = 0

**Figure 23:** ACT compatibility with TCV autothrottle - responses to elevator inputs

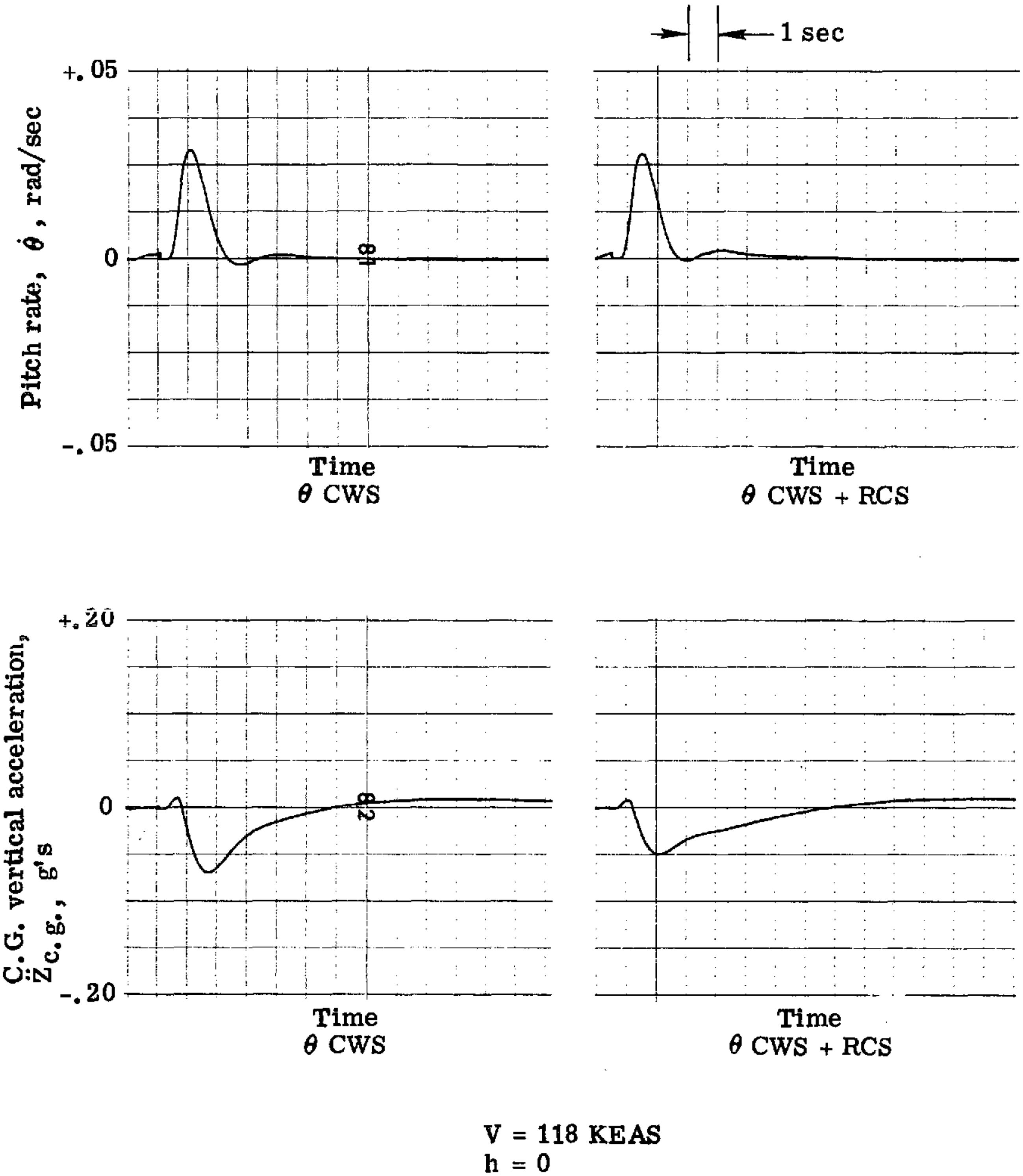



**Figure 24:** ACT compatibility with TCV CWS - responses to elevator inputs

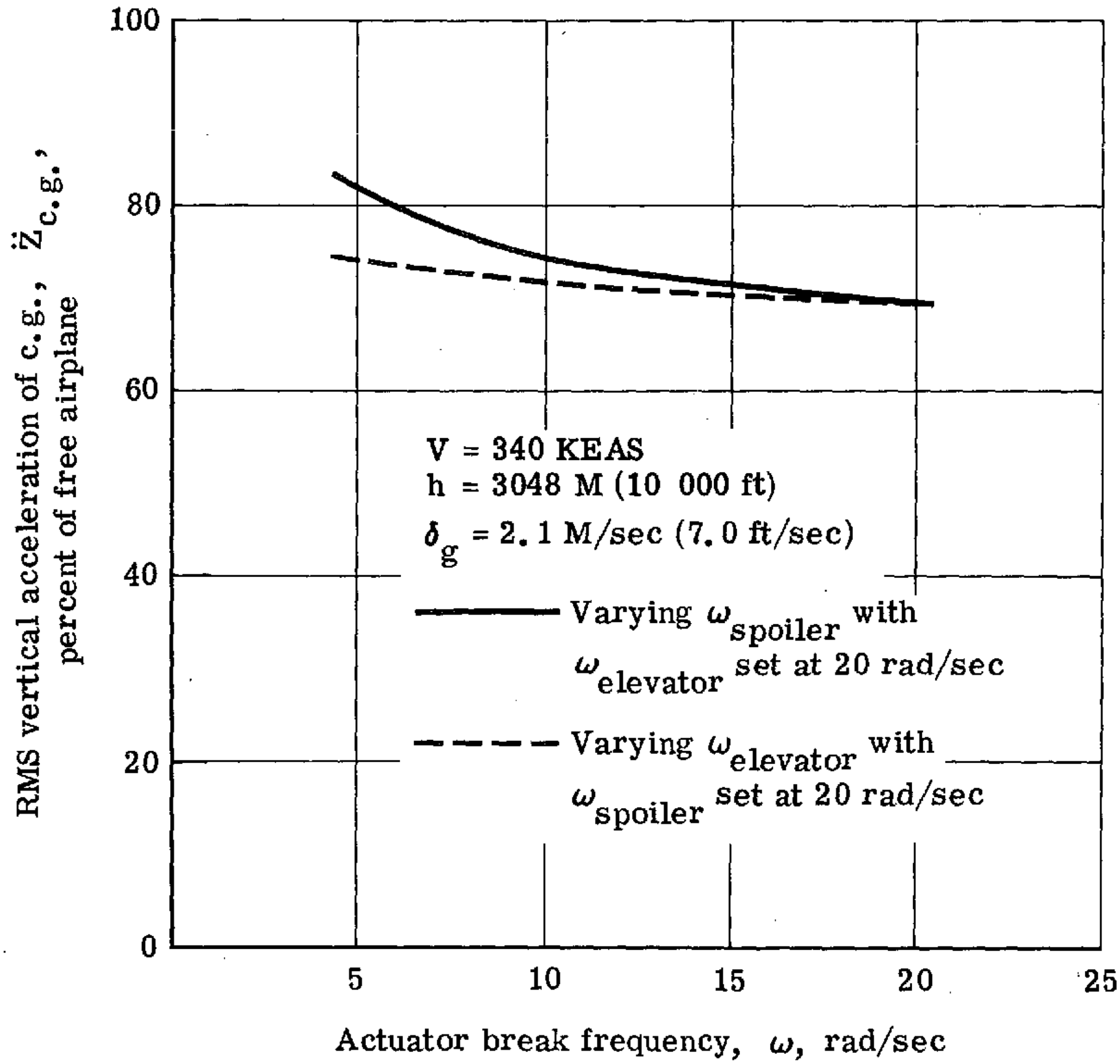



Figure 25: The effect of spoiler actuator break frequency on ride control system performance

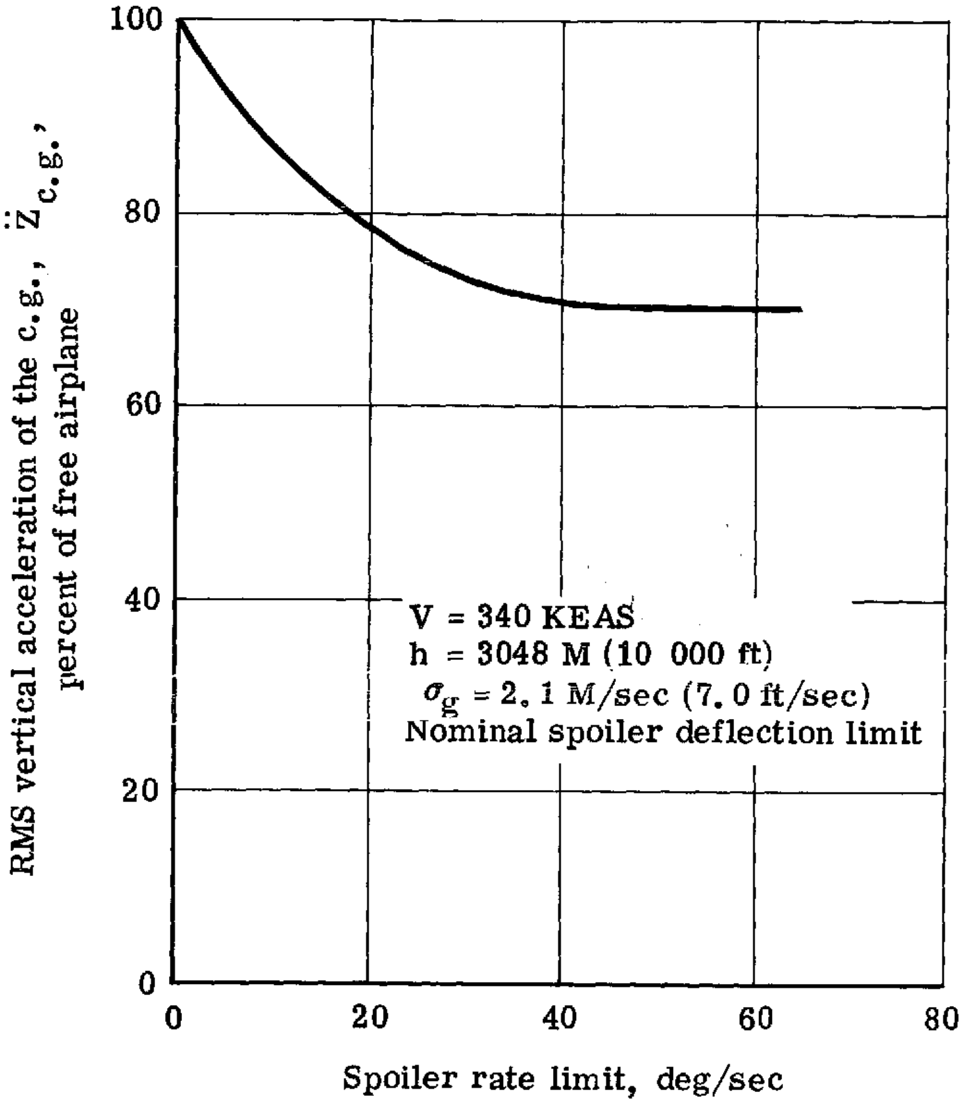



Figure 26: The effect of spoiler rate limits on ride control system performance

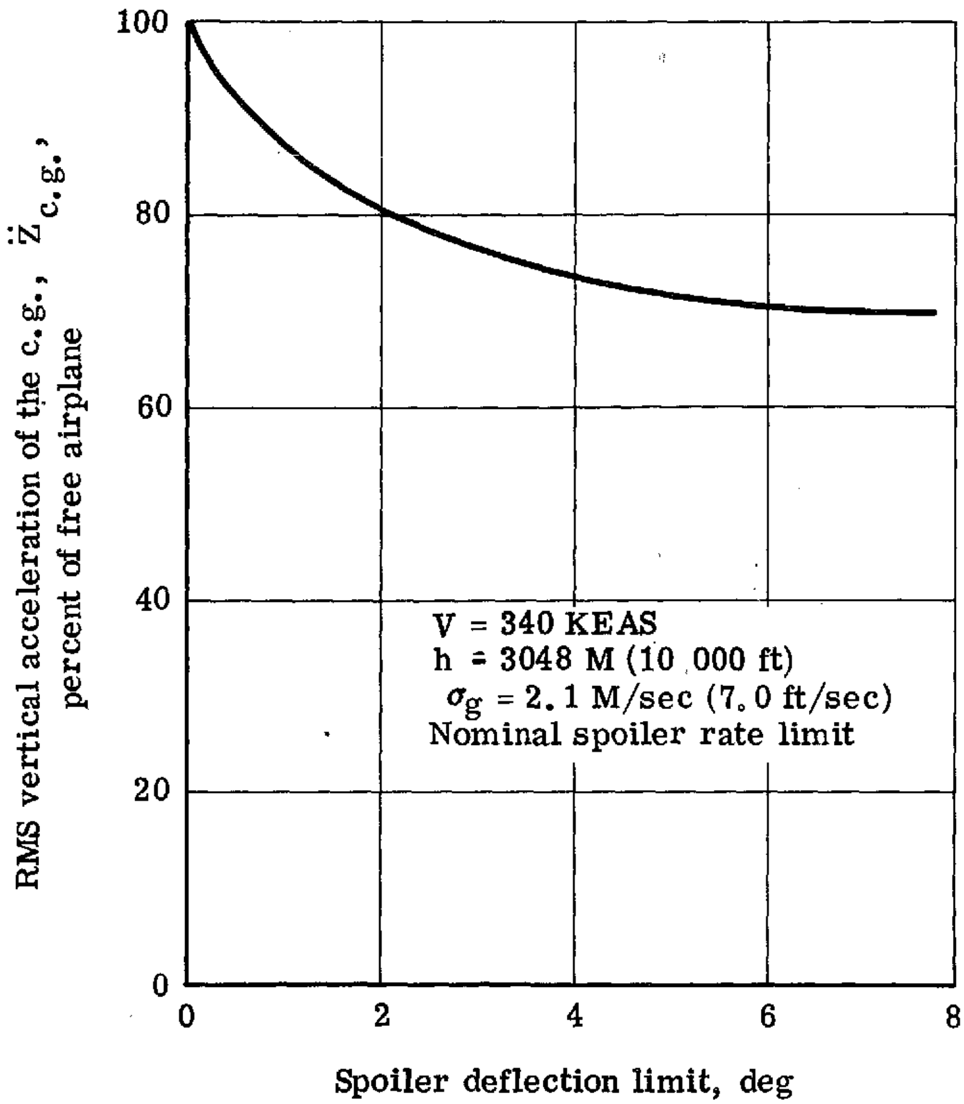
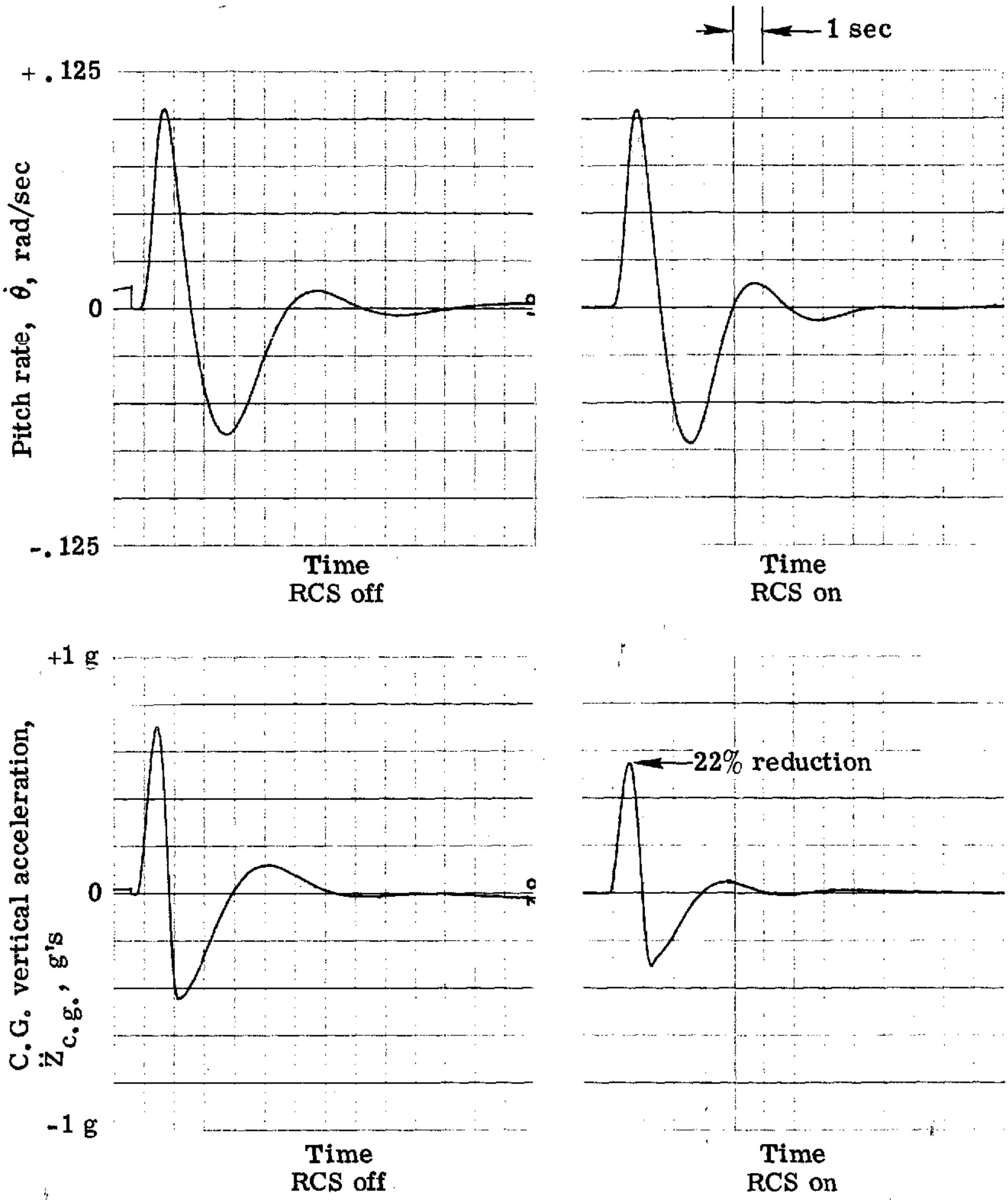



Figure 27: The effect of spoiler deflection limits on ride control system performance




V = 340 KEAS
h = 3048 M (10 000 ft)

Figure 28: Ride control system response to 1-cos discrete gust

## 5.0 CONCEPT FEASIBILITY AND DEFINITION OF NASA 515 ACT PROGRAM PHASES

The practicability of implementation of the various ACT concepts is considered in Paragraph 5.1, along with the analysis results of Section 4, to formulate a conclusion as to whether meaningful flight validations are feasible. Phase I and Phase II NASA 515 ACT flight validation programs are defined in Paragraph 5.2.

### 5.1 Concept Feasibility

The feasibility of each concept is considered individually in the following subparagraphs, in the same general order that the concept flight validation status was considered (Figure 3, Paragraph 3.2).

5.1.1 Digital fly-by-wire. - The aft flight deck of the RSFS presently features FBW control through triply redundant digital computers and interface electronics, with essentially single thread electrical commands to the actuators. It is certainly feasible to incorporate ACT concepts with minor modifications of the existing FBW controls within the scope of Phase I. It is also feasible to modify the flight control hydraulic and electrical power supplies, computers, electronics and actuators to obtain the redundancy required for an all-FBW control system in Phase II.

5.1.2 Relaxed static stability/c.g. control. - Implementation of RSS will require modification of the electrical command capability to the elevator actuators. The primary requirement is for increased redundancy. Existing electrical command authority to the elevator (through the autopilot transfer valve) is sufficient to exceed design structural limits for c.g. positions aft of 27 percent M.A.C. at the cruise flight condition. The 27 percent c.g. position would not constitute a meaningful demonstration of RSS, since the analysis in Paragraph 4.5.3 showed the airplane to be neutrally stable at 31 percent M.A.C. with Mach trim off and 42 percent with the Mach trim on.

At least a fail-operational philosophy should be implemented in Phase I to allow reconfiguration of the c.g. or a change of flight conditions after the first failure. It would be feasible to implement RSS through a redundant secondary (servo) actuator to drive the elevator actuator main valve rod, and the forward flight deck column in parallel. The elevator actuator main power stage is already dual redundant. The rate and frequency response capabilities of this arrangement would be sufficient for RSS. The redundant actuation required for all-FBW in Phase II is satisfactory for RSS.

The feasibility of automatic CGC is considered with RSS because demonstration of either concept is somewhat dependent upon demonstration of the other. A method of controlling the c.g. is required to set up a number of RSS test points in a single flight, and to reconfigure for safe takeoff and landing. And, although useful information can be obtained with CGC regarding fuel savings, demonstration by itself would not be justified, since prototype and production CGC systems have been flown.

The total range over which the c.g. is moved as a result of fuel configurations is only approximately three percent of M.A.C. (from 17 to 20). The feasibility of controlling c.g. with forward and aft water ballast tanks, with appropriate pumps and plumbing between them, was studied. Apparent space for water tanks was observ in the lower forward luggage compartment and at the aft end of the passenger deck.

Figure 29 illustrates the results of preliminary calculations made to determine the amount of ballast required. The farthest forward c.g. of a typically loaded RSFS is 17 percent M.A.C. and occurs when the airplane gross weight is 346 960 N (78 000 lbs). The farthest aft c.g. is 20 percent, and occurs at 382 545 N (86 000 lbs). The vertical bars show the range of c.g. control that can be accomplished with 22 241 N (5000 lbs) of water. It allows control of the c.g. aft to 31 percent, the point of neutral static stability in the cruise condition with Mach trim off.

This amount of water is also about the maximum that can be accommodated in the available space. Adding 26 689 N (6000 lbs) of static ballast to the 382 545 N (86 000 lbs) airplane with water ballast results in the structural limit on RSFS gross weight of 431 475 N (97 000 lbs). This amount of ballast, distributed so that its c.g. is at BS 22.9 M (902 in), will allow control of the airplane c.g. aft to 42 percent M.A.C., the point of neutral stability with the Mach trim on. The complete range of control is indicated in Figure 29. The installation of water and static ballast is feasible, and the c.g. can be controlled within safe limits for landing and takeoff.

5.1.3 Gust load alleviation. - In a Phase I ACT program, protection against electrical hardovers will be achieved by limiting the electrical command authority to approximately half of manual control authority, as it is in the TCV program, rather than by a high degree of redundancy. It is not felt that effective demonstration of load alleviation from large discrete gusts is feasible in conjunction with manual electric control with the limited authority.

Figure 6, Paragraph 4.4, indicated that an exceptionally high bandpass elevator actuator, and therefore high deflection rates and hydraulic flow, would be required to accomplish random vertical GLA on the aft body. The TCV method of electrical commands through the autopilot transfer valve with parallel motion of the forward flight deck column would certainly not be adequate. In fact, the existing elevator power control units (PCU), the main stage, would not be adequate. The rate and bandpass are insufficient, and the force path from the actuators to the elevators is purposely quite compliant because of early actuator/load dynamic problems.

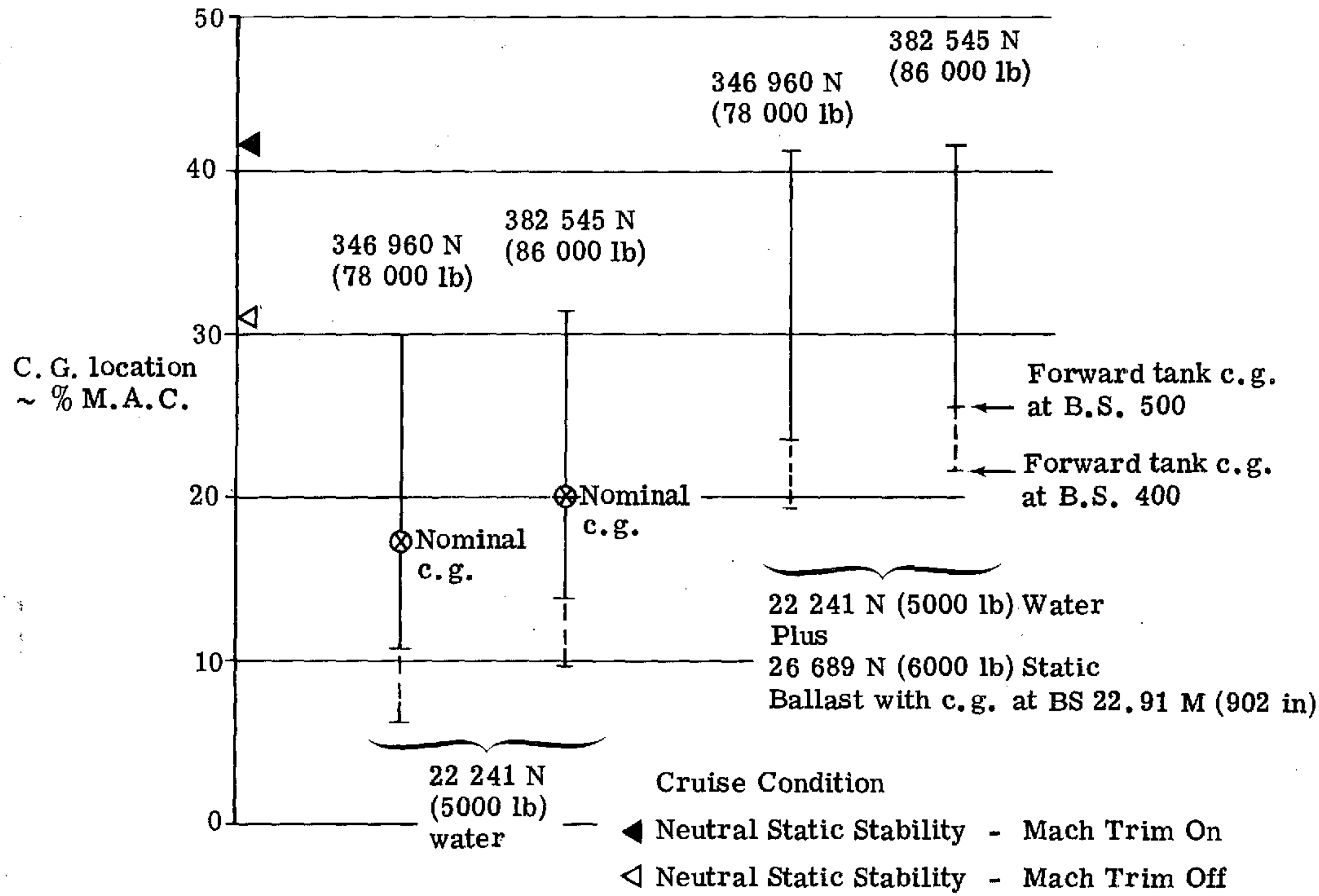



Figure 29: C.G. control range of NASA 515 airplane

The required performance could only be obtained with series control. Series control would have to be accomplished through splitting the elevator surfaces and adding ACT actuators to dedicated elevator segments to maintain parallel control for the manual electric mode and manual reversion for the forward flight deck after two hydraulic failures. In summary, aft body GLA is feasible with split elevators and additional actuators. However, the high feedback control frequencies (up to 8 or 9 Hz) would constitute a very severe requirement on digital implementation.

The 737 aileron actuators are identical to the elevator actuators, and are inadequate for wing GLA for most of the same reasons that the existing elevator actuators are inadequate for aft body GLA. In addition, the existing aileron actuators are located in the wheel well and act through cable runs, bell cranks, and rods to the ailerons, altogether unsatisfactory for GLA. Split ailerons are required with added actuators to dedicated ACT aileron segments. However, the bandpass and rate requirements are not as severe as for the aft body. The required aileron deflection and rate are feasible if the system synthesized in Paragraph 4.5.2 were implemented with a 40 percent span segment with the feedback gain compensated accordingly.

There will be little problem in summing GLA feedback into the all-FBW actuators for Phase II, and the redundancy will allow use of full authority for ACT, accommodating large discrete gust alleviation.

5.1.4 Maneuver load alleviation. - The analysis in Paragraph 4.5.5 showed that an effective demonstration of MLA is not feasible with existing surfaces (Phase I). Greater authority will be available in Phase II, making the demonstration feasible. The main requirement for flight validation of MLA is compatibility and interaction with other concepts over the full envelope. A MLA concept can be implemented for this purpose with very little additional cost in Phase II, utilizing control surfaces that are required for other concepts (wing GLA and RC via direct lift). Although this MLA implementation is satisfactory for flight experimentation, it is not considered adequate for existing commercial aircraft fleet usage. To develop the lift authority required, a technique is needed for articulating the aft segment (flaperon) of a multi-slotted Fowler flap in both the extend and retract position. Development of this type was not within the scope of this study.

5.1.5 Ride control. - Ride control via direct lift must be accomplished with symmetrical spoilers (biased in a partially-up position), especially for the minimum-modification scope of Phase I. Use of the flight spoilers would require integrated spoiler actuators; i.e., mechanical input from the manual control or TCV roll control plus series electrical input for RC (symmetrical). This would require considerable time and expense for an actuator development program.

Use of ground spoilers, numbers 1, 4, 5 and 8 (Reference Figure 1), is more feasible for Phase I so that there is no interface with the complex lateral control through the flight spoilers. Simple actuators can be used with electrical input and electrical feedback.

Implementation of ride control via direct lift is feasible, and the analysis of Paragraph 4.5.2 showed that significant payoff could be demonstrated.

Ride control via structural mode suppression is feasible by addition of horizontal canards and actuators on the forward body. A reduction in rms pilot station vertical acceleration of 43 percent was shown by the analysis in Paragraph 4.5.4.

5.1.6 Flutter suppression. - It was concluded in Section 3.2 that the real requirements regarding FS concept validation involve multiple modes, higher frequency modes, and violent modes. NASA has expressed that this kind of flight testing should be accomplished with a remotely piloted vehicle. Furthermore, accurate definition of the airplane modification required to produce a moderate mode for flight validation on the NASA 515 would require considerably more analysis. The costs of a FS flight test program would be high including analyses, airplane modifications and ground vibration tests. Considering these factors, active FS is not recommended for demonstration on the NASA 515; however, it can be accomplished with sufficient analysis.

5.1.7 Envelope limiting. - Envelope limiting is feasible on the NASA 515 with fewer airplane and hardware modifications than any of the other ACT concepts. Most of the design task will involve computer software. System and implementation concepts are presented in Section 7.

## 5.2 Definition of NASA 515 ACT Flight Validation Program Phases

Considering the guidelines specified for Phases I and II, the flight validation requirements, analysis results, and feasibility assesments in the foregoing sections, Phase I and Phase II programs are defined in Paragraphs 5.2.1 and 5.2.2, respectively. System designs are described in Section 7.

5.2.1 Phase I. - The purpose of Phase I has been defined to be flight validation; that is:

- validation of ACT concepts in several specified areas
- validation of ACT digital system performance
- validation of analytical and flight test techniques for design of commercial ACT aircraft.

Repeating, the scope of Phase I is restricted to relatively low cost, minimal modifications of the airplane, and minimal impact on the TCV program schedule and functions. Control laws will be mechanized digitally, and the systems will be flight tested at non-critical safety of flight conditions.

Program options have been prepared for the NASA 515 aircraft that satisfy the Phase I purpose and scope, with costs and schedules for each. The options are summarized in Table V, below, with the primary ACT deficiency satisfied by each. Costs and schedules for each option are presented in Section 8.

TABLE V

PROPOSED ACT FLIGHT TEST VALIDATION PROGRAMS
(PHASE I)

| Proposed Program Options | ACT Deficiency Satisfied |
|---|---|
| 1. Ride control via direct lift and direct lift for maneuvering | - Conceptual validation of RC via direct lift<br>- Handling qualities |
| 2. Gust load alleviation (wing root) | - Modal suppression performance with digital systems |
| 3. Envelope limiting | - Concept validation |
| 4. Relaxed static stability and automatic c.g. control | - Handling qualities<br>- Range improvements |
| 5. Ride control (modal suppression) | - Modal suppression performance with digital systems |

5.2.2 Phase II. - The purpose of Phase II (Option 6) has been defined to be development, as opposed to the validation in Phase I. Specifically, the purpose is to develop and flight validate FBW/ACT digital implementation for large commercial aircraft. The goals are to develop hardware and software design techniques and criteria and to flight validate system design guidelines, performance (over the full flight envelope), compatibility, failure immunity, and implementation techniques.

The scope of Phase II (Option 6) will include implementation of all the concepts from Options 1 through 5, plus maneuver load control. The digital FBW and critical ACT flight control systems will be quadruply redundant. Design descriptions are presented in Section 7. Sufficient flight tests will be conducted to verify system design, techniques, safety, performance and compatibility throughout the flight envelope.

The scope of Option 6 is significantly larger than any of the other options. For instance, flight validating failure immunity includes failure detection, isolation, and compensation for quadruply redundant sensors, computers, electronics, and actuators. It also includes the redundancy management of electrical and hydraulic power supplies.

## 6.0 DEMONSTRATION PROGRAM GUIDELINES AND CRITERIA

This section contains suggested guidelines and criteria for ACT flight validation program options 1 through 6 (reference Paragraphs 5.2.1 and 5.2.2). These criteria are expressed in somewhat general terms. Salient points are addressed to help define implementation philosophy and expected program results for planning of the respective options. More detailed criteria development will be required during the analysis phase of flight validation programs.

Paragraph 6.1 pertains to Options 1 through 5 (Phase I), and paragraph 6.2 pertains to Option 6 (Phase II).

### 6.1 Options I Through 5 (Phase I)

All options will be built upon the nucleus of digital fly by wire, and existing NASA 515 equipment will be utilized to the maximum extent possible. All ACT concepts will be designed to operate only in conjunction with the aft flight deck piloted modes.

6.1.1 Performance. - Analysis will be conducted to determine performance at least for the flight phases indicated for each concept in Table VI, and for an appropriate range of weights. The "design" flight condition for each concept is indicated by an asterisk. Performance goals for the design flight conditions are stated in the following subparagraphs.

TABLE VI

ACT ANALYSIS FLIGHT CONDITIONS

| Option | Concept | Flight Phase: Climb | Low Alt. Cruise | High Alt. Cruise | Holding | Terminal Maneuvers | Approach |
|---|---|---|---|---|---|---|---|
| 1 | RC via direct lift | * | X | X | X | X | X |
| | Direct lift for maneuvers | | | | X | X | * |
| 2 | GLA - wing root | X | * | X | X | X | X |
| 3 | Envelope limiting | X | * | X | X | X | * |
| 4 | Relaxed static stability and c.g. control | X | * | X | X | | |
| 5 | RC via mode suppression | * | X | X | X | X | X |



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For reference, the altitudes and velocities used to represent flight phases in Table VI for ACT feasibility and the TCV studies are listed in Table VII, below:

TABLE VII

TCV FLIGHT CONDITION DEFINITIONS

| Flight Phase | Altitude M | Altitude (Ft) | Velocity, KEAS |
|---|---|---|---|
| Climb | 3048 | (10 000) | 340 |
| Low alt. cruise | 6096 | (20 000) | 350 |
| High alt. cruise | 8534 | (28 000) | 311 |
| Holding | 1524 | (5000) | 230 |
| Terminal maneuvers | 1524 | (5000) | 180 |
| Approach | 229 | (750) | 130 |

With all analysis parameters fixed at nominal, each ACT system concept will achieve the following performance.

6.1.1.1 Option 1 (ride control/direct lift): The vertical ride control system, utilizing direct lift control surfaces to translate the c.g., will reduce vertical acceleration at all points along the fuselage by a minimum of thirty percent.

Direct lift control will produce a minimum of 0.10 g vertical acceleration for pilot maneuvers in the approach condition, and will retain a capability for rapid pullup if required.

6.1.1.2 Option 2 (gust load alleviation - wing root): The wing GLA system will reduce rms vertical bending moment at the root chord plane by a minimum of twenty percent. The PSD of vertical bending moment will be reduced for all frequencies up to thirty rad/sec.

6.1.1.3 Option 3 (envelope limiting): The airplane will automatically be limited to stall angles of attack, and to each of the following RSFS operating limits (in a production system these would be structural limits):

| | |
|---|---|
| Mach number | .84 |
| Velocity, KEAS | 350 |
| Load factor, g's | 1.5 |

6.1.1.4 Option 4 (relaxed static stability/c.g. control): Acceptable flight characteristics will be obtained with the pitch SAS for the full range of c.g. control, as indicated by:

1. maneuvering stability - prescribed stick force per g

2. static stability - prescribed stick force per knot

3. stall characteristic - pull with increasing force to reach stall

4. transient response - short period roots in a prescribed frequency/damping region and low frequency roots stable with a maximum time to half amplitude.

The c.g. control system will automatically provide commanded c.g. locations to the normal in-flight forward limit (10.3 percent M.A.C), and aft c.g. locations to 31 percent M.A.C. without static ballast. A maximum aft c.g. location will be determined where the longitudinal axis is unstable, with a time to double amplitude, $(t_2)$, $\geq 20$ seconds with the Mach trim system off (approximately 36 percent M.A.C.). The rate of fuel consumption will be determined as a function of c.g. location between the two limits.

6.1.1.5 Option 5 (ride control via mode suppression): The rms of vertical acceleration at the pilot station will be reduced by at least sixty percent in the frequency band above the rigid body response, without increasing the rms in the rigid body range of frequencies.

6.1.2 System failures. - All FBW/ACT control surface commands will be fail operational/fail safe through the digital computers and interface electronics, for all options, 1 through 5.

Elevator actuation for FBW/ACT will be fail operational/fail safe. All other control surface actuation in Options 1 through 5 will be fail safe.

"Fail operational" is defined as continued operation without degradation in the event of any single failure, other than those whose probability is extremely remote. "Fail safe" is defined as automatic detection of the first failure and shutoff of the FBW/ACT systems without unsafe transients. Immediate indication of the failure will be given to both crews, and the forward flight deck will resume control. An event is "extremely remote" when, although it is theoretically possible, it is not expected in the life of an individual aircraft.

In option 4, the c.g. control system will be able to empty either the forward or aft tank after a single failure of the water transfer system. Either dumping the water overboard or transferring it to the other tank is permissible.

6.1.3 Tolerances. - Nominal system performance of each ACT concept will be met with mathematical model tolerances included in the analyses. Model tolerances will consider the basic airplane dynamic characteristics, control surface dynamics and effectiveness, and system hardware. Each of these variations will be evaluated independently with all other parameters held at nominal values:

- Feedback gain variations of ± 25 percent
- Feedback time constant variations of ± 25 percent
- Feedback sensor location variations of ± 36 inches parallel to the local elastic axis
- Feedforward gain variations of ± 25 percent

6.1.4 Stability margins. - Required gain and phase variations about nominal are defined in Table VIII for all aerodynamically closed loops. Table VIII is taken from the user guide to Military Specification MIL-F-9490D[8]. With either these gain or phase variations included, no instabilities shall exist except oscillations with amplitudes within those allowed for residual oscillations in Paragraph 6.1.6. In multiple loop systems, variations shall be made with all gain and phase values in the feedback paths held at nominal except for the path under investigation. A path is defined to include those elements connecting a sensor to a force or moment producer.

TABLE VIII

GAIN AND PHASE VARIATION REQUIREMENTS
(GV and PV)

<table>
<tr><th colspan="2">Mode Frequency</th><th colspan="2">$V_{O_{min}}$ (minimum operational airspeed) to $V_{O_{max}}$ (maximum operational airspeed)</th></tr>
<tr><td colspan="2">$f_M < 0.06$ Hz</td><td colspan="2">GV = ± 4.5 db<br>PV = ± 30°</td></tr>
<tr><td colspan="2">$0.06 \leq f_m <$ first aeroelastic mode</td><td colspan="2">GV = ± 6.0 db<br>PV = ± 45°</td></tr>
<tr><td colspan="2">High frequency modes</td><td colspan="2">GV = ± 8.0 db<br>PV = ± 60°</td></tr>
<tr><td rowspan="2">$f_M$ > high frequency modes</td><td>$f_M \leq 8$ Hz</td><td>GV = ± 8.0 db</td><td rowspan="2">Magnitude $\leq$ 0 db at all frequencies</td></tr>
<tr><td>$f_M > 8$ Hz</td><td>GV = ± $f_M$ DB</td></tr>
</table>

Each new or modified control surface will be analytically shown to have positive stability with the hydraulic actuator pressure at 0 and 3000 psi from $V_{O_{min}}$ to $V_{O_{max}}$.

6.1.5 Actuators. - New control surface actuators will be designed to have a minimum of 6 db gain margin and 40 degrees phase margin with surface attached.

new primary actuators will have electrical feedback to provide flexibility in obtaining dynamic response and stability margins.

The actuator/surface mode will have a damping ratio greater than 0.05. The dynamic response, surface position to commanded position, will have an amplitude ratio less than 1.0 at the actuator/surface mode frequency.

6.1.6 Residual oscillations. - Sustained residual oscillations at the pilot station or any passenger station due to either structural vibration or limit cycles in the control system will be less than 80 percent of the acceleration levels (zero to peak) shown in Figure 30. These criteria are based on human vibration perception data contained in Reference 9.

6.1.7 Flying qualities. - Each ACT concept will be synthesized so that existing handling qualities from the aft flight deck will not be significantly degraded. Quantitative requirements will be developed in at least the following areas during the analysis phase of flight validation programs, to assure that this criterion is met:

- short period frequency and damping
- phugoid damping
- pitch rate and c.g. vertical acceleration responses to column commands
- column force and displacement gradients to airplane vertical acceleration
- Dutch roll frequency and damping
- roll mode time constant
- spiral mode time to double amplitude

Control authority from the forward flight deck will not be reduced except after an ACT system failure; e.g., loss of the GLA aileron segment.

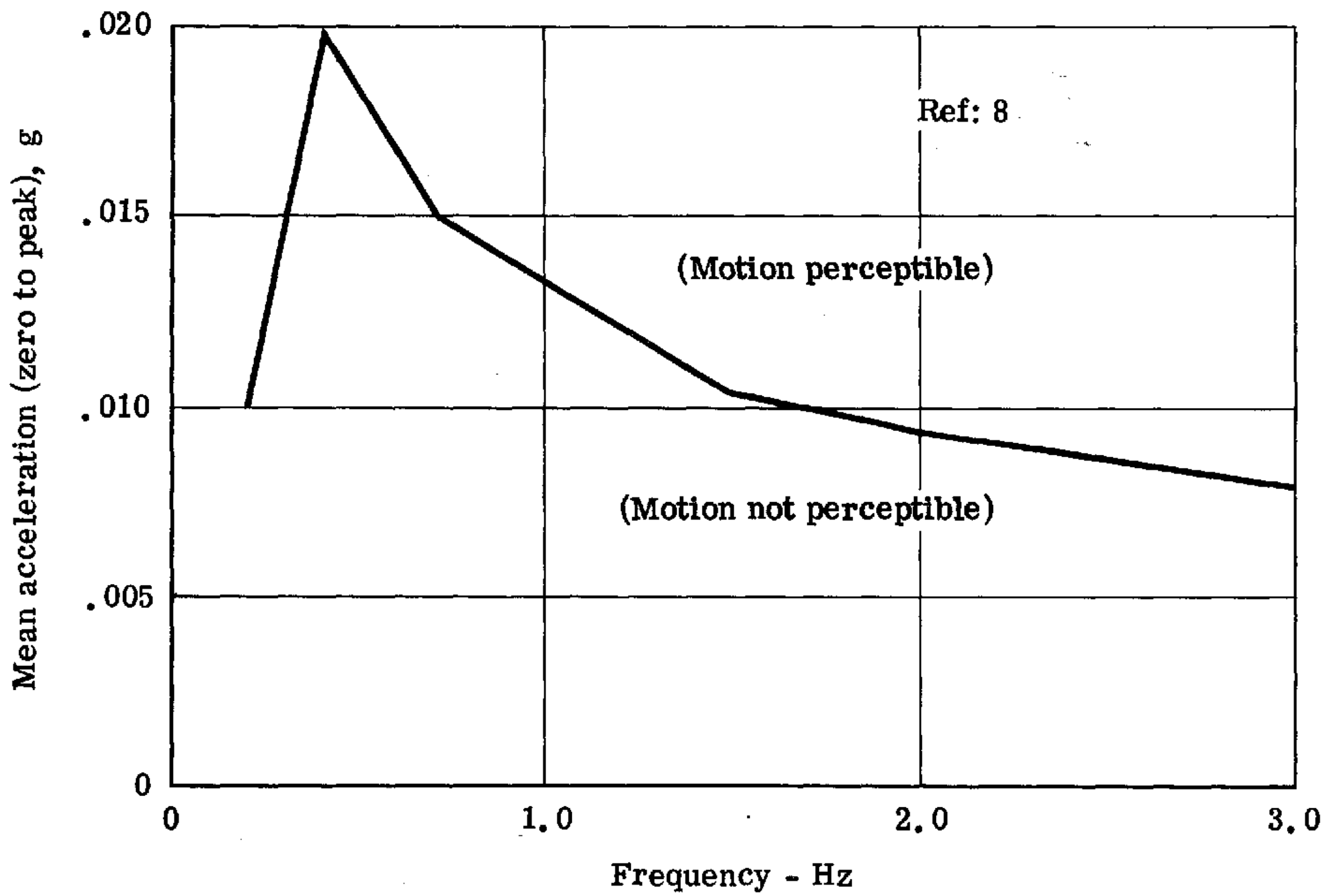


**Figure 30:** Perceptible vertical acceleration thresholds

6.1.8 Dynamic gust loads. - The atmospheric model to be used for analytical predictions will be the von Karman spectrum described in Paragraph 4.3.1. The turbulence scale length, L, will be a function of altitude above the local terrain, h, as follows:

| h | L |
|---|---|
| $h \leq 152$ M (500 ft) | 152 M (500 ft) |
| 152 M (500 ft) $\leq h \leq 762$ M (2500 ft) | L = h |
| $h \leq 762$ M (2500 ft) | 762 M (2500 ft) |

ACT systems will be stable for turbulence levels from 0 to 4.6 M/sec (15 ft/sec) rms. The performance requirements of ACT concepts will be met for turbulence levels with a probability of exceedance of 0.01 at the respective design conditions.

None of the ACT concepts will increase dynamic gust loads by more than five percent at critical locations.

6.1.9 Ride qualities. - None of the ACT concepts will increase rms accelerations along the fuselage by more than five percent.

6.1.10 Operating limits. - None of the ACT concepts will decrease the RSFS Mach, velocity, maneuvering, or gross weight operating limits.

6.2 Option 6 (Phase II)

The fly by wire control and ACT systems will be implemented digitally, and all critical functions will be quadruply redundant. The criteria suggested for Option 6 (Phase II) relate primarily to failure philosophy and safety.

6.2.1 General failure mode requirements. - FBW/ACT concepts will have at least the level of system integrity indicated in Table IX. Failure philosophy terms used in Table IX are defined below:

Two fail operational - Flight critical system functions shall not be lost after any two failures with the exception of single failures or failure combinations that have an extremely remote probability of occurrence.

Fail operational/passive - Flight critical system functions shall not be lost after a single failure with the exception of failures that have an extremely remote probability of occurrence. A second failure shall not cause a control surface command of greater than ten percent of full scale deflection.

Fail passive - Any failure shall not cause a surface command of greater than ten percent of full scale deflection.

TABLE IX

SYSTEM FAILURE PHILOSOPHY - OPTION 6

| ACT Concept | Failure Philosophy |
|---|---|
| Fly by wire | Two fail operational |
| Relaxed static stability | Two fail operational |
| Gust load alleviation (wing root) | Fail operational/passive |
| Envelope limiting | Fail operational/passive |
| Automatic c.g. control | Fail operational* |
| Direct lift control for maneuvers | Fail passive |
| Ride control via direct lift | Fail passive |
| Ride control via mode suppression | Fail passive |
| Maneuver load control | Fail passive |

*Must be able to empty either tank after first failure in transfer system by either dumping overboard or transferring to other tank.

6.2.2 Design criteria. - To be accepted by the commercial air transport industry as a credible FBW/ACT demonstrator, the aircraft must be equipped with systems which possess safety and predicted reliability commensurate with existing transport airplane standards. The unmodified 737 (or the RSFS, from the forward flight deck) can be flown with only mechanical inputs from the pilot's controls to the control surfaces. Electrical and hydraulic power supplies are not essential for continued safe flight. In the FBW/ACT configuration complete reliance is placed on the availability of electrical and hydraulic power systems as well as on the FBW system. As part of the NASA 515 Option 6 program, it will be necessary to establish the reliability and redundancy configuration of these systems in considerable detail. A comprehensive set of implementation requirements and ground rules must be developed during Option 6, based on FAR Part 25, MIL-F-9490C and MIL-F-8785B, and also on Boeing experience with the design of safety-of-flight systems. The following is a tentative set of such groundrules.

6.2.2.1 General FBW/ACT implementation: The existing flight control system and existing hydraulic and electrical power supplies shall provide the basis for the new configuration. When system modification is required to meet the special FBW/ACT requirements developed hardware shall be used, except when new design and construction is the least costly approach or when developed hardware is not available. Accepted commercial transport aircraft design practices shall be followed when applicable.

The FBW and ACT systems shall use digital computation and shall be designed to satisfactorily demonstrate the safety and functional reliability of all-FBW and critical ACT control systems. The system shall employ multi-channel state-of-the-art digital computers that are completely programmable with cross channel monitoring techniques for failure detection and reporting.

The FBW/ACT system and power supplies shall be designed so that the airplane is controllable for continued safe flight and landing up to 20 minutes following loss of both engines. It will be assumed that electrical and hydraulic power are not available from windmilling engines at landing approach speeds.

6.2.2.2 Flight critical elements: Design shall be based on use of a quadruply redundant system from sensors through the servo elements. The flight-safety critical elements of the FBW/ACT system shall take automatic fault clearing action and continue to operate without excessive motion transient or degradation in performance for the following failure conditions:

a) Any single failure except (1) a jam of a surface actuator to a level greater than the maximum force capability of the actuator, or (2) a jam of an actuator valve or linkage system to a level greater than the output capability of the secondary actuator.

b) Multiple dissimilar failures which leave intact three independent success paths for any system function.

c) Second like failures which occur after the first failure has been cleared.

d) Multiple dissimilar failures which leave intact two of the three remaining success paths after the second like failure has been cleared.

Simultaneous occurrence of two or more failures shall be extremely remote (see Paragraph 6.1.2).

6.2.2.3 FBW/ACT non-flight-critical elements: Those elements of the FBW/ACT system which are not essential for safe flight may have a different configuration from the flight critical elements of the system. The following groundrules apply:

A clearly distinguishable warning shall be provided for a failure which could result in an unsafe condition if the pilot were not aware of the failure.

Failures occurring within the normal flight envelope which require pilot corrective action shall not require exceptional piloting skill either for deactivation of the system or for overriding the failure by movement of the flight controls in the normal sense.

The airplane shall be safely controllable when the failure occurs at any speed or altitude within the approved operating limitations that is critical for the type of failure being considered.

Following the failure the airplane characteristics shall not be impaired below a level needed to permit continued safe flight and landing.

6.2.2.4 Electrical power supplies: The reliability of the electrical power supply systems shall be compatible with the overall safety requirement (6.2.2.1).

Four independent electrical power sources for the flight critical equipment shall be provided to be consistent with the requirement for continued undegraded operation after two electrical failures.

Electrical power sources must be isolated electrically, thermally and mechanically such as to prevent a single failure from degrading power beyond tolerable limits on two or more flight critical busses simultaneously.

At least two isolated electrical supplies shall be provided which will continue to power the FBW/ACT system for 20 minutes following loss of both engines.

6.2.2.5 Hydraulic power supplies: The reliability of the hydraulic power supply systems shall be compatible with the overall safety requirement (6.2.2.1).

Four independent hydraulic power supply systems shall be provided to be consistent with the requirement for continued operation following loss of any two hydraulic power supplies.

Complete loss of one hydraulic power supply shall not result in a degradation of system performance.

The hydraulic systems shall be designed with strict segregation, with no interconnect provisions.

Actuators shall be supplied by at least three hydraulic systems or be provided with a sufficient restraint to prevent flutter or buzz in the event of a double hydraulic system failure.

At least one isolated hydraulic supply shall be provided which will continue to power the FBW/ACT system for 20 minutes following loss of both engines.

## 7.0 PRELIMINARY DESIGN

A preliminary design was accomplished for each program option to define systems that would perform the ACT functions and be representative regarding costs. The salient feasures of the preliminary designs are described for Options 1-5 (Phase I) in Paragraph 7.1 and for Option 6 (Phase II) in Paragraph 7.2.

### 7.1 Options 1-5 (Phase I)

This paragraph describes general design requirements pertaining to all options, 1-5. Paragraphs 7.1.1 through 7.1.4 discuss design requirements in specific areas, still pertaining to all options, 1-5. Paragraph 7.1.4 and its subparagraphs present design requirements pertaining to specific program options, and describe the design configurations developed for program planning.

The NASA 515 airplane, as modified for the TCV program, is the baseline configuration from which to modify for each option. No prior, simultaneous, or subsequent implementation of any other option is assumed. ACT systems will be used only in conjunction with aft flight deck piloted modes.

The failure philosophies specified in Section 6.1.2 will be observed, which requires a fail operational/fail safe system from sensors through computer and interface electronics for all control surface commands, and extended through the hydraulic actuators for the elevators.

Existing aft flight deck pilot control will not be degraded by the ACT systems, and the authority of the forward flight deck control will not be reduced except after an ACT system failure.

In forward flight deck control, manual reversion will be retained through the elevator and aileron actuators after two hydraulic failures.

ACT concepts may be implemented with parallel motion of forward flight deck controls for any concept (1-5) if ACT response requirements can be met.

Performance of the ACT concepts will be demonstrated only at selected design conditions, which, in general, eliminates the requirements for scheduling system elements as a function of flight condition.

7.1.1 Elevator actuation. - The preliminary design for all options, 1-5, includes a triplex secondary (servo) actuator that would drive the main valve rod on the elevator actuators, and the forward flight deck column in parallel. Figure 31 shows a triplex secondary actuator conceptual block diagram drawn for the utilization of three 747



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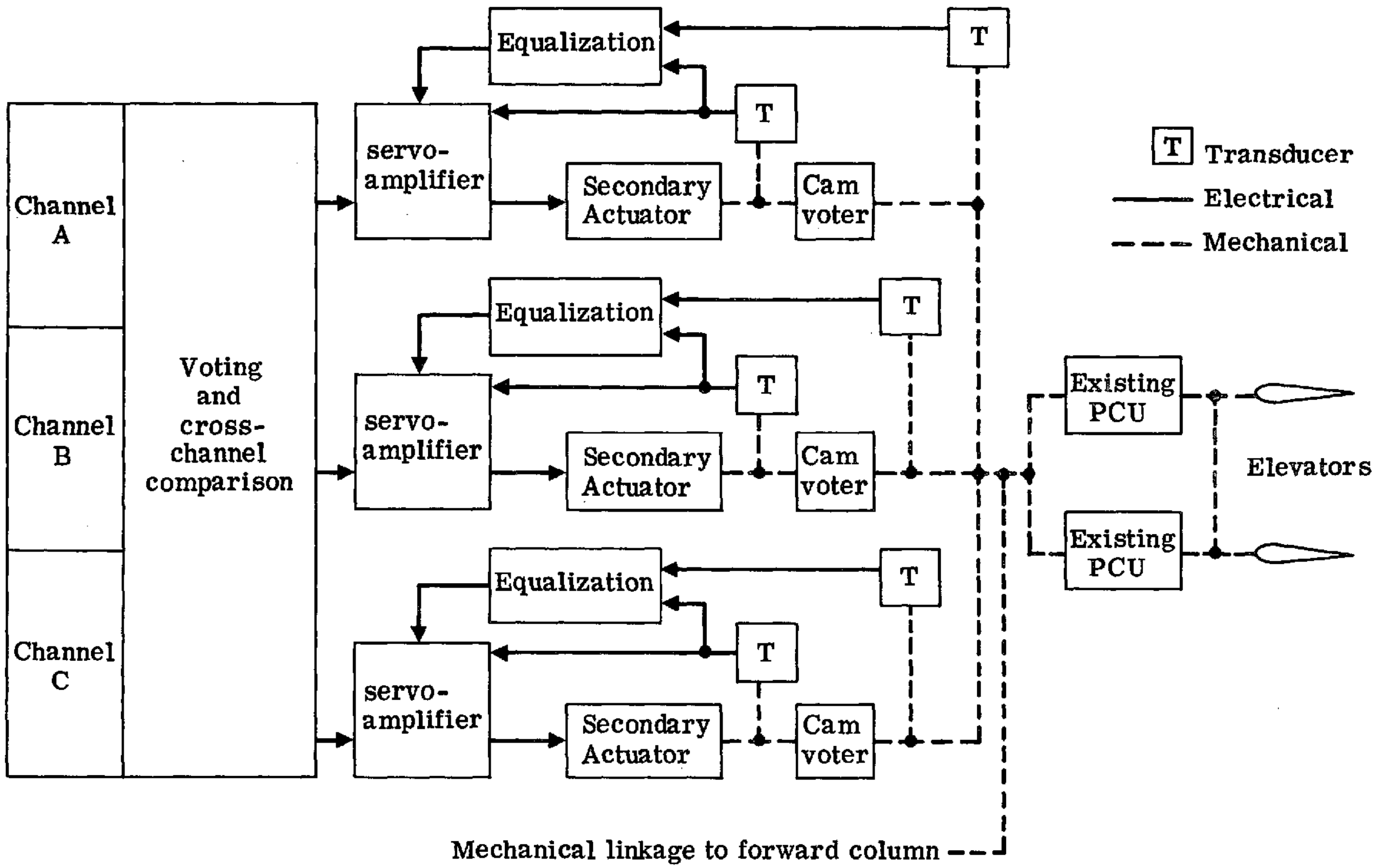



Figure 31: Triplex secondary actuator concept

autopilot actuators, as used in a fail operational autoland system[10] on the 367-80 experimental 707 airplane.

The main purpose for the triplex secondary actuators is to take full advantage of the existing redundant computers, electronics and main power control units (PCU), making the elevator control fail operational/ fail safe from FBW and ACT system sensors through the actuators. The failure detection, isolation, and compensation techniques in the actuation stage are unique to that portion of the system. The flight validation of completely triply redundant digital computers, interface electronics and actuation would be a positive step in the evolution of actuation redundancy management. Furthermore, the fail operational characteristic is required for at least two of the ACT concepts -- relaxed static stability and envelope limiting.

Other advantages accrue from incorporation of a redundant secondary actuator to receive electrical commands rather than the existing autopilot transfer valve. The autopilot valve in the elevator PCU and identical aileron PCU had inadequate flow (control surface rate) capability for the TCV program, which resulted in a sluggish manual electric control system and a roll limit cycle. The autopilot valve was modified to increase the flow, thereby increasing elevator rate capability. The rate is still marginal for additional functions. For instance, the modification only increased the no-load maximum elevator rate from seven deg/sec to eleven deg/sec in the cruise and climb flight conditions. The ratio of autopilot valve rate capability to main valve rate capability is now at a limit for safety. The secondary actuator will improve the maximum electrical command rate, assuring adequate capability for the addition of any of the ACT program options, 1-5. The redundancy would allow the safe increase of both surface rate and deflection limits.

The present autopilot valve flow is also quite nonlinear regarding direction of flow. When an electrical actuation mode is engaged, the main control valve is locked into a detente and only the autopilot valve flow moves the piston. This is a moving-body actuator, and the main valve crank pulls the mechanical control system with it. When the elevator deflection becomes too large, the force required to pull the mechanical control through the feel system causes the main valve to "cam out" of detente, allowing flow across the main valve and negating the autopilot valve flow. The cam-out is set to protect against electrical command failures, and should not occur until an electrical mode deflection limit is reached. However, the cam is "soft" and flow across the main valve occurs almost immediately when deflecting away from zero. Consequently, the maximum autopilot valve flow is considerably less when going away from zero. This phenomenon is all on the input side of the actuator, in addition to the ideal load/rate parabolic relationship. The effect of this nonlinearity on ACT systems has not been studied.

In summary, the autopilot valve is sufficient for its originally intended purpose, but is marginal for the integration of TCV and

ACT control systems.

7.1.2 Electrical and hydraulic power supplies. - Hydraulic flow demands can be satisfied by existing power supplies for any of the options, 1-5. The maximum flow estimated for any of the options is approximately $3.2 \times 10^{-4}$ $M^3$/sec (5 gal/min), required for either Option 1 (ride control/direct lift) or Option 5 (ride control via mode suppression). The existing cockpit switching must be modified to permit operation of the existing hydraulic SYANDBY system while both A and B are pressurized. This will be the third independent source for the pitch triplex secondary actuator.

The existing AC electrical busses will also carry the additional electrical load for any of the options.

7.1.3 Flight control computers and interface electronics. - The RSFS presently has triply redundant General Electric ICP-723 digital computers, which, with their auxiliary equipment, are called the Incremental Control Processor Subsystem (ICPS). The Boeing Commercial Airplane Company, who designed and modified the NASA 515 for the RSFS/TCV configuration, has recommended that the ICPS be replaced with a triplex General Electric Whole Word Computer Subsystem (WWCS), featuring MCP-703 computer units.

It is felt that the WWCS is required to demonstrate state-of-the-art digital computers that would be used in a commercial application of active control systems. The WWCS will be an improvement over the ICPS for the following reasons:

- WWCS has storage capability making possible preprogramming of such items as c.g. locations versus fuel load and flight envelopes versus altitude and flap settings.
- WWCS has increased computation capability. ICPS is limited to 128 algorithms.
- WWCS has superior self test and system test capabilities.
- WWCS has increased input/output capability.
- WWCS capacity is expandable by adding memory.
- WWCS has superior logical decision making capabilities.

Essentially 100 percent of the ICPS capability is utilized by the TCV program. TCV functions would have to be deleted to accommodate ACT functions if the ICPS were retained, and a method would probably have to be developed to re-load computer memory rapidly and reliably between flights.

The triplex configuration of the WWCS has not been previously installed in an airplane. However, it has been checked out in the laboratory. In addition, an MCP-703 computer unit has been installed and flight tested on NASA 515 for another application. Some flight experience has therefore been obtained.

The triplex WWCS and interface electronics have been incorporated in the design for all options, 1-5. Analog sensor inputs are scaled, biased, and buffered in the Flight Control Interface (FCI) Pallet. These are then input to the Control Interface Unit (CIU), where A/D conversion occurs. Digital sensors are input directly to the CIU for D/D conditioning. Sensor selection and failure monitoring also occurs in the CIU. The sensor signals are voted by mid-value logic and are input serially to the MCP-703 computer unit. Control law computations are performed and the results are output serially to the Servo Transmitter/Receiver Units (STRU). The STRU performs the D/A conversion for the analog servo actuator command.

7.1.4 Control and display panels. - For Options 1-5, all control panels (mostly switches) are installed on the overhead panel in the aft flight deck. This panel now contains only "dummied" instruments.

In addition, most options require failure lights for channels, A, B and C in the FCI Pallet.

7.1.5 Design descriptions of ACT program options. - Design details are discussed in the following paragraphs that are peculiar to the respective options, 1-5.

7.1.5.1 Option 1 (ride control/direct lift): The RC/DLC system block diagram was shown in Figure 9, Paragraph 4.5.1. Performance predictions were computed using the flight control spoilers, numbers 2, 3, 6 and 7 (reference Figure 1), as direct lift surfaces. Subsequently a design decision was made to use the ground spoilers (1, 4, 5 and 8). The ground spoilers have the lift capability to achieve the predicted performance. Both pitch rate and acceleration feedback gains will require revision because of new spoiler derivatives.

Ground spoilers will become dedicated surfaces for the RC/DLC functions. This configuration is proposed to eliminate interfacing with the exceptionally complex spoiler roll control system, which would necessitate a long and costly actuator development program (reference Paragraph 5.1.5). It did not appear feasible to make the flight spoilers dedicated surfaces for RC/DLC, losing them for roll control. On landing, full wheel bank angle in one second would drop from 13.5 deg to 3.5 deg, which is marginal roll rate capability.

Ground spoilers 1 and 8 on the NASA 515 were originally operated as flight spoilers and will not require reinforcement to withstand flight loads. New spoiler assemblies will be required at positions

4 and 5, with heavier skins, doublers, hinge fittings, and actuator fittings. New hinge and actuator support fittings and skin doublers are also required on the wing trailing edge.

The four RC/DLC spoiler actuators will be simple, electrically commanded actuators with electrical feedback. Figure 26 (Paragraph 4.6.2) showed a maximum spoiler rate requirement of approximately 50 deg/sec at a turbulence intensity of 2.1 M/sec (7.0 ft/sec) rms. The spoilers will be biased partially upward to 7.5 degrees, with ±7.5 degrees deflection limits for ride control in the design condition (climb). The ACT spoilers will be mechanically limited to approximately 25 degrees for DLC in approach and electrically limited at approximately 15 degrees for RC at higher $\overline{q}$ conditions to minimize spoiler induced tail buffeting. The maximum hinge moment required for each RC/DLC actuator is approximately 1763 N-M (1300 ft-lb).

All elevator linkage and control functions for primary flight control from the forward flight deck will remain unchanged. The redundant secondary elevator actuator described in Paragraph 7.1.1 will be mounted on the bulkhead at BS 29.36 M (1156 in), with an appropriate mounting fitting and local reinforcement of the bulkhead. The secondary actuator will be controlled by electrical signals only, and will perform the elevator actuation required for aft flight deck FWB, DLC, and RC. It is estimated that the maximum elevator rate required for this option is approximately 15 deg/sec for either DLC on approach or RC at high-speed climb. The maximum elevator deflections for DLC and RC are approximately seven degrees for approach and 1.5 degrees for climb.

7.1.5.2 Option 2 (gust load alleviation - wing root): The GLA system block diagram was shown in Figure 11, Paragraph 4.5.2. The existing aileron will be split and the approximately 40 percent span outboard of the geared aileron tab will be utilized for GLA control.

The GLA actuation concept is illustrated in Figure 32. A new GLA actuator will be required on each outboard aileron. The new actuators will receive only electrical commands and will have electrical feedback. The GLA span of the aileron will be slaved to the inboard section for manual (or TCV) control as shown in Figure 32. The estimated GLA aileron actuation requirements are listed below for the design condition (climb):

| Parameter | Maximum required |
|---|---|
| Aileron rate | 125 deg/sec |
| Aileron deflection | ±9 deg |
| Hinge moment | 1017 N-M (750 ft-lb) |

New hydraulic plumbing runs will be required to supply power to the GLA actuators.

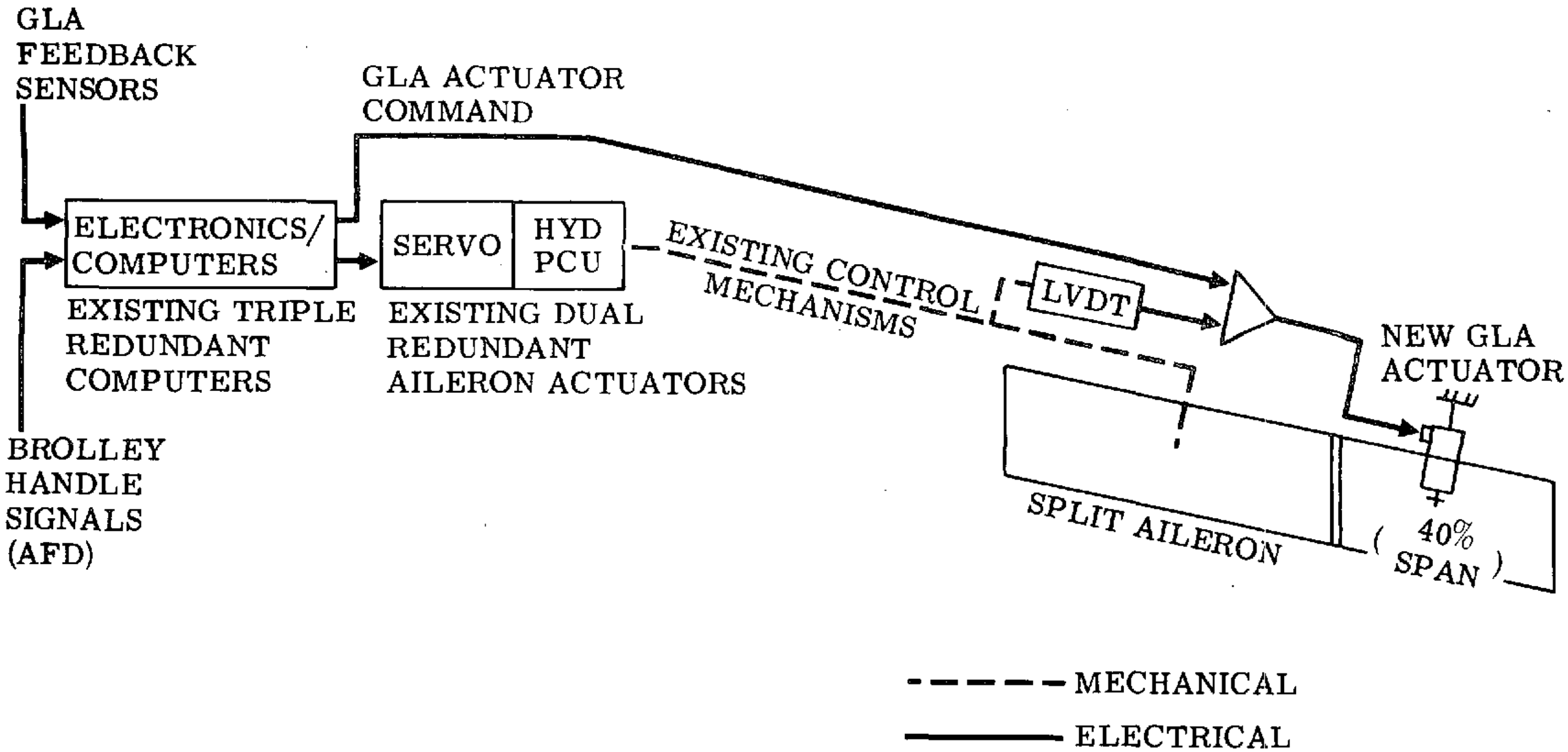



Figure 32: Implementation concept - wing gust load alleviation

New aileron assemblies will be fabricated with the split between the two segments at WBL 12.20 M (480.25 in). A new aileron hinge fitting at the outboard end of the inboard aileron, WBL 12.08 M (475.6 in), new end ribs, balance panels and seals will be required. A new hinge, bell crank, and actuator fitting will be added at the center of the outboard (GLA) ailerons.

The wing trailing edge will be modified by adding an aileron support rib for the inboard GLA aileron hinge, and a double rib to support the new GLA aileron actuator. All aileron segments will be statically balanced.

The elevator actuator installation is the same as for Option 1, described in Paragraphs 7.1.1 and 7.1.5.1.

7.1.5.3 Option 3 (envelope limiting): Figure 33 shows a typical maneuver flight envelope, taken from a 737 design loads document. The envelope is bounded by aerodynamic stall at the lower airspeeds, and then by normal acceleration and airspeed structural limits. The envelope varies with flap position and altitude and at higher altitudes there will be a structural Mach number limit.

An envelope limiting concept is suggested in which the angle of attack, normal acceleration, equivalent airspeed and Mach number are automatically limited according to prescribed scheduled envelopes.

Figure 34 is a conceptual block diagram of the structural envelope limiting. Logic is performed in the digital computers to detect the exceedance of a Mach number or equivalent airspeed limit, and the speed is controlled by throttle servos. Similarly, normal acceleration is controlled by the elevator. An algorithm is suggested using true airspeed and pitch rate to predict when exceedance of a structural limit is imminent and to actuate controls in time to prevent overshoot. Figure 35 shows a similar concept for limiting the airplane to speeds above aerodynamic stall (probably above buffet), using first the elevator and then the throttles.

Existing throttle servos will be used, and the elevator actuator installation is the same as for Option 1, described in Paragraphs 7.1.1 and 7.1.5.1.

7.1.5.4 Option 4 (relaxed static stability/c.g. control): The system block diagram of the SAS for RSS was shown in Figure 16, Paragraph 4.5.3. Only elevator actuation is required, and the actuator installation is the same as for Option 1, described in Paragraphs 7.1.1 and 7.1.5.1. The actuator requirements for the pitch SAS are moderate. At the worst case, i.e., with the c.g. at 10.3 percent M.A.C., in the cruise condition, the maximum elevator deflection and rate are approximately 1.0 deg and 10 deg/sec, respectively.

The c.g. will be controlled by forward and aft water tanks, conceptually illustrated in Figure 36. The fuel load affects the c.g. and will be used in the computations, but only the water will be used

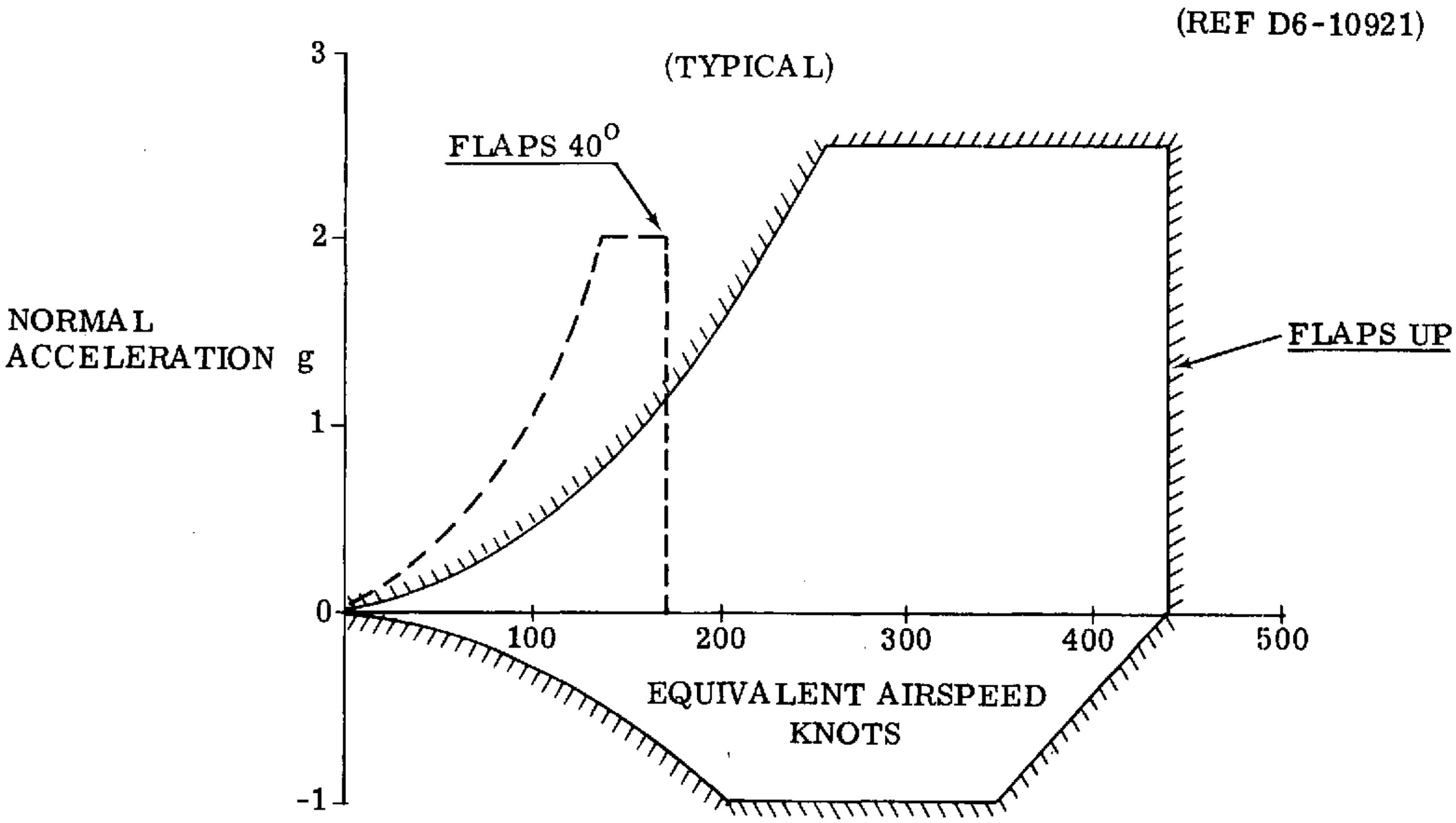



Figure 33: Typical 737 maneuver flight envelope

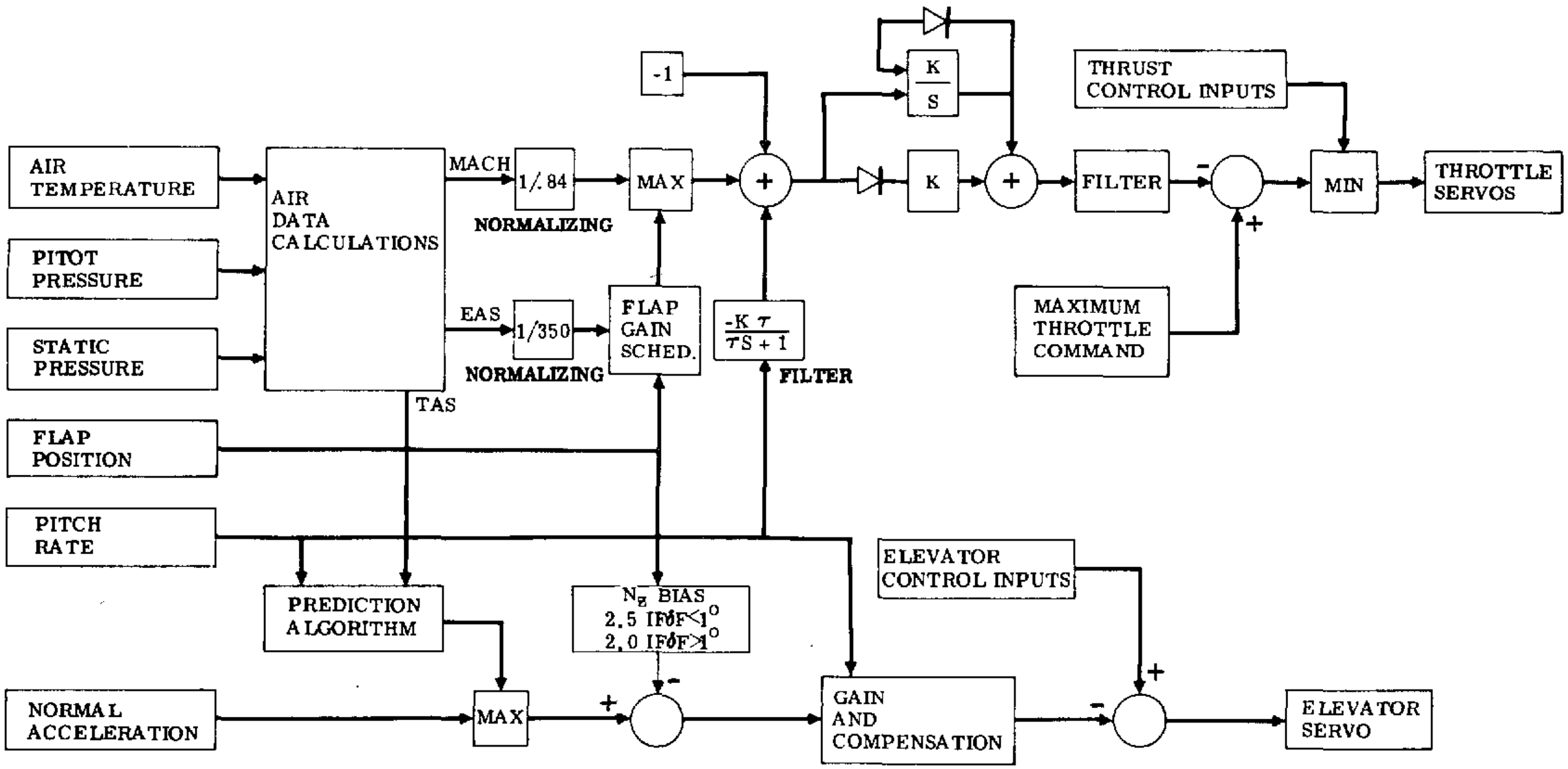



**Figure 34: Structural envelope limiting concept**

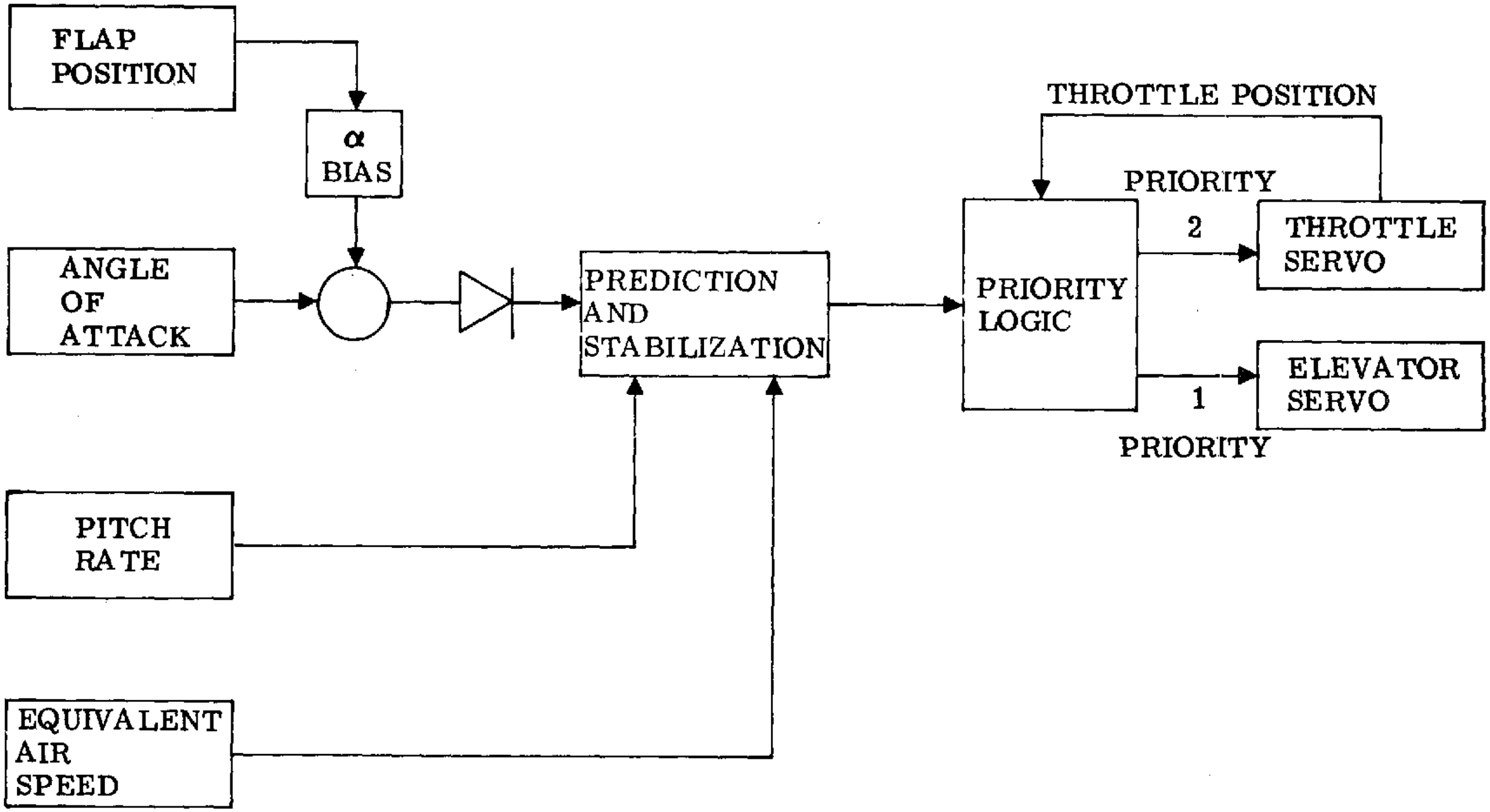
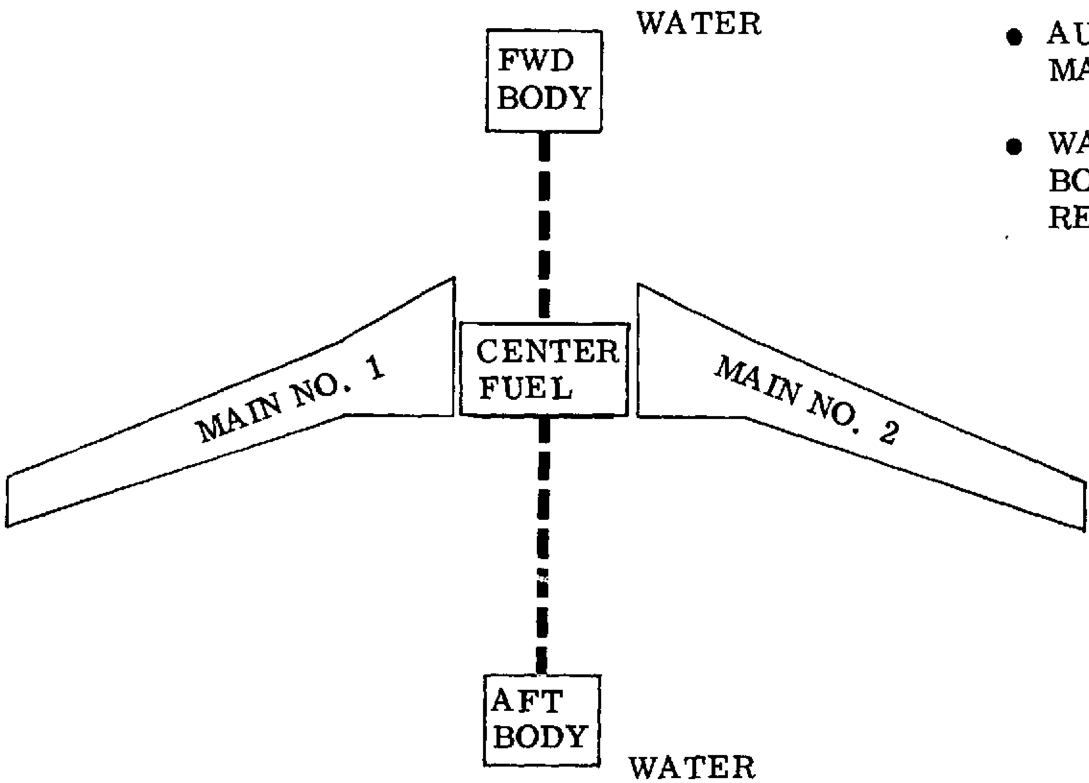



**Figure 35: Aerodynamic stall limiting concept**




- ONLY WATER USED TO CONTROL C.G. ~ 22 241 N (5000 lb) TOTAL
- AUTOMATIC C.G. CONTROL WITH MANUAL OVERRIDE
- WATER QUANTITY PROBE IN AFT BODY TANK ONLY (FLOAT-RESISTANCE TYPE)

**Figure 36:** C.G. control concept

for control. A total of 22 241 N (500 lbs) of water is required, and tank capacities must be adequate to shift the total amount forward or aft. The c.g. will be automatically controlled to a commanded location. A manual override will be provided to cause flow in either direction. A water quantity probe will be placed in the aft tank only, and the forward tank quantity will be computed.

Figure 37 shows the water tank and plumbing arrangement. Two production fuel range tanks will be used in the aft end of the forward baggage compartment. Two of the same tanks will be installed at the aft end of the passenger deck, one on each side of the aisle, and rotated 90 degrees to the normal orientation, as shown. An interconnecting line will be installed under the two tanks, across the ceiling of the aft baggage compartment. The floor beams will be reinforced to support the tanks.

A preliminary water transfer system design is illustrated in Figure 38. The diagram shows dual pumps (727 centrifugal fuel pumps proposed) in each tank, utilizing existing fuel tank plumbing. The dual pumps will provide 0.005 $M^3$/sec (40 000 lb/hour) of flow. In addition, manual overboard dump valves are provided. Both dual pumps and manual dump valves are not required for the single-fail-operate criterion specified in Paragraph 6.1.2. Single pumps could be used by plugging the appropriate existing plumbing fixtures in the fuel tanks.

Figure 39 shows the automatic c.g. control and indication concept. Signals are taken from existing fuel quantity indicators, conditioned and converted from analog to digital. This information, plus the aft tank water quantity, ground weighing and special loading data, along with fuel and water moment arms stored as a function of quantities, are used in the computation of c.g. An error between the selected c.g. and computed c.g. will generate a signal to drive the pumps. The c.g. location and aft body water quantity are indicated as shown. A total of 26 689 N (6000 lbs) of static ballast (lead ingots) will be spread on a pallet across the floor of the aft baggage compartment, so that the c.g. of the static ballast is at approximately BS 22.91 M (902 in). The compartment floor will be reinforced to support the load.

In addition to the on/off switches and warning lights, water transfer override switches (forward and aft) and c.g. selector and indicator will be located on the aft flight deck overhead panel.

7.1.5.5 Option 5 (ride control via modal suppression): The block diagram for the RC system to suppress structural mode acceleration in the forward body was shown in Figure 18. The system will be implemented by adding a pair of forward body horizontal canards with an actuator for each canard. The canards will be located at a bulkhead/flight deck intersection at approximately BS 5.69 M (224 in) and WL 5.08 M (200 in). Additional analyses will be required to ensure that the vortex from the canards will not adversely affect wing aerodynamic characteristics.

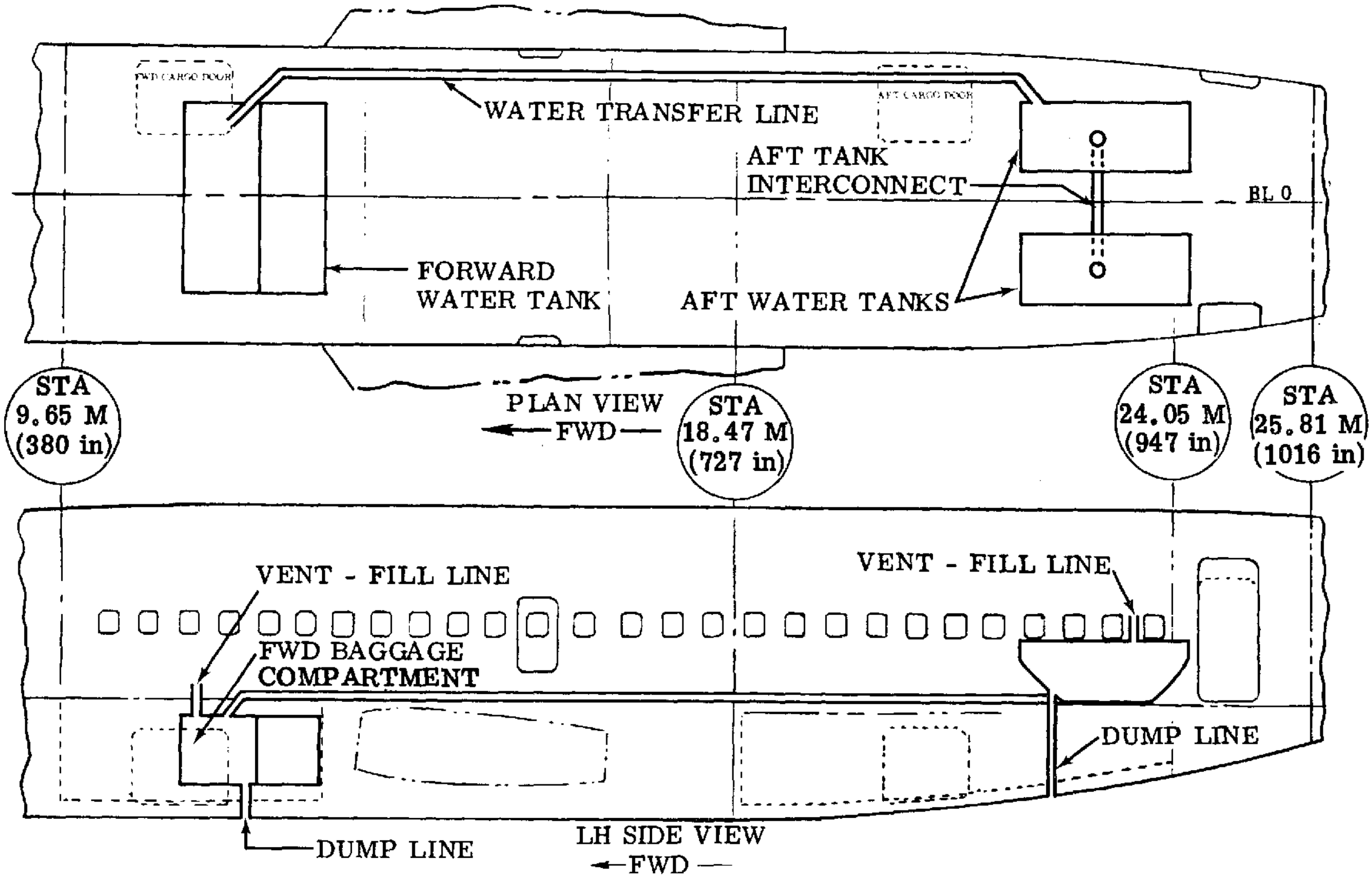



**Figure 37: Water tank and plumbing arrangement**

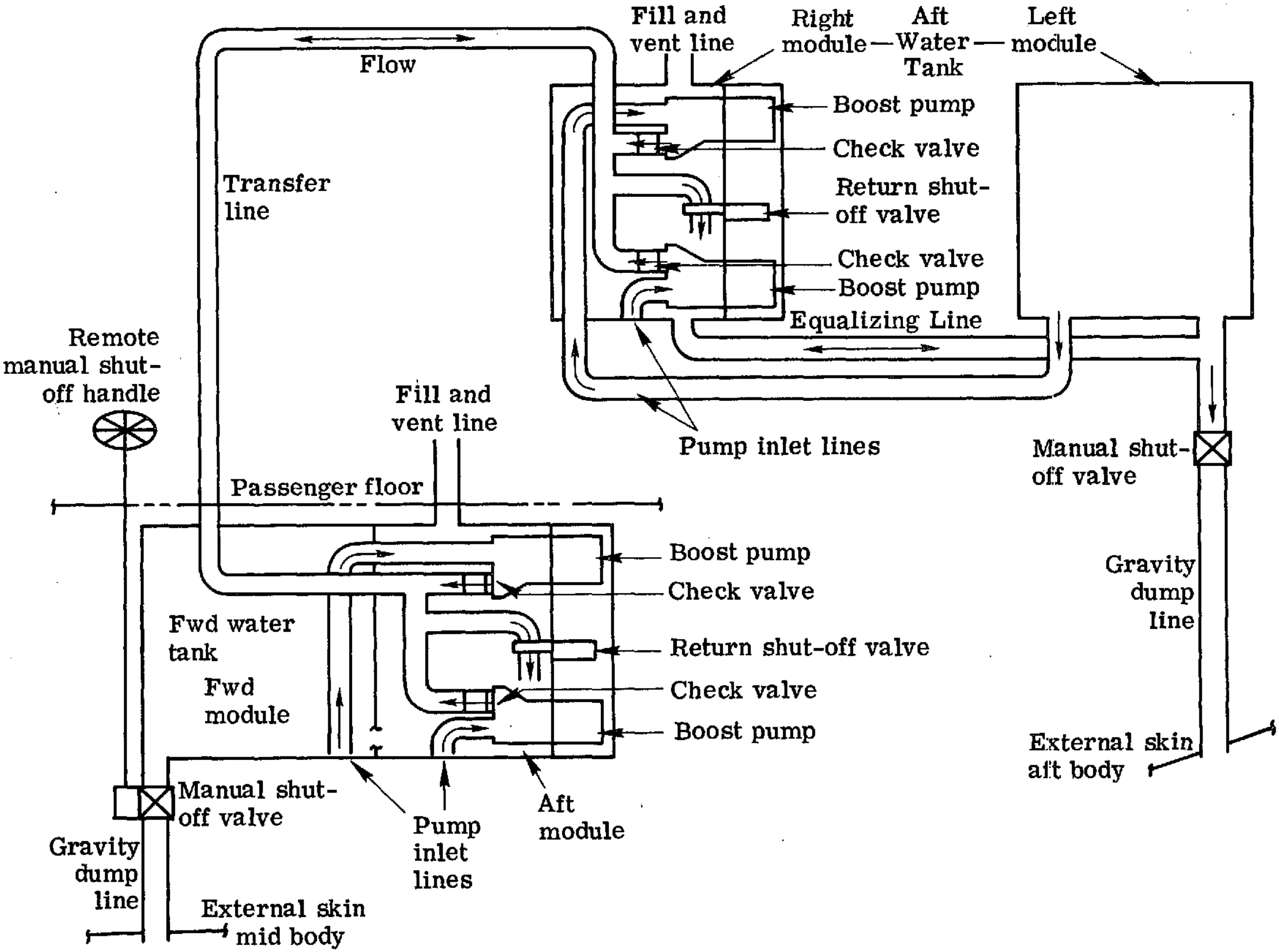
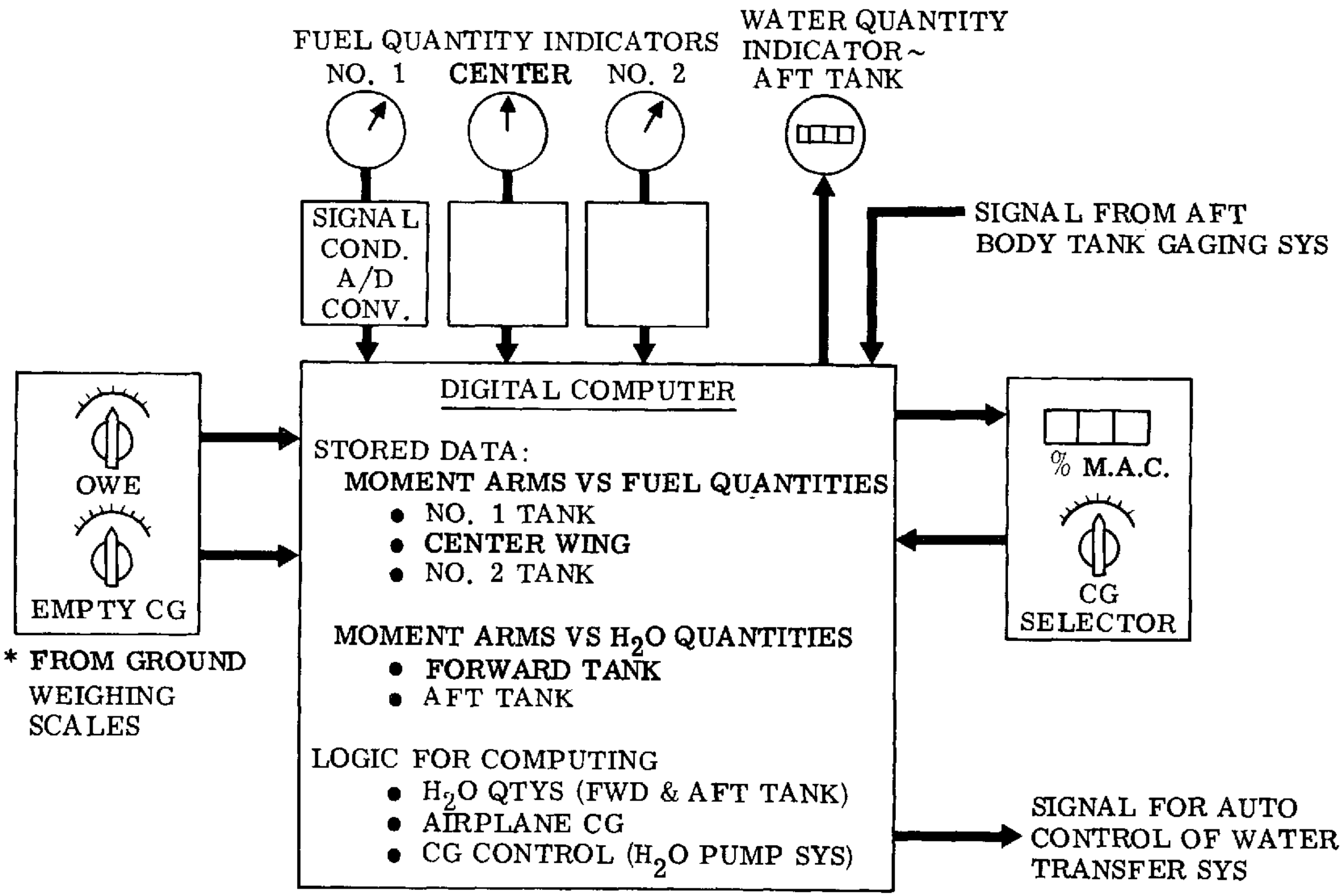



Figure 38: Water tank and transfer system diagram




*DIAL-IN ANY CHANGE TO GROSS WEIGHT & CG SUBSEQUENT TO WEIGHING

Figure 39: C.G. indication and automatic control concept

The fuselage will be modified to add canard bearing fittings, actuator support fittings, and structural reinforcement required to distribute canard loads into the airframe.

The canards will have approximately 0.93 $M^2$ (10 $ft^2$) of area. The construction will be of aluminum sheet metal ribs and skin with an extruded aluminum spar. A fairing will be fabricated and installed between the canards and fuselage.

The canard actuators will be simplex actuators that receive only electrical signals, and will have electrical feedback. The maximum deflection and rate required at a turbulence intensity of 2.1 M/sec (7.0 ft/sec) are approximately 5 deg and 100 deg/sec, respectively. The actuators will be required to act against 1763 N-M (1300 ft-lb) of hinge moment. Hydraulic plumbing runs will be added from the wheel well area to the canard actuators.

The elevator actuator installation is the same as for Option 1, described in Paragraphs 7.1.1 and 7.1.5.1.

## 7.2 Option 6 (Phase II)

The purpose and scope of Phase II are repeated below for emphasis:

General purpose: To develop and flight validate FBW/ACT digital implementation for large commercial aircraft.

Specific goals:

- To develop hardware and software design techniques and criteria
- To flight validate:
  - design guidelines
  - performance over the full flight envelope
  - system compatibility
  - failure immunity
  - implementation techniques

The scope of Phase II (Option 6) will include implementation of all the concepts from Options 1 through 5, plus maneuver load control. Preliminary design and program planning are accomplished assuming no prior implementation of any option.

7.2.1 General design description. - It is assumed that quadruply redundant flight control systems are required to meet the safety criteria for critical FBW/ACT systems. This includes sensors, computers, electronics, actuators, electrical power supplies, and hydraulic power supplies. Figure 40 is a general functional block diagram of this redundancy (excluding power supplies).

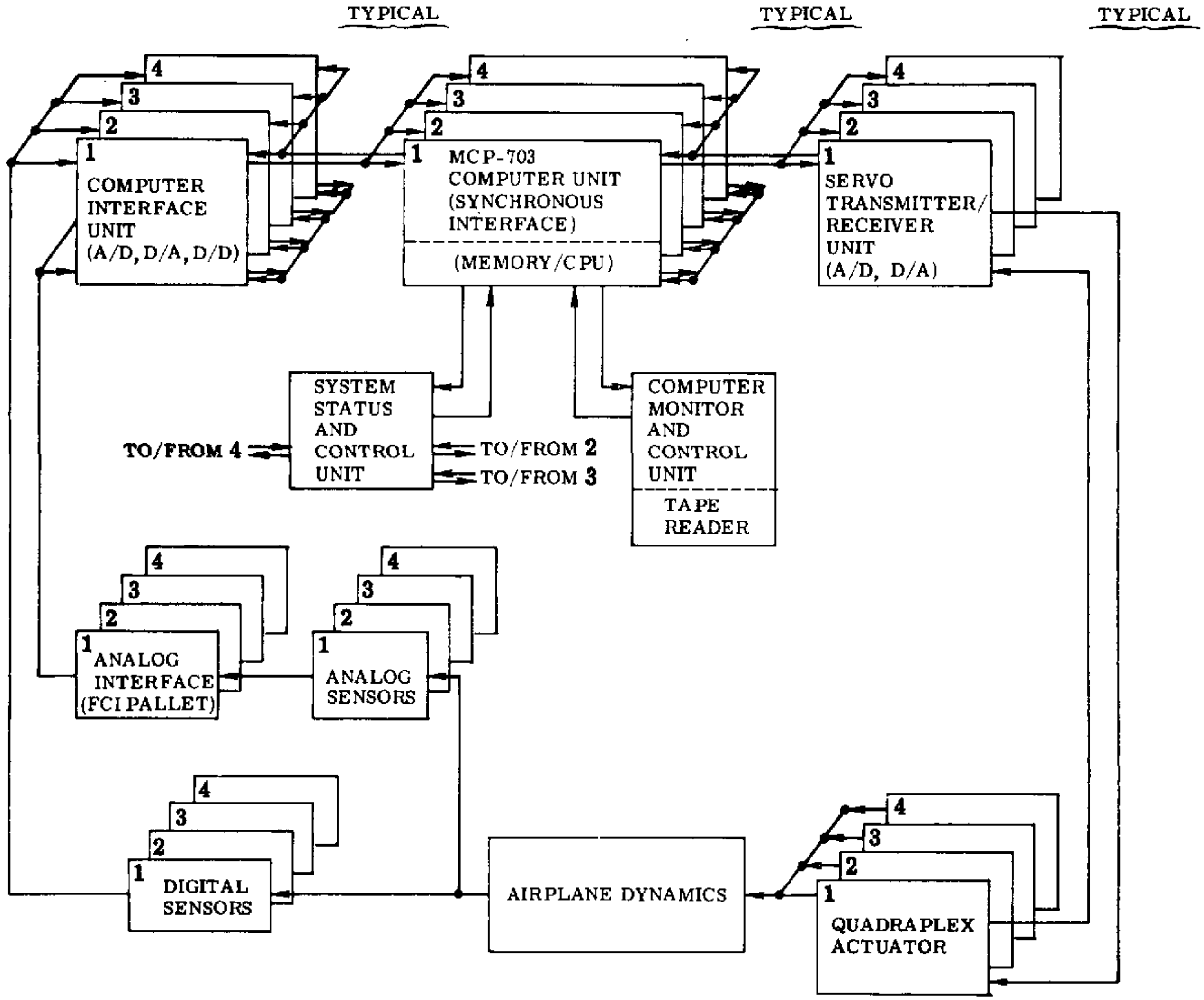



**Figure 40: General ACT system redundancy - Option 6**

A FBW control system will be designed using the forward flight deck left hand controls. The system will be implemented so that the design guidelines and implementation, the performance, and failure immunity are fully validated for an all-FBW control system. The existing mechanical control system will be disconnected from the left hand controls and modified as required to retain mechanical control from the forward flight deck right hand seat for the primary controls. Spoiler control will be all-FBW. The mechanical control is retained to preserve the airplane for other uses and to provide back-to-back comparison of FBW control with the mechanical control of a modern commercial transport under identical conditions.

Biased spoilers will be used as the direct lift surfaces for ride control, direct lift control, and maneuver load control. Modern commercial transports typically have multi-slotted Fowler trailing edge flaps for high lift capability. A viable flaperon configuration for commercial usage which would not degrade the performance of the multi-slotted Fowler flaps has not been developed. A technique is required for articulating the aft segment (flaperon) of this kind of flap in both the retracted and extended positions. This would require an extensive design and development program that would place an inordinate emphasis on that element of the program relative to FBW/ACT per se. Flight validation of a flaperon that is not a viable commercial configuration has no more value than using the biased spoilers, and would be much more costly. The biased spoilers are therefore used as an alternative to demonstrate the concepts, validate analysis, handling qualities, and compatibility with other systems.

The flight control spoilers will be used for direct lift in Option 6 instead of the ground spoilers, which are proposed in Option 1. The ACT functions are easily summed into the all-FBW flight control actuators, and the problem of interface with the mechanical roll control system does not exist. Furthermore, this would be the typical commercial configuration.

More specific design descriptions are contained in subsequent paragraphs.

7.2.2 Actuator requirements. - The actuation redundancy proposed to satisfy the failure philosophies for Option 6 is shown in Figure 41. The elevator and rudder controls feature quadraplex secondary actuators that have mechanical outputs to redundant power control units, with parallel motions of the controls at the right hand seat of the FFD. The same concept cannot be utilized for the ailerons. Summation of symmetric and antisymmetric commands results in unequal motion of the l.h and r.h. ailerons, and they cannot be tied to the same mechanical system. An integrated actuator (electrical and mechanical commands to same package) is required, with series inputs relative to the mechanical backup.

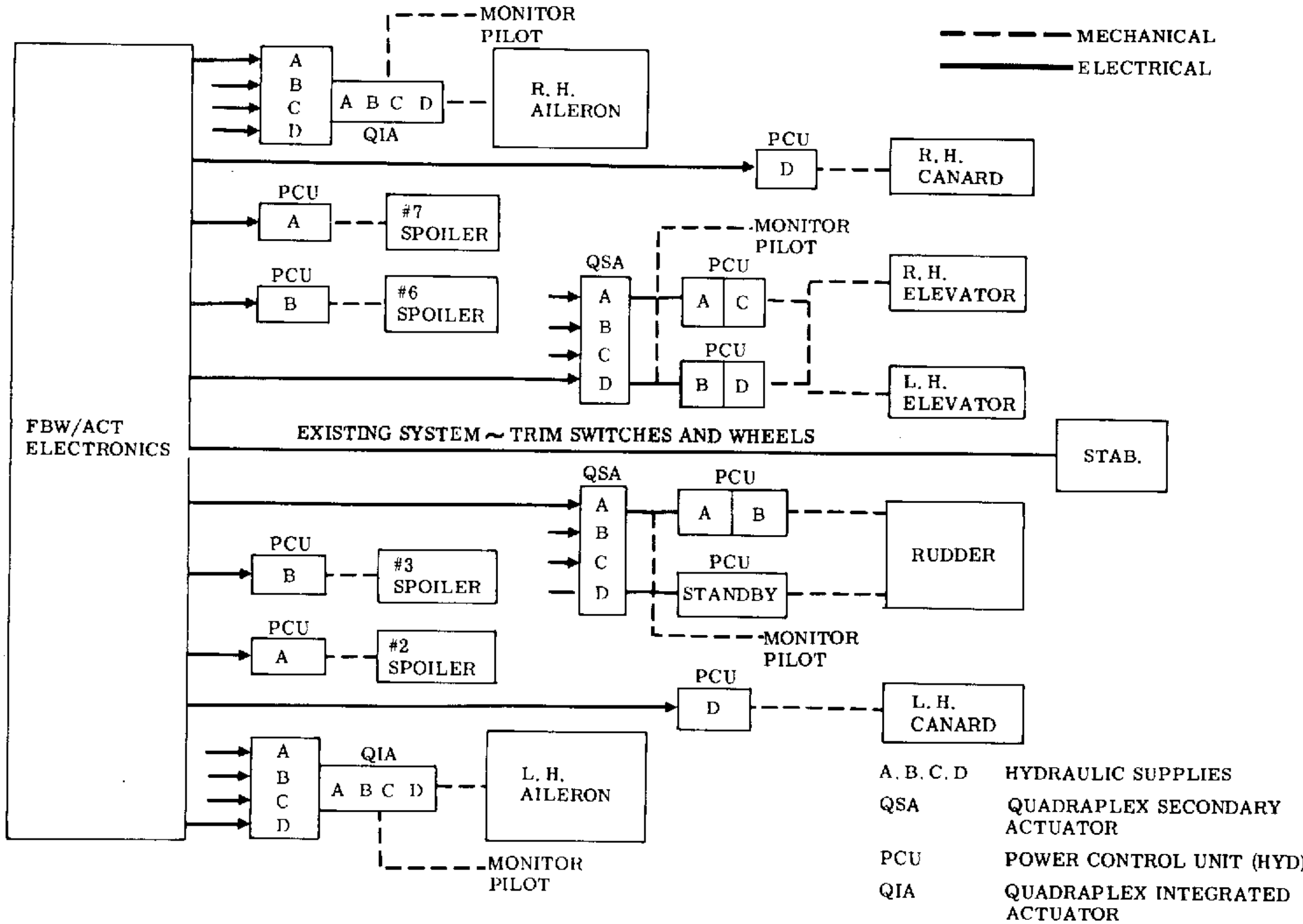



**Figure 41: Actuation redundancy concept - Option 6**

The canards have simplex PCU's that can be compared with each other, with an appropriate sign change, to satisfy the fail passive criterion for ride control.

The flight control spoiler actuators are also simplex because of the roll control redundancy between the ailerons and spoilers and also the redundancy among the spoilers themselves. In addition, the ACT functions implemented with spoilers are non-critical.

Table X presents the maximum requirements per actuator for Option 6, including deflection, rate, torque, and bandpass frequency. All of the values are practicable.

TABLE X

FBW/ACT ACTUATOR MAXIMUM REQUIREMENTS - OPTION 6

| Control Surface | Maximum deflection deg | Maximum rate deg/sec | Torque N-M (ft-lb) | Bandpass frequency rad/sec |
|---|---|---|---|---|
| Elevator | 21 | 35 | 2495 (1840) | 20 |
| Aileron | 20 | 80 | 4067 (3000) | 30 |
| Spoiler | +40<br>- 0 | 60 | Ext. 2847 (2100)<br>Ret. 1762 (1300) | 20 |
| Canard | 5 | 100 | 1762 (1300) | 60 |
| Rudder | Existing | | | |

7.2.3 Hydraulic power supplies. - The proposed hydraulic power supply system is composed of four active systems and one standby system. A schematic of the general relationship is shown in Figure 42. Following is a description of each of the proposed subsystems:

System "A": System "A" will consist of the existing right-hand engine driven pump and the following equipment which must be added (similar to the existing "A" system):

1) Pump pressure module
2) Reservoir
3) Heat exchanger (fuel/oil)
4) Case drain filter
5) Return module

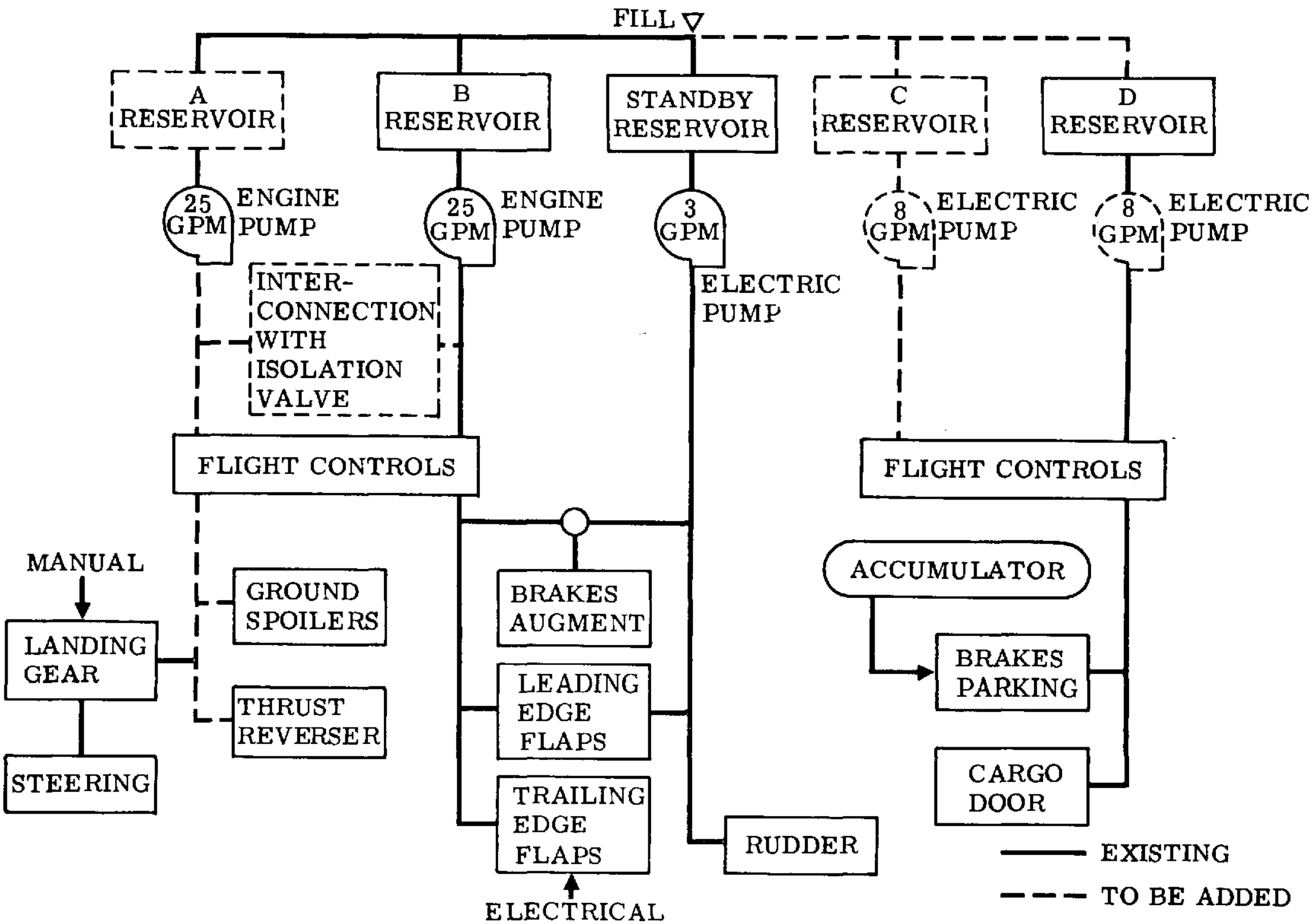



**Figure 42: Hydraulic power redundancy concept - Option 6**

6) Associated valves and system plumbing

This system will provide $5 \times 10^{-4}$ $M^3$/sec (8 gal/min) to flight controls and power to the landing gear and steering system, ground spoilers and thrust reversers. Reservoir level and pressure indication will be added in the cockpit.

System "B": System "B" will consist of the existing left-hand engine driven pump and all utility equipment, except that landing gear, steering and ground spoiler requirements will be removed. An isolation valve will be installed as shown. Control for this valve will be added in the cockpit.

System "C": System "C" will be a new system consisting of equipment similar to the existing "B" system.

1) $5 \times 10^{-4}$ $M^3$/sec (8 gal/min) electric motor pump
2) Pressure module
3) Return module
4) Reservoir
5) Heat exchanger (fuel/oil)
6) Case drain filter
7) Associated valves and plumbing

System "C" will provide $5 \times 10^{-4}$ $M^3$/sec (8 gal/min) to the flight controls system. All new plumbing runs will be required.

System "D": System "D" will consist of the existing "B" system except the two $3.8 \times 10^{-4}$ $M^3$/sec (6 gal/min) pumps will be replaced with a single $5 \times 10^{-4}$ $M^3$/sec (8 gal/min) electric-motor driven pump. The system plumbing will be reconfigured as required for the new flight control installation.

Standby System: The existing standby system will remain unaltered.

In order to allow the "A" and "B" systems to interconnect, an isolation valve is proposed. The isolation valve is presently used on the 707 tanker airplane.

In the normal "worst case" conditions for systems "A" and "B" the flow available for system "B" is slightly below the demand when flight control, TE and LE flap demand is simultaneously maximized. The pumps which supply fluid are existing ABEX engine driven pumps with a displacement of $2.9 \times 10^{-5}$ $M^3$/revolution (1.77 $in^3$ revolution). Higher flows would be availabe with a modified pump displacing $3.3 \times 10^{-5}$ $M^3$/revolution (2.0 $in^3$/revolution).

In case of demand approaching a "worst case" condition, priority valves would cut off flow to the secondary devices and all fluid would be required to satisfy flight control requirements first.

7.2.4 Electrical power supplies. - The NASA 515 airplane is equipped with three independent sources of power. This was accomplished by altering the APU control circuit to allow the APU generator to power a separate A.C. bus. A need for a fourth independent A.C. bus capable of providing 300 watts of power has been introduced on the NASA 515 airplane, to supply the load requirements of the proposed Phase II flight critical equipment. This could be accomplished by using a transformer-rectifier powered from a main A.C. bus and a lead acid battery. Both sources will supply a 500 VA static inverter to provide a fourth single phase A.C. bus as shown in Figure 43.

A panel would be fabricated to house the transformer-rectifier unit and the static inverter. The location of this panel would depend upon space availability on existing pallets. The lead acid battery would be installed in a pressurized portion of the airplane, and a sump jar installed to collect any spillage. Also, an overboard vent for removing toxic fumes would be added. The suggested location for the battery is in the forward cargo area, as there is no additional space in the electrical/electronic bay.

7.2.5 Design description of the primary FBW and mechanical controls. - The primary FBW and mechanical pilot control configurations are described in the following subparagraphs.

7.2.5.1 Elevator: The left hand pilot's control column will be disconnected from the bus torque tube and will be independently mounted. A quadraplex position transducer and a hydra-mechanical feel unit will be installed below the flight deck and suitably connected to the pilot's control column. A worm and gear drive unit will be connected to the stabilizer trim manual control mechanism to provide a mechanical input representative of stabilizer position to the elevator feel unit. This mechanical input will change the elevator feel unit neutral position as stabilizer attitude changes, thus producing the programmed relationship between elevator position and stabilizer position that exists on unmodified 737 airplanes.

An elevator bus torque tube will be installed on the bulkhead at BS 29.36 M (1156 in) and it will be connected to each elevator by the existing bellcranks and pushrods. Dual tandem primary power actuators (2) which respond only to mechanical signals will be installed to replace the existing power actuators with their output shafts attached to bellcranks on the bus torque tube. The control input torque tube will be modified to provide control input motion to the dual tandem primary actuators and to accept output motion from a quadraplex secondary actuator which will be located above and to the left of the existing elevator feel unit. The quadraplex secondary actuator will have four transfer valves controlled by four separate electronic signal channels and will be powered by four separate hydraulic systems.

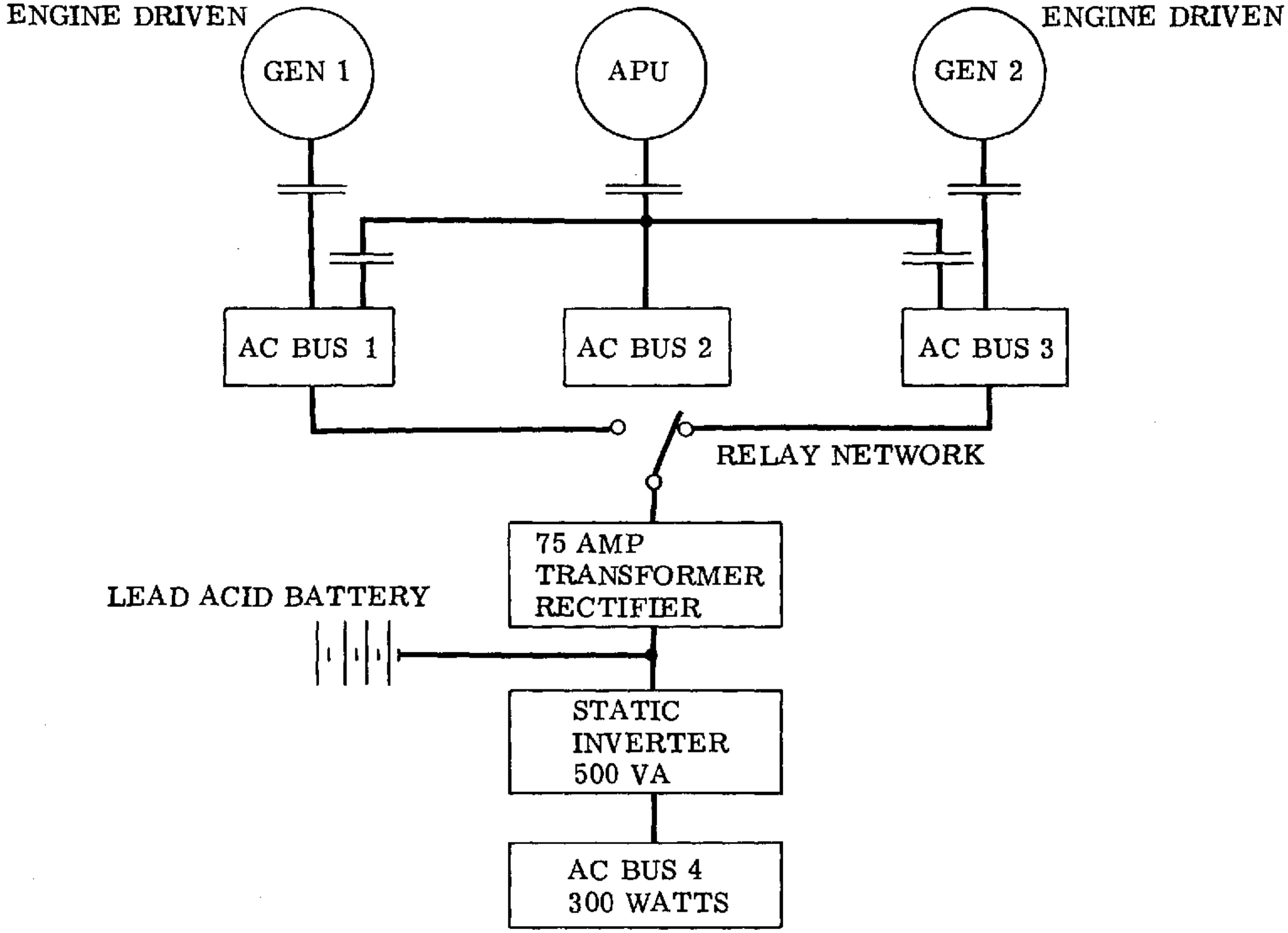



Figure 43: Electrical power redundancy concept - Option 6

7.2.5.2 Ailerons: The left hand pilot's control wheel will be disconnected from the aileron bus cables and the aileron control cables at the base of the control column. A quadraplex position transducer and a mechanical feel unit will be connected to the pilot's aileron control linkage at this location. An electro-mechanical linear actuator will be connected to the feel unit so that it can change the feel unit neutral position for aileron trim.

A quadraplex integrated actuator (described in Paragraph 7.2.2) will be installed at each aileron. The mechanical input will receive commands from the right hand wheel via the existing cable runs. The electrical input will receive commands from the FBW/ACT systems.

7.2.5.3 Rudder: The left hand pilot's rudder pedals will be disconnected from the rudder pedal bus linkage. A quadraplex position transducer and a hydra-mechanical feel unit will be connected to the pilot's rudder pedals. An electro-mechanical linear actuator will be connected to the feel unit so that it can change the feel unit neutral position for rudder trim. Two dual tandem primary power actuators will be installed in the positions occupied by the existing primary and standby actuators. A quadraplex secondary actuator identical to the elevator secondary actuator will be connected to the aft torque tube assembly.

7.2.6 Structural modifications. - Attach fittings for the elevator secondary actuator will be fabricated and installed on the bulkhead at BS 29.36 M (1156 in), and the bulkhead will be locally reinforced.

Water tanks and ballast will be installed the same as described for Option 4 in Paragraph 7.1.5.4.

Horizontal canard surfaces and actuator attach fittings will be installed the same as described for Option 5 in Paragraph 7.1.5.5.

# 8.0 NASA 515 ACT FLIGHT VALIDATION PROGRAM COST AND SCHEDULE ESTIMATES

For the purpose of estimating program costs it was assumed that Boeing will accomplish the following tasks in any of the programs:

- system synthesis and analysis
- system and structural design
- component laboratory tests
- system structural modifications
- airplane instrumentation
- system and structural airplane ground tests
- functional flight checkouts
  (two for Options 1-5 and four for Option 6)
- ferry flights
- onsite support at NASA

Each organization involved in the listed program tasks developed work statements, material lists and manhour estimates required to accomplish the tasks. The following guidelines were used"

- Each option is estimated separately, without assuming prior or simultaneous implementation of any option.
- All estimates are in 1975 year-end dollars.
- NASA will furnish NASA 515 airplane and a crew chief for normal service items while at Boeing (not ACT system modification).
- NASA will conduct flight tests for data collection and demonstration at LRC.
- The airplane will not be restored after flight tests.
- It is assumed that the triply redundant G.E. whole-word computers are installed and operational on NASA 515 at contract go-ahead.
- Piloted simulations are required for Options 1, 2, 3, 4 and 6, and will be conducted on the existing Boeing-Seattle 737 cockpit simulator.
- Engineering drawings will be of experimental quality and quantity.
- Hardware will be functionally representative of production hardware, but may be manufactured by hand assembly, machining or processing rather than production techniques.

Program budget planning costs were estimated from the manhour estimates and material lists. The costs are listed in Table XI, below. The program durations indicated are discussed in succeeding paragraphs.

TABLE XI

NASA 515 ACT PROGRAM OPTION SUMMARY

| Option | ACT Concept | Cost/Duration | Priority |
|---|---|---|---|
| 6 | All ACT/FBW | \$8.9M/36 mos. | 1 |
| 1 | Ride Control/Direct Lift | \$2.0M/17 mos. | 2 |
| 2 | Gust Load Alleviation | \$2.7M/17 mos. | 3 |
| 4 | Relaxed Static Stability | \$2.3M/17 mos. | 4 |
| 3 | Envelope Limiting | \$1.5M/17 mos. | 5 |
| 5 | Ride Control (Modal Suppression) | \$2.4M/17 mos. | 6 |

The options are listed in order of Boeing's evaluation of cost effectiveness; i.e., the technology advancement per dollar. Option 6 not only satisfies technology deficiencies addressed by Options 1-5, but demonstrates compatibility among the ACT systems, over the full envelope. Option 6 also provides significant advancement in the development of digital/FBW hardware and software implementation techniques. The estimated cost of Option 6 is \$8.9 million.

The estimated costs of Options 1-5 range from \$1.5 million to \$2.7 million. It is estimated that a 30 to 50 percent saving could be realized by combining several of the concepts in a single option.

It was found that any one of the first five options would require approximately the same duration, about 17 months, to complete functional flight tests and deliver the airplane to NASA. Figure 44 shows the typical schedule for Options 1-5. The critical path is actuator design and procurement. The airplane down time would be 3-½ months for modification, ground tests, and functional flight tests.

Figure 45 is a suggested schedule for Option 6, requiring 36 months through flight checkout and delivery. The critical path is computer procurement, software development and laboratory tests. The duration required for airplane modification, ground tests and functional flight tests would be approximately seven months.

These schedules were developed to show the least amount of time required. It may be advantageous to lengthen the analysis schedules, and perhaps specification of long lead time hardware and materials, to interface efficiently with other programs on the NASA 515. This work could be accomplished in an expanded schedule for the same cost.

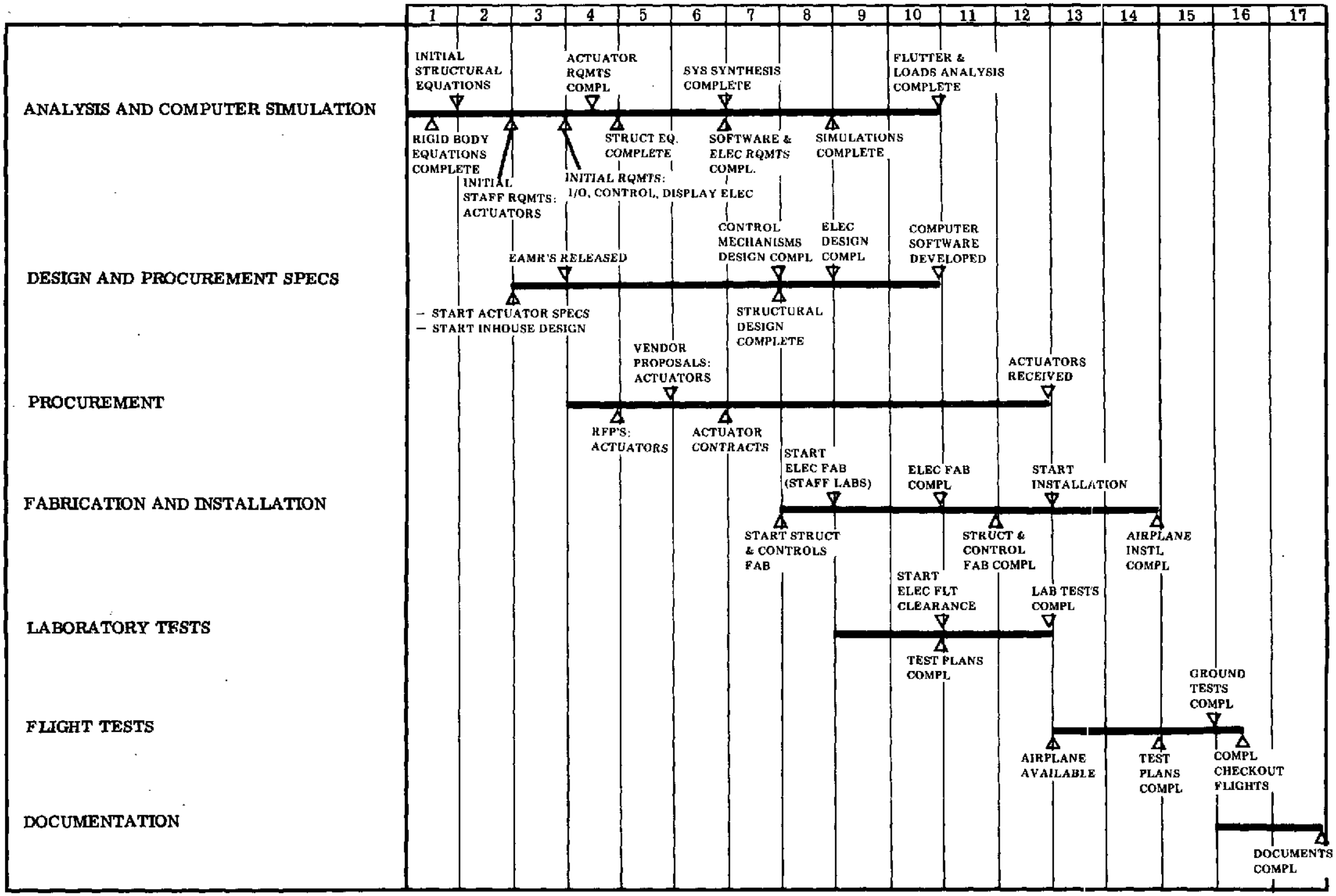



**Figure 44: Typical schedule - NASA 515 ACT program options 1-5**

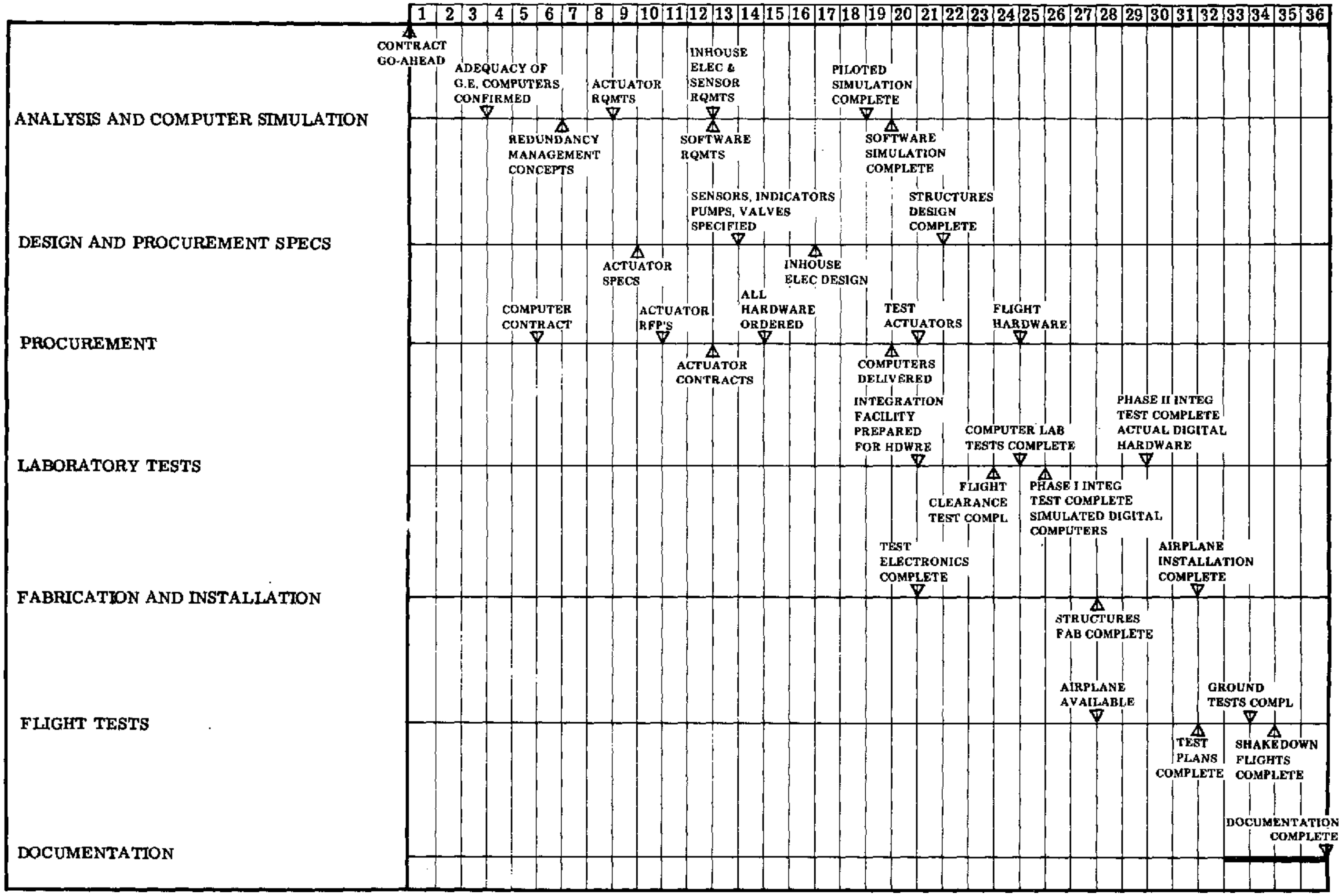



**Figure 45: Suggested schedule for NASA 515 program Option 6**

# 9.0 CONCLUSIONS AND RECOMMENDATIONS

## 9.1 CONCLUSIONS

The greatest need for flight research in the area of FBW/ACT development for commercial applications is to develop techniques for digital implementation of FBW/ACT for large commercial transports and flight verify that the techniques are safe, reliable, and cost effective.

The NASA 515 airplane is an excellent test bed with which to evaluate digital system performance at a minimum cost. The airplane will produce credible results as a modern commercial transport. It has already been modified to a commercial type research vehicle for flight navigation and controls testing. The airplane is equipped for research in digital flight controls, with palletized triply redundant digital computers and interface electronics. It can consequently be modified at a minimum of cost to increase system redundancy for the development of safe and reliable FBW/ACT implementation techniques.

The feasibility of most of the ACT concepts has been demonstrated. However, none of them have been implemented digitally, although a number of advantages have been claimed for digital flight controls. The concepts that are most likely to present difficulties to digital techniques are the concepts involving structural mode suppression, because of the higher frequencies.

A few voids do exist in concept validation, such as ride control via direct lift and a full envelope limiting system. Beyond validating concepts, per se, some further validation of practicability is required; e.g., resulting handling qualities and pilot acceptance and the interaction of systems. In addition, flight validation over the full flight envelope is required for all ACT concepts.

A workable flaperon configuration for commercial usage has not been developed. A technique is required for articulating the aft segment (flaperon) of a multi-slotted Fowler flap in both the retracted and extended positions.

## 9.2 RECOMMENDATIONS

Major advancements in the development of active control systems and subsequent acceptance by industry and commercial airline users will require flight test validation of FBW/ACT. Flight test recommendations identified as a result of this study in order of priority are: (1) all ACT/digital FBW flight validation over the full flight envelope; (2) ride control/direct lift control concept validation; (3) gust load alleviation to validate digital system performance at structural mode frequencies; (4) relaxed static stability to validate handling qualities and range improvements; (5) envelope limiting concept validation; and (6) ride control (modal suppression) to validate digital system performance at structural mode frequencies.

Although flight test experimentation such as that recommended is a vital element in the overall active control technology development, flight testing alone does not provide a sufficient technology base. High risk areas which require further research include flutter suppression, digital control techniques for structural modes, the new envelope limiting concept, and the effects of hydraulic actuation system saturation on aircraft stability. In addition, a summary of the technical experience and knowledge obtained in past ACT research and production programs should be provided to practicing aircraft designers in the form of design criteria and guidelines. This information will assist the aircraft designer in obtaining maximum aircraft performance benefits by properly incorporating ACT concepts in the complex preliminary design process.

Several aircraft and wind tunnel model research programs have been conducted by NASA and the Air Force to demonstrate experimentally that active aerodynamic control surfaces can be used to suppress symmetrical flutter modes with relatively low frequency and mild onset of flutter characteristics. Although these studies provide an excellent baseline in the development of active flutter suppression technology, there is considerable additional research required to fully demonstrate system technical feasibility and performance and reduce the flight safety risk to an acceptable level. New technology research areas include control of modes with higher frequency, violent onset of flutter, deeper flutter penetration and asymmetrical characteristics. It is recommended that a wind tunnel model research program be conducted in these areas with a modified version of the LRC B-52 aeroelastic model. This model is an excellent candidate test model because of its easily modified flutter mode characteristics and the extensive data base and experience obtained during past wind tunnel test programs.

Almost all structural mode control systems demonstrated to date have used analog or continuous type hardware. Recent developments in the cost, size and computational speed of digital hardware have led to significant interest in exploiting the inherent advantages of increased accuracy, reliability, flexibility and adaptability. Although digital implementation techniques are currently being used in flight control systems it is primarily in the control of low frequency "rigid body" motion. Little experience is available on the unique problems of digital control of higher frequency structural modes. It is recommended that a laboratory and wind tunnel test program be conducted to develop a multiple channel flutter suppression system for demonstration in the wind tunnel on the B-52 aeroelastic model. This program should (1) study single channel problems associated with sampling rates, frequency fold over effects, computation cycle times and filter mechanization techniques, and (2) develop redundancy management concepts for multiple channel structural mode control systems. The performance of hardware and software developed and verified in a laboratory system should be validated in the "quasi real world" environment of a wind tunnel test.

An automatic envelope limiting concept was considered in this study as an ACT concept because previous studies have indicated potential flight safety and performance benefits. Although automatic envelope limiting systems have been implemented and flown on spacecraft, very little comparable work has been accomplished on aircraft. It is recommended that an analytical and piloted simulation study be conducted to provide a detailed system design for a piloted simulation evaluation of system performance.

Control system saturation is an area of concern when active control technology is used to provide stability for an aircraft with negative static margin and for unstable structural modes. The general approach in designing control systems for unstable aircraft is to oversize the system to prevent saturation. The result is a weight penalty that degrades system benefits. The two aspects of this problem that need to be addressed are design criteria and system mechanization techniques. Saturation criteria need to be established for active control systems when these systems are used to provide the aircraft stability required for safe flight. Secondly, mechanization techniques are required which provide high reliability with minimum effect on system weight.

If the maximum benefits of Active Controls Technology are to be realized in future aircraft preliminary designs, it is necessary that design criteria and guidelines and parametric design computer programs be developed. Incorporation of ACT in future aircraft designs will require an integrated approach consisting of active control technology in addition to the conventional structure/aerodynamics technologies. Design guidelines and criteria must be developed which assist the designers in this new approach to aircraft design. In addition, preliminary design computer programs should be developed to assist in determining important ACT design parameters and the relationship of these parameters to fundamental performance measures such as direct operating cost (DOC) and return-on-investment (ROI). This information will be vitally important in encouraging the commercial aircraft industry to use Active Controls Technology,

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