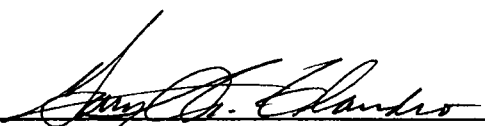


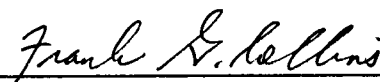
To the Graduate Council:

I am submitting herewith a dissertation written by Chris Barret entitled "Aerodynamic Flight Control to Provide Flight Control Augmentation for Aft CG Future Launch Vehicles." I have examined the final copy of this dissertation for form and content and recommend that it be accepted in partial fulfillment of the requirements for the degree of Doctor of Philosophy, with a major in aerospace engineering.

  
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**AERODYNAMIC FLIGHT CONTROL TO PROVIDE FLIGHT CONTROL  
AUGMENTATION FOR AFT CG FUTURE LAUNCH VEHICLES**

**A Dissertation  
Presented for the  
Doctor of Philosophy  
Degree  
The University of Tennessee, Knoxville**

**Christian Barret  
December 1996**



## DEDICATION

This dissertation is dedicated to

a friend who

directed my life,

Dr. Wernher von Braun

## ACKNOWLEDGMENTS

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## ABSTRACT

This document reports on the past six years of the extensive and comprehensive flight mechanics — stability and control, special research project conducted at the NASA Marshall Space Flight Center where the author was the designated Principal Investigator and only investigator. The author has proposed, designed, wind-tunnel tested, and analyzed the static and dynamic stability and control characteristics of both forward and aft located, all-movable, blunt-trailing-edge, flight control augmentors (FCA's) for future aft-cg large Saturn-class launch vehicles (LV's) that would not be controllable with engine gimbaling alone. To the author's knowledge, all-movable FCA's have never been used on a large Saturn-class LV.

A review of our national heritage of LV's that have used aerodynamic surfaces (V-2, Pershing, Redstone, Saturn I, Saturn IB, Saturn V, X-15) and current usage of these surfaces by other nations has been conducted. Stability derivatives and control derivatives have been calculated from both wind tunnel test programs data and the U.S. Air Force DATCOM computer code. The enhanced static and dynamic flight stability of the aft-cg experimental LV, when treated with the author-designed FCA's, has been assessed and quantified using static margin and short-period damping as criteria. The increased control effectiveness of the blunt-trailing-edge FCA design was found to provide the maximum flight control augmentation. A titanium matrix composite was selected as the FCA material and a design review was held at the manufacturer, Textron Specialty Materials Company. The change in aeroelastic effects of the aft-cg LV were considered. From the NASTRAN generated modal data, significant mode shapes and response spectra have been analyzed and presented for both the original and aft-cg experimental LV.

## PREFACE

The work reported herein was conducted by the author at the NASA Marshall Space Flight Center. This was a special research project commenced by the author in 1990, and formally proposed and accepted for research in 1992. The proposal was accepted for the NASA Marshall Space Flight Center, Center Director's Discretionary Fund (CDDF) Research Program. The special research project has been designated the special project code: CDDF 93-05 and the author was designated the Principal Investigator and only investigator.

During the course of this extensive and lengthy investigation, the author has submitted two patent disclosure packages, had two NASA Tech Briefs publications, published two NASA Technical Papers, made many conference presentations, mentored two students in research, and has been appointed to serve a 3-year term as the NASA Marshall Space Flight Center representative on the AIAA Atmospheric Flight Mechanics Technical Committee. All static and dynamic wind tunnel testing for this work was conducted in the 14" X 14" NASA Marshall Space Flight Center Trisonic Wind Tunnel. The work reported herein is that of the author. The author of this report is Chris Barret, Flight Mechanics and Control Branch, NASA Marshall Space Flight Center.

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## LIST OF SYMBOLS AND ACRONYMS

|               |                                |
|---------------|--------------------------------|
| $\alpha$      | angle of attack                |
| ac            | aerodynamic center             |
| A             | aspect ratio                   |
| b             | span of FCA                    |
| $\beta$       | angle of sideslip              |
| BC            | boundary conditions            |
| $\bar{c}$     | mean aerodynamic chord         |
| $C_A$         | coefficient of axial force     |
| CCC           | carbon-carbon composites       |
| cg            | center of gravity              |
| $C_l$         | coefficient of rolling moment  |
| $C_m$         | coefficient of pitching moment |
| $C_N$         | coefficient of normal force    |
| $C_n$         | coefficient of yawing moment   |
| cp            | center of pressure             |
| $C_Y$         | coefficient of side force      |
| $\delta$      | deflection                     |
| $\delta_{FP}$ | pitch deflection of FCA        |
| $\delta_{FR}$ | roll deflection of FCA         |
| $\delta_{FY}$ | yaw deflection of FCA          |
| d             | diameter                       |
| DOF           | degrees of freedom             |
| ER            | electrorheological             |

|                                |                                           |
|--------------------------------|-------------------------------------------|
| ESA                            | European Space Agency                     |
| ET                             | external tank                             |
| FCA                            | flight control augmentor                  |
| FEM                            | finite element model                      |
| GTO                            | geosynchronous transfer orbit             |
| $\bar{h}$                      | momentum = ( $h_x$ , $h_y$ , $h_z$ )      |
| HSCT                           | High Speed Civil Transport                |
| $I_x$ , $I_y$ , $I_z$          | moments of inertia                        |
| $I_{xy}$ , $I_{xz}$ , $I_{yz}$ | cross products of inertia                 |
| $\Lambda$                      | sweep of planform                         |
| $\lambda$                      | eigenvalue                                |
| $\ell_{cp}$                    | moment arm from cg to cp                  |
| $\ell_E$                       | moment arm from cg to engine gimbal plane |
| LV                             | launch vehicle                            |
| LaRC                           | NASA Langley Research Center              |
| LE                             | leading edge                              |
| LEO                            | low-Earth orbit                           |
| LeRC                           | NASA Lewis Research Center                |
| LH2                            | liquid hydrogen                           |
| LOX                            | liquid oxygen                             |
| $\mu$                          | launch vehicle mass                       |
| M                              | Mach number                               |
| $\bar{M}$                      | vector moment = ( $l$ , $m$ , $n$ )       |
| MAC                            | mean aerodynamic chord                    |

|                     |                                              |
|---------------------|----------------------------------------------|
| max Q               | maximum dynamic pressure                     |
| MEMS                | microelectrical-mechanical systems           |
| MMC                 | metal matrix composite                       |
| MRP                 | moment reference point                       |
| MSFC                | NASA Marshall Space Flight Center            |
| N                   | number of cycles                             |
| NASP                | National Aerospace Plane                     |
| NLS                 | National Launch System                       |
| NP                  | neutral point                                |
| $\bar{\omega}$      | vector angular velocity = ( p, q, r)         |
| $\omega_n$          | natural frequency                            |
| $\tilde{\phi}$      | phase angle                                  |
| P                   | period of oscillation                        |
| PE                  | piezoelectric materials                      |
| $\bar{r}$           | radial distance                              |
| RCS                 | reaction control system                      |
| RLV                 | reusable launch vehicle                      |
| Re                  | Reynolds number                              |
| S                   | reference surface area                       |
| $S_F$               | surface area of flight control augmentor     |
| $S_{\text{wetted}}$ | wetted surface area of entire launch vehicle |
| SM                  | static margin                                |
| SMA                 | shape memory alloy                           |
| $\tau$              | taper ratio                                  |
| TE                  | trailing edge                                |

|                                                                     |                                                                |
|---------------------------------------------------------------------|----------------------------------------------------------------|
| TMC                                                                 | titanium matrix composite                                      |
| TVC                                                                 | thrust vector control                                          |
| $\bar{V}$                                                           | vector linear velocity = ( u, v, w )                           |
| $\bar{V}_c$                                                         | velocity of center-of-mass                                     |
| $X_{cg}$                                                            | location of center of gravity from moment reference point      |
| $X_{cp}$                                                            | location of center of pressure from moment reference point     |
| $\psi, \theta, \phi$                                                | Euler angles                                                   |
| $\zeta$                                                             | damping ratio                                                  |
| $C_{m_0}$                                                           | coefficient of pitching moment at zero angle of attack         |
| $C_{m_\alpha} = \frac{\partial C_m}{\partial \alpha}$               | coefficient of pitching moment gradient                        |
| $C_{n_\beta} = \frac{\partial C_n}{\partial \beta}$                 | coefficient of yawing moment gradient                          |
| $C_{l_\beta} = \frac{\partial C_l}{\partial \beta}$                 | coefficient of rolling moment gradient                         |
| $C_{N_\alpha} = \frac{\partial C_N}{\partial \alpha}$               | coefficient of normal force gradient                           |
| $C_{m_\delta} = \frac{\partial C_m}{\partial \delta}$               | longitudinal control effectiveness                             |
| $C_{m_{\dot{\alpha}}} = \frac{\partial C_m}{\partial \dot{\alpha}}$ | coefficient of pitching moment change due to $\dot{\alpha}$    |
| $C_{m_q} = \frac{\partial C_m}{\partial q}$                         | coefficient of pitching moment change due to pitching velocity |
| $C_{m_{\dot{\alpha}}} + C_{m_q}$                                    | longitudinal dynamic stability pitch damping derivative        |



$C_{\ell_p} + C_{\ell_{\dot{\beta}}} \sin \alpha$  lateral dynamic stability roll damping derivative

$C_{n_r} + C_{n_{\dot{\beta}}} \cos \alpha$  directional dynamic stability yaw damping derivative

## CHAPTER I. INTRODUCTION

### 1.1 Background

The Space Shuttle was only the first step in achieving routine access to space. Recently, the NASA Marshall Space Flight Center (MSFC) has been studying a whole new spectrum of launch vehicles (LV's) for space transportation. In this research, a LV is distinguished from a missile in that it is defined as a large heavy-lift vehicle designed to carry large payloads to orbit. Some of these could transport components of the Space Station to orbit, and some could take us to Mars and beyond to boldly expand our frontiers of knowledge.

In all our future LV designs, decreasing the structural weight will always be of great concern. This is tantamount to increased payload capability, which in turn means reduced cost-per-pound to orbit. One very significant increase in payload capability has been defined. In a LV family recently studied at MSFC, it has been shown that a sizable weight savings can be realized by a rearrangement of the internal propellant tanks. Studies have been conducted both at MSFC and at Lockheed-Martin Corporation, maker of the Space Shuttle External Tank (ET), which show that approximately 10,000 lb can be saved by inverting the relative positions of the liquid hydrogen ( $LH_2$ ) and the liquid oxygen (LOX) propellant tanks in the ET used in the National Launch System (NLS) family of LV's.

As the vehicle sits on the launch pad, in the conventional configuration the heavier LOX tank is located on top of the lighter  $LH_2$ . This requires a heavy structural member between the two tanks to prevent the lighter  $LH_2$  tank from being crushed. This configuration also requires large, long, and even drag producing LOX feed lines running the length of the vehicle on the exterior fuselage. If the relative position of the propellant tanks is inverted, both the heavy structural separation member and the long LOX feed lines could be deleted. This would also reduce the fueling time.

While the structures community at MSFC was elated with this finding, the LOX tank aft configuration gave the family of vehicles an aft center-of-gravity (cg) location which surfaced controllability concerns. In the conventional configuration, the LV's are controlled in the ascent trajectory by the gimbaling of their rocket engines. Studies have been conducted at MSFC which showed that the resulting aft cg configured LV's would not be adequately controllable with the engine gimbaling alone.

## **1.2 Problem Statement**

It is known that the controllability of an aft cg LV is decreased since the moment arm between the cg and the engines is decreased. Today, more aft cg LV's are appearing. In addition to an aft cg being caused by an internal rearrangement of propellant tanks, aft cg LV's are appearing due to heavier rocket engines, larger numbers of aft engines, and aft end strap-on boosters. Therefore, in the new spectrum of LV's being considered, the controllability of the aft cg configured vehicle must be assessed. When the available control authority has been determined to be inadequate or marginal using engine gimbaling alone, some means of flight control augmentation is required.

In this research effort, the author has proposed a novel solution to provide the required flight control augmentation for an aft-cg configured LV when needed most in the ascent trajectory, the time during maximum aerodynamic loads on the vehicle. [1-8] The LV used in this research is the 1.5 stage NLS with design modifications. The LH<sub>2</sub> and LOX propellant tanks in the ET have been interchanged allowing a significant increase in payload capability, but giving the vehicle an aft cg. It has been determined that engine gimbaling alone does not provide adequate control. The required flight control augmentation is provided by all-movable aerodynamic flight control augmentors that could incorporate smart materials for actuation and advanced composites for reduced thermal protection requirements. This solution not only solves the original problem of

augmenting the control of this LV family, but can also be used in any other LV with marginal control capability to augment control, alleviate loads, reduce engine gimbaling requirements, provide engine actuator failure protection, and enhance crew safety and vehicle reliability by providing more control in engine-out events. Aerodynamic surfaces have not been designed for a large LV in nearly 40 years in the U.S. Use of aerodynamic surfaces on LV's has been discontinued because of their weight. The all-movable flight control augmentors designed in this research have never been used on a large Saturn-class LV to the author's knowledge.

The flight control augmentors designed in this research can reduce wind restrictions. Conventionally, the LV aerodynamic loads during ascent are alleviated by turning the vehicle into the wind, thereby reducing the flight angle-of-attack ( $\alpha$ ) and/or angle-of-slideslip ( $\beta$ ). Thus, load relief is accomplished at the expense of trajectory deviation. The Saturn V never flew at  $\alpha > 4^\circ$ . In a gravity-turn nominal trajectory, the vehicle flies at nearly  $0^\circ$  of  $\alpha$  during ascent to produce minimum bending stress on the vehicle. Thus, the need to correct for wind and fly at low  $\alpha$  results in sizable deviations from the ideal lifting trajectory and the gravity-turn trajectory is not an efficient one. The near zero-lift trajectory is not the most efficient one because a lift trajectory can carry more payload. The designed flight control augmentors can provide load relief control which is conventionally provided by engine gimbaling alone. Reducing the need for thrust vectoring can reduce trajectory deviations. Therefore, taking advantage of aerodynamic control can significantly improve trajectory efficiency. Load relief is most necessary when the aerodynamic loads on the vehicle are greatest. However, it is at this very same time that the flight control augmentors would provide the most significant assistance.

The added control capability through the use of these surfaces allows greater tolerance of wind magnitudes and a minimization of bending moments on the vehicle both during ascent and during launch. For prelaunch, the unfueled vehicle on the pad is

assumed to withstand peak loads of 75 knots, and when fueled at lift-off peak winds of 50 knots. The environmental disturbances are multiplied by 1.5 to account for von Kármán vortex shedding effects. Wind profiles show that the greatest steady wind speeds occur between 20,000 and 60,000 ft, with a gust overshoot of up to 50%. [9] The more the engines are required to gimbal, i.e., the greater the required engine gimbal deflection angles, the more engineering design and cost is involved to have the propellant ducts move with the gimbal action while maintaining a full flow of fuel. Gimbale engines are difficult to seal and have high actuation torques. The extension, compression, and torsion of propellant ducts become limiting factors of engine gimbaling. Thus, the designed surfaces of this research provide not only the required control augmentation but a plethora of additional significant benefits.

### **1.3 Approach**

Although aerodynamic surfaces have not been designed for use on large LV's in 40 years because of their weight, NASA has a rich national heritage of LV's that have used aerodynamic surfaces to provide flight stability and control. The primary current U.S. heavy-lift expendable LV's, the Atlas, Delta, and Titan, do not use aerodynamic surfaces. Every engineering research endeavor should start by looking at what has been done before. Since this research has proposed to design aerodynamic flight control augmentors and analyze their effects on the flight stability and control of the LV, the first two years were spent on extensive literature searches and investigations. There is a paucity of flight dynamics stability and control research on LV's in the literature. The flight stability and control analysis of Chapters VII and IX hopefully expand this area, prior chapters being preliminary to this analysis. NASA MSFC has designed and built the LV's used at the start of the U.S. Space Program and keeps a large repository of documents in Huntsville, Alabama. The author has searched through these documents for

years to ferret out information on the very small percentage of LV's that have used aerodynamic surfaces in order to lay the groundwork for the designs of this research. [10-14] The next two chapters illuminate the fact that the method of design was to test many varieties of planform shapes. The results of this exhaustive research and the only use of aerodynamic surfaces on LV's in the entire history of the U.S. Space Program are presented in Chapter II. The Saturn V LV has flown 13 successful missions and took us to the Moon using 300 ft<sup>2</sup> of aerodynamic surfaces. The focus of Chapter III is the aerodynamic surface design details of the Saturn series and the X-15. Although no longer used in the U.S. because of weight, aerodynamic surfaces are currently being used by other nations on large LV's. Today there are a myriad of LV's actively launched from nearly two dozen geographic sites. Current use of aerodynamic surfaces by French (ESA), Russian, Chinese, and Japanese LV's is summarized in Chapter IV.

In Chapter V the control augmentation requirements of the experimental aft cg configured LV of this research are determined. The determined flight control augmentation requirements, together with the above reviews, the U.S. Air Force Stability and Control DATCOM as a preliminary design tool, the LV mass properties, and the LV ascent trajectory data are used to design the candidate flight control surface augmentors. The design of a static stability wind tunnel test program for these candidate flight control augmentors and the host experimental aft cg LV is outlined in Chapter VI. The wind tunnel test program produces force and moment data from which the static stability derivatives and the center-of-pressure (cp) location on the vehicle are calculated. The fabrication of the candidate flight control augmentor designs and the experimental LV test article are covered. In Chapter VII the wind tunnel test data is reduced and utilized to conduct the vehicle static stability and control analyses. Results are compared to Saturn V and Missile DATCOM aerodynamic prediction code data. The best candidate designs are then chosen. [15] In Chapter VIII the design requirements of the dynamic stability wind

tunnel test program, the new LV test article, and the chosen augmentors for this program are outlined. The dynamic stability analyses, based on the dynamic stability wind tunnel test program, is described in Chapter IX and compared to analytic results.

This research effort draws on advancements in many other fields. LV aerodynamic surfaces have not been designed in the U.S. for nearly 40 years because of weight. However, since NASA went to the Moon with the Saturn V using 300 ft<sup>2</sup> of aerodynamic surfaces, the wealth of smart materials and advanced composites that have been developed allow for the design of very lightweight, strong, and innovative LV flight control augmentors with possible smart actuators. In Chapter X a state-of-the-art assessment of smart materials and advanced composites which are directly applicable to aerodynamic flight control augmentors of future LV's is conducted. From this assessment an advanced composite, titanium matrix composite, was selected for the designed flight control augmentors. Then, as part of this research the author spent several days at the extensive production facilities for titanium matrix composite (TMC) at the Textron Specialty Materials Company in Lowell, Massachusetts. This is the world's first plant (which has been government funded) built solely for the production of TMC (for NASA's National Aerospace Plane). While there, the author studied the TMC fabrication technology including chemical deposition plants and manufacturing plants. A Flight Control Augmentor Design Review was held with the Director of Research, Director of Fabrication, Director of Product Applications, Director of Operations, and the TMC Plant Manager. The 12 sets of designed flight control augmentors were reviewed, together with their prospective loads and temperature environments. Material and fabrication details were discussed at length and cost estimates were obtained for the full scale augmentors. Thus, any added weight concerns are addressed in Chapter X, and it is further shown that smart materials could be incorporated in the designed augmentors for smart actuation. [16]

While the designed flight control augmentors have remedied the aft cg controllability problem, it was desired to know how the vehicle bending frequencies had changed with the aft cg. In reality a LV is not a rigid body, but an elastic body. The experimental LV has a very far aft cg. In Chapter XI changes in elastic properties due to the aft cg are assessed to check whether there will be any coupling of elastic body frequencies and control system frequencies. When a LV flies through the atmosphere during its ascent trajectory, aerodynamic forces act over its entire fuselage and the propellants are sloshing internally. Resulting vibrations can and have caused catastrophic failure. Internal acoustic fields generated by the random vibrations can have adverse effects on the crew or result in damage to sensitive payloads. LV deformations due to bending and torsion can significantly affect the stability and control, hence vehicle performance. The lowest elastic vehicle frequency must be well separated from the frequency of the control system. The significant modes of vibration were identified and response spectra were compared for the original LV configuration and the experimental far aft cg configuration. [17]

The augmented control authority achievable with the use of the selected flight control augmentor designs, the conclusions of this research, and applications to other flight vehicles and future work to build on this research are summarized in Chapter XII. Figure 1 shows the flow chart of this conducted research effort.



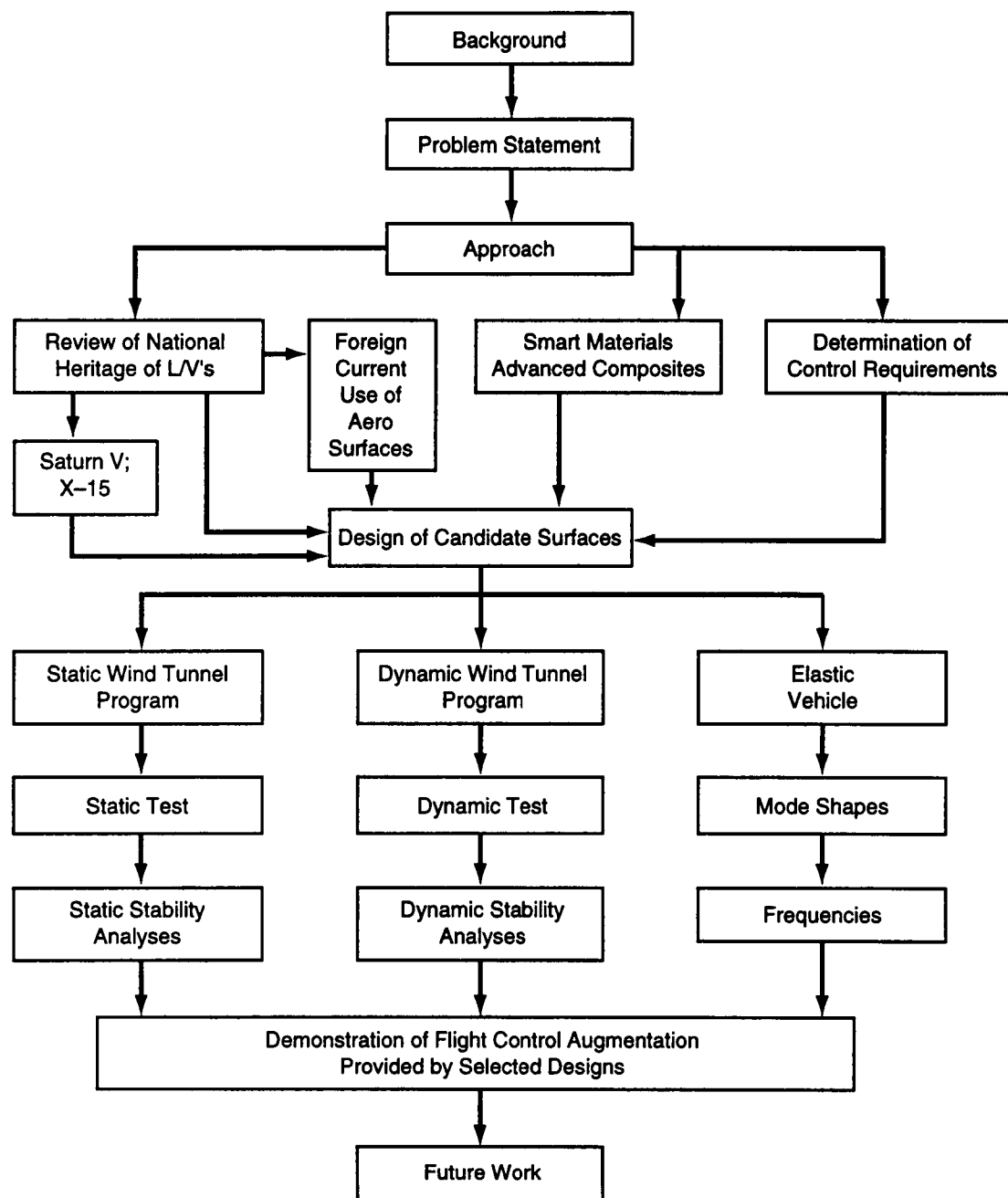


Figure 1. Flow chart of conducted research.

## **CHAPTER II. REVIEW OF NATIONAL HERITAGE OF LAUNCH VEHICLES USING AERODYNAMIC SURFACES TO PROVIDE FLIGHT STABILITY AND CONTROL**

A professor of long ago preached a principle that the author has tried to practice throughout her NASA career. He said "In any engineering research endeavor, the first step should be to see what has been done before." This chapter serves as a precursor study to the design of the aerodynamic flight control augmentors of this research.

Aerodynamic surfaces have not been designed for a large LV in the U.S. in nearly 40 years. The primary current U.S. heavy-lift expendable LV's, the Atlas, Delta, and Titan, do not use aerodynamic surfaces. We have stopped using aerodynamic surfaces on LV's because of their weight. The all-movable flight control augmentors designed in this research have never been used on a large Saturn-class LV to the author's knowledge. The author has conducted many dozens of literature searches, sifted through the entire U.S. Space Program, and had many consultations with designers of the first U.S. LV's in order to compile this information on the few LV's that have used aerodynamic surfaces to provide flight stability and/or flight control.

NASA MSFC has designed and built the LV's used at the start of the U.S. Space Program and keeps a large repository of documents in Huntsville, Alabama. The author has searched through these documents for years to ferret out information on the small percentage of LV's that have used aerodynamic surfaces in order to lay the groundwork for the designs of this research. The results of this exhaustive research illuminate and underline the fact that the method of design was to test many varieties of planform shapes, and it is shown that LV's have flown successfully with the LOX tank located aft.

Although not designed for use on large LV's in 40 years because of their weight, NASA has a rich national heritage of LV's that have used aerodynamic surfaces for both flight stability and flight control. The Saturn V LV covered here is very similar in many

respects to the LV of this research and was used as a standard of comparison in the design of the 1.5 stage NLS. Traditionally, as opposed to space craft [18-20], the required control torque for a LV can be provided by aerodynamic flight control surfaces, a reaction control system (RCS), or thrust vector control (TVC) which includes both movable jet vanes internal to the exhaust flow or gimbaling of the entire engines.

This review of how controls were actuated and coupled also gives insight on how smart material actuators might be used in the designed FCA's of this research. The problems discussed of hot exhaust gas disposal, together with the close proximity of the aft located cryogenic LOX, could lend themselves to operating a smart material actuator which requires the alternate application of cold and heat. This chapter also highlights the importance of care in gimbaling the engines in order not to resonate with any low natural bending frequencies of the vehicle. Figure 2 lists the more prominent use of aerodynamic surfaces in our national heritage of large LV's.

## **2.1 Dr. Wernher von Braun's V-2**

Although today, movable canards, trailing edge (TE) air vanes, movable fins, movable wings, and TE flaps are used on many missiles, our heritage of aerodynamic surface use on large Saturn class LV's begins with Dr. Wernher von Braun's V-2. Figure 3 shows the V-2 with its use of aerodynamic surfaces for flight stability and for flight control (air rudders), and jet vanes internal to the rocket engine exhaust flow (jet rudders) for TVC. The V-2 was an aerodynamically stable vehicle. The jet vanes provided primary control; however, since they were located close to the single engine's longitudinal axis, they provided only weak roll control. The effect of side winds on the stability fins created rolling moments due to combined pitch and yaw, which was proportional to the dynamic pressure. The jet vanes could not overcome these moments, therefore, small aerodynamic flight controls were added to the tips of the fins. These flight control surfaces were coupled with the jet vanes (Fig. 4) and were driven by the

same actuators. Likewise the FCA's of this research may be coupled to the engine actuators. Figure 3 shows that the V-2 alcohol fuel was located forward and the heavier LOX oxidizer was located aft. Figure 5 shows the winged version of the V-2, the Wasserfall, and the planforms that were tested subsonically and supersonically. These control surfaces had sharp TE's. Figure 6 shows the winged V-2 trial planforms. Figure 7 shows the Hermes II that was a V-2 with blunt TE aerodynamic flight controls located on the forward fuselage (pitch control canards) and enlarged aft aerodynamic surfaces for flight stability, since it was the first attempt to fly an unstable LV. In spite of the enlarged fins, the large wing with ramjet engines rendered the LV unstable. An accelerometer control was supposed to stabilize the vehicle. That control system worked well in bench tests, but failed in flight because the vehicle vibrations swamped the accelerometer. The vehicle lost control during a White Sands launch and the project was canceled when the Redstone project was started. [10, 11]

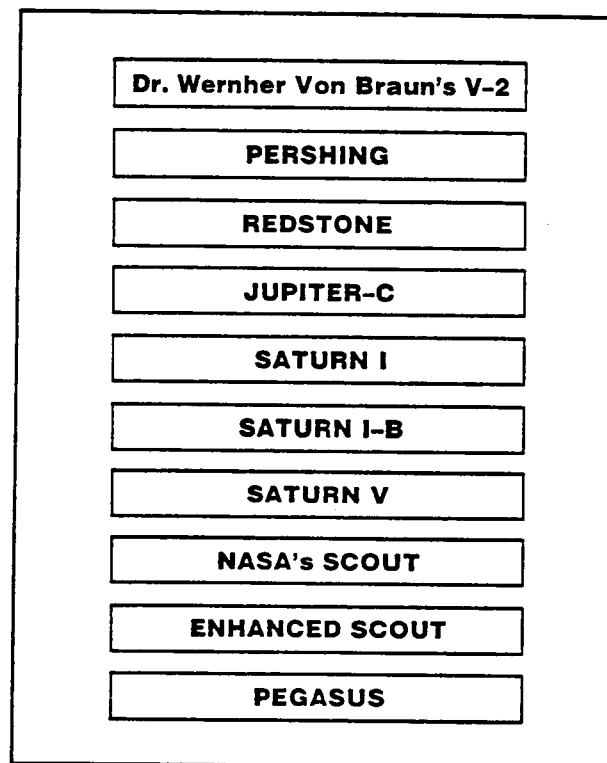


Figure 2. National heritage of launch vehicles using aerodynamic surfaces.

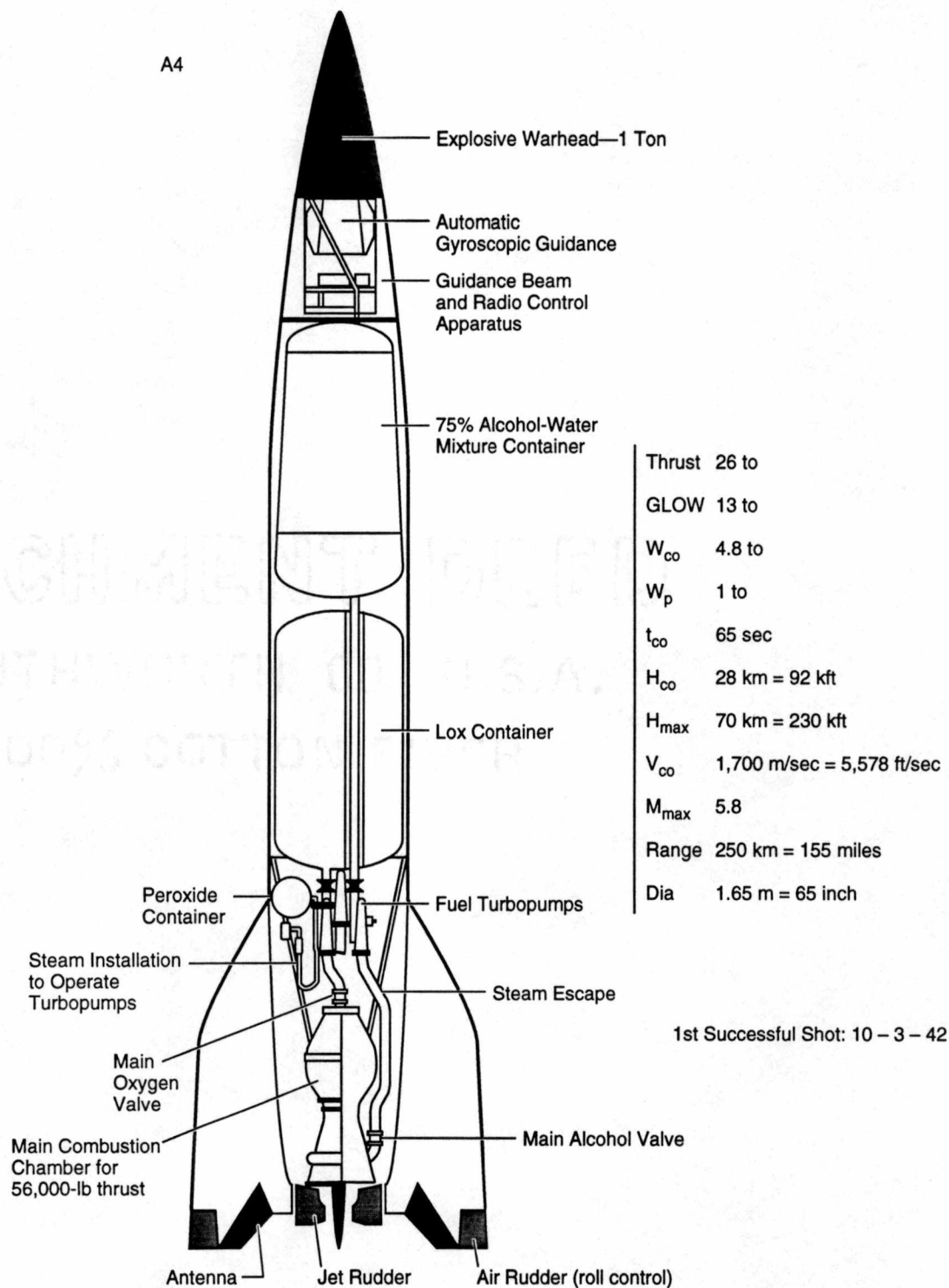


Figure 3. V-2 ballistic missile.

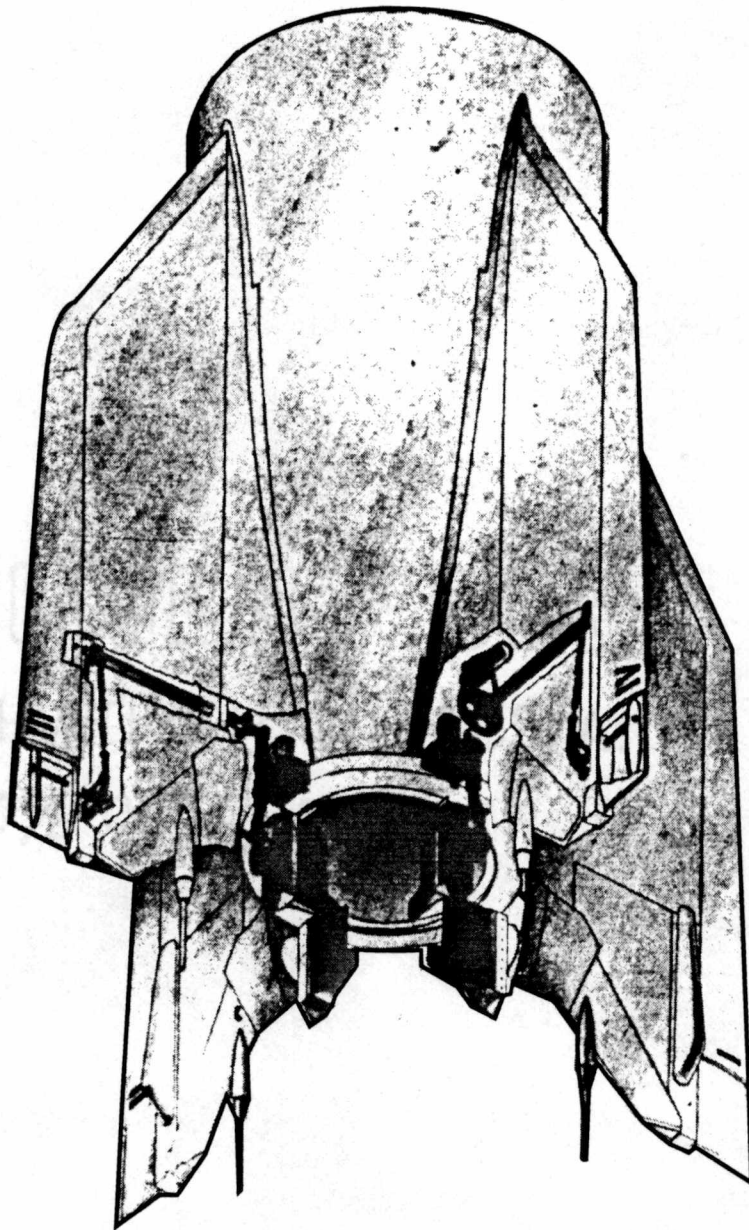


Figure 4. V-2 flight controls and jet vanes.

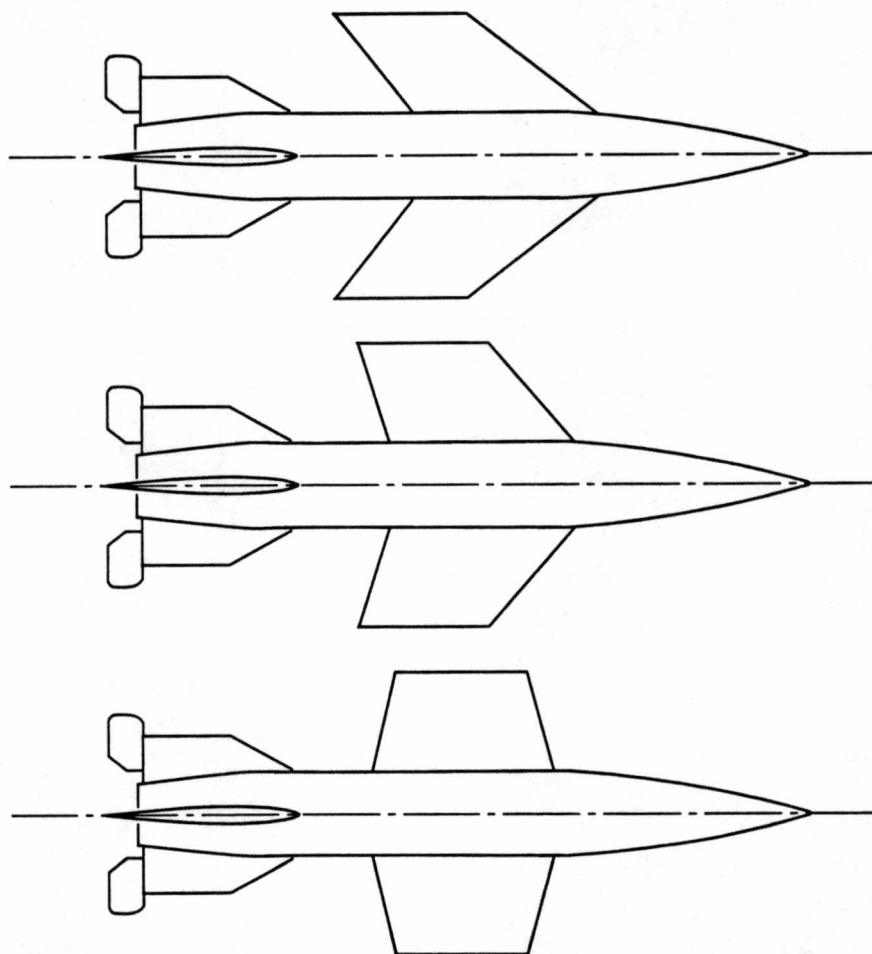
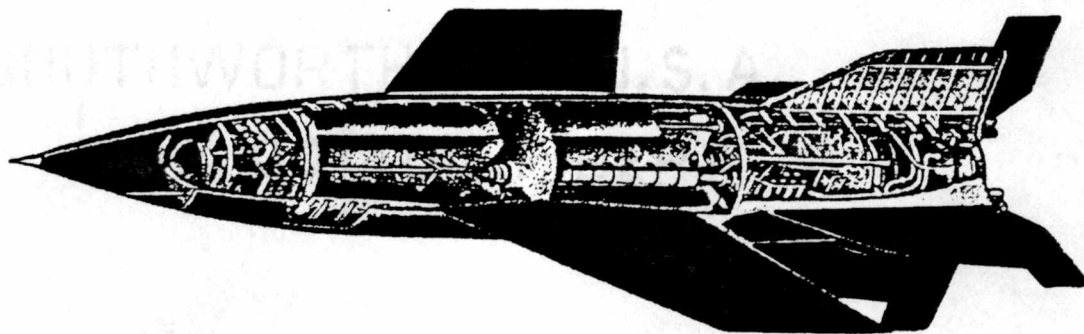


Figure 5. Winged V-2 configurations tested in subsonic and supersonic flow.

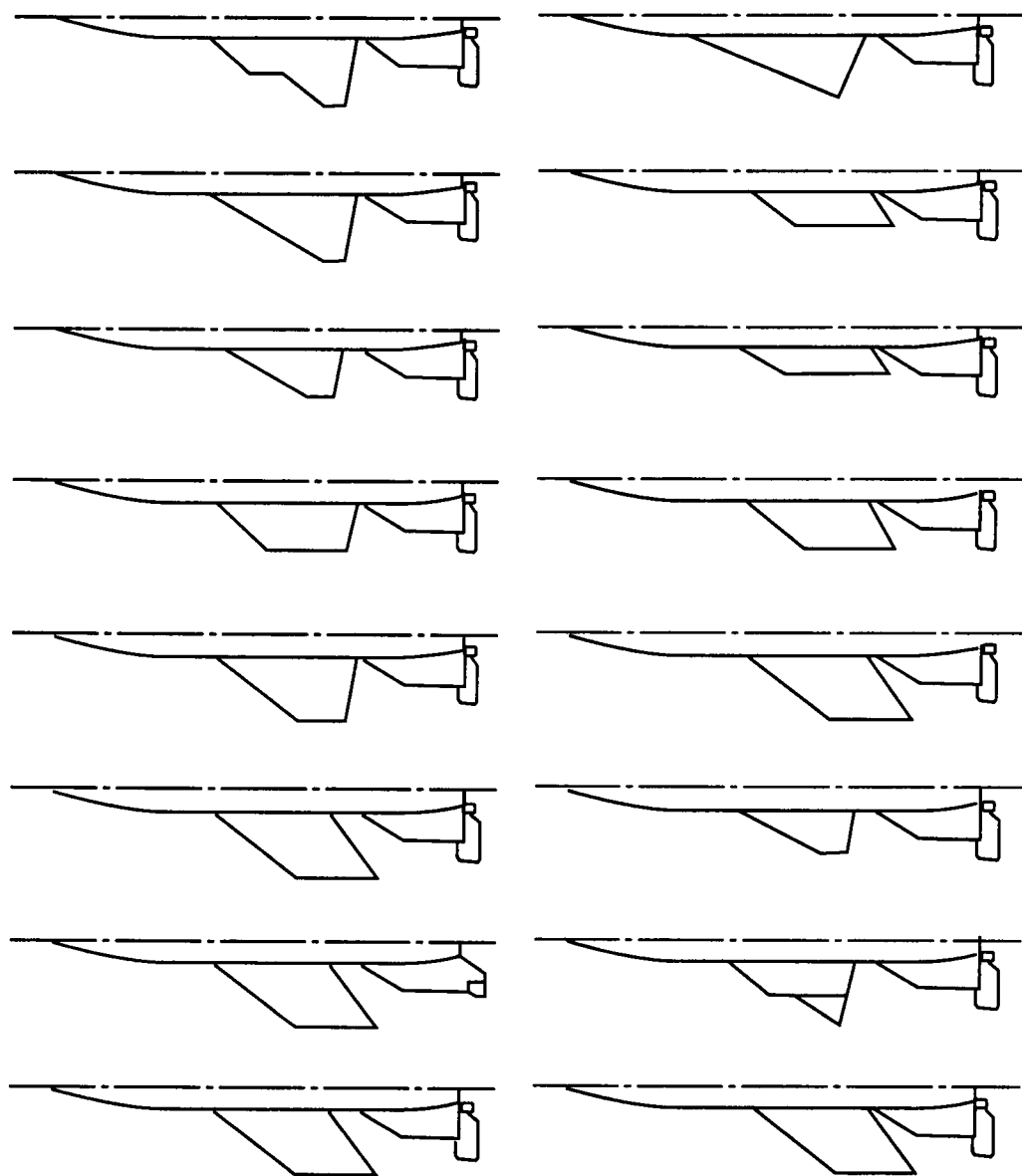


Figure 6. Winged V-2 trial platforms.



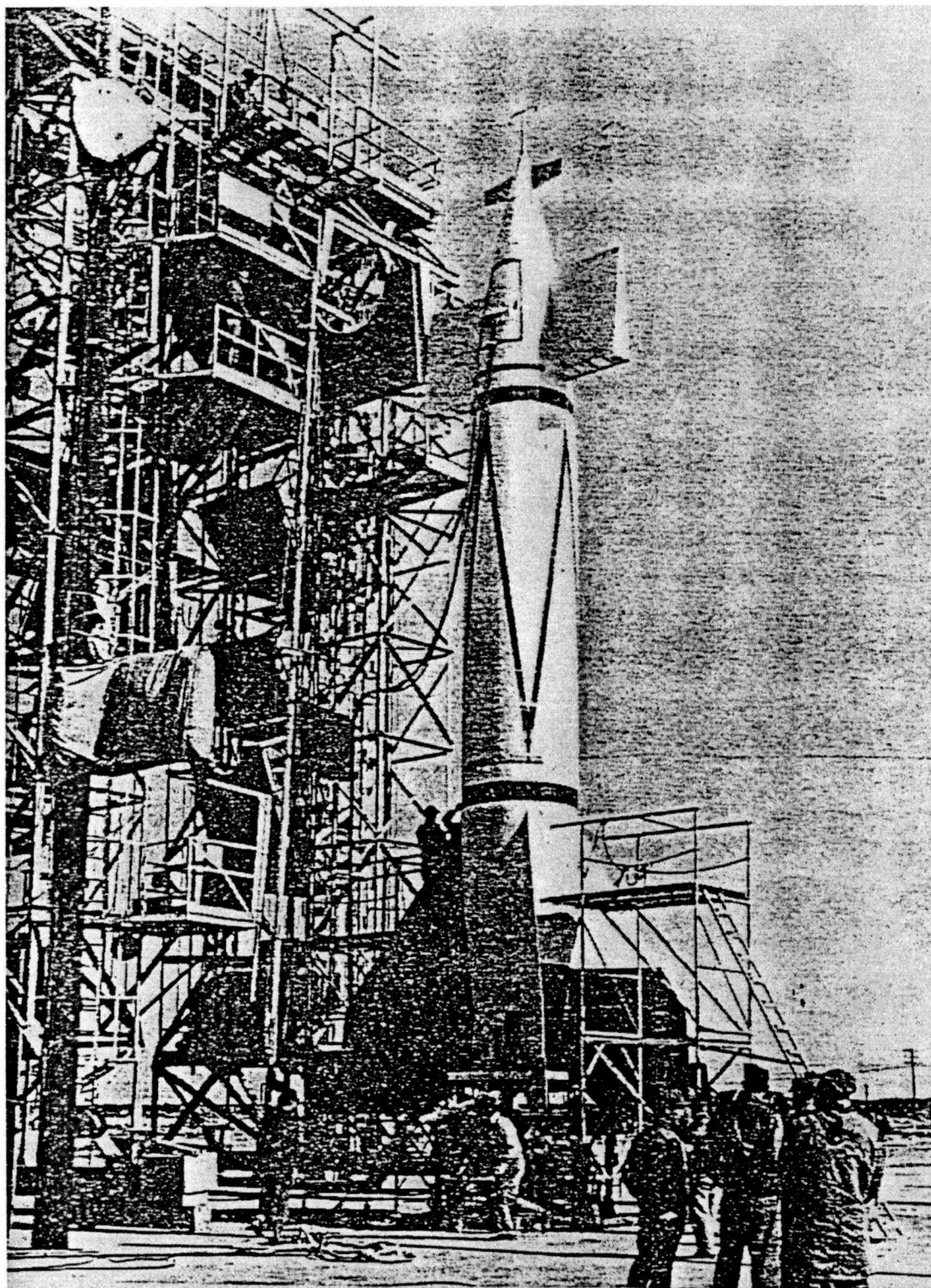


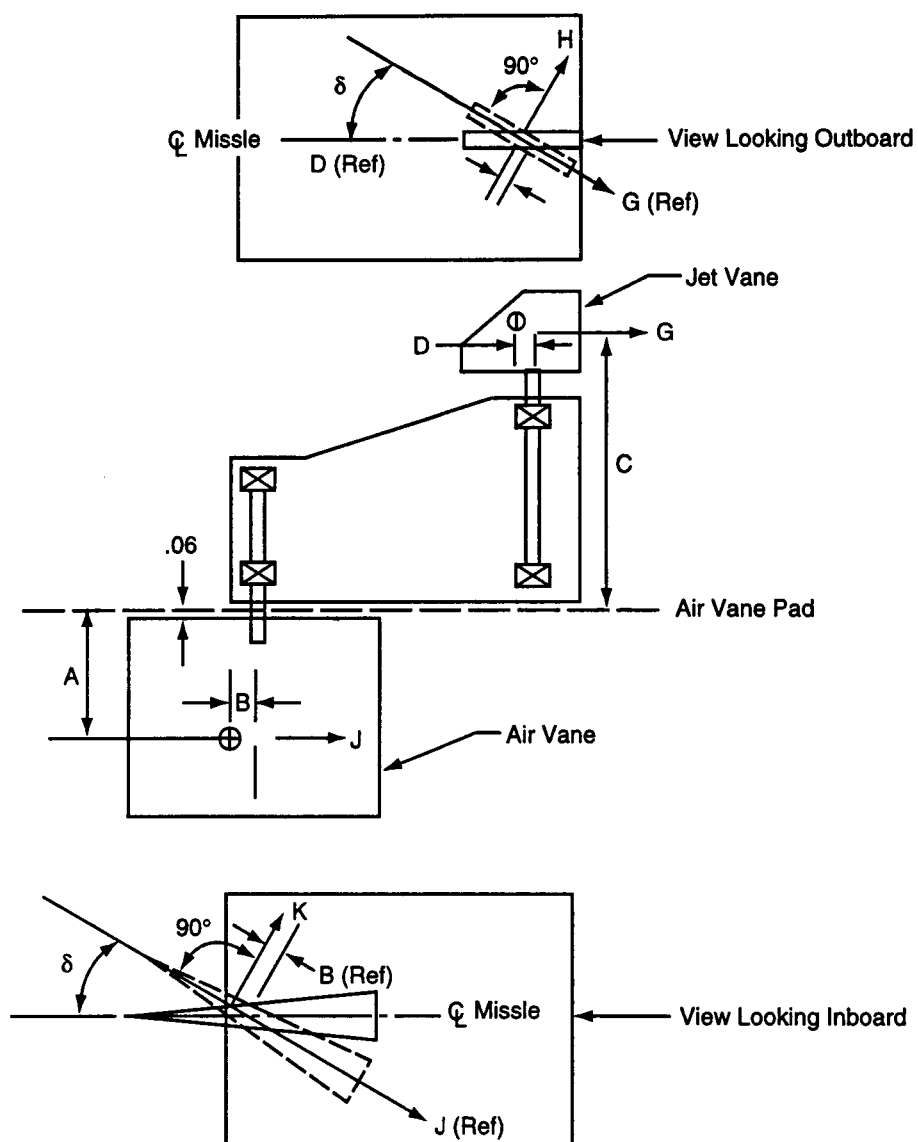
Figure 7. The Hermes II.

## 2.2 Pershing

Many MSFC engineers worked on the Pershing, which was a two-stage solid-propellant flight vehicle. Launch weight was 10,229 lb, length was 34.05 ft, diameter was 40 in, and range was 400 nmi. Pershing used the solid propellant polybutadiene acrylic acid, giving 20,000 lb of thrust. The Pershing was an unusual three-aerodynamic-flight-control design with the flight controls located 120° apart around the exterior fuselage circumference. Also used for TVC flight control were three jet vanes in the exhaust flow.

Figures 8 and 9 show the two sets of three aerodynamic flight controls (air vanes) located forward and aft on each stage of the fuselage. [21] The three aerodynamic surfaces located aft provided flight stability, and the three jet vanes located internal to the exhaust flow provided TVC. Each aerodynamic flight control was connected to its respective jet vane and was synchronously deflected proportionally to the error signal input in a one-to-one ratio by a unitized hydraulic package. The unitized hydraulic package consisted of an electric motor pump, accumulator, reservoir, servovalve, actuator, and connecting linkages in one integrated package. The maximum output stall torque of the unitized package was 3,000 in-lb, and the normal maximum operating torque was 2,104 in-lb at a deflection rate of 250° per second.

The three rectangular aerodynamic flight controls had blunt TE's and were made of plastic laminate with a high-pressure-molded, phenolic leading edge (LE). The three internal jet vanes were made of arc cast alloy of 15% molybdenum and 85% tungsten by weight. Pershing was wind tunnel tested with 8% and 4% thick aerodynamic flight controls as well as 10% thick biconvex. The 8% thick double-wedge aerodynamic flight controls yielded a slope of the normal force vs.  $\alpha$  curve which was 50% higher in the subsonic range and 10% higher in the supersonic range than the original 12.8% thick aerodynamic flight controls that were used. [21]



| All Loads (Ultimate) |      |             |      |            |            |     |      |     |      |
|----------------------|------|-------------|------|------------|------------|-----|------|-----|------|
| Conditions           | A    | B           | C    | D          | $\delta$   | G   | H    | J   | K    |
| Max. HGH & Torque    | 8.13 | $\pm 1.112$ | 13.6 | $\pm 1.18$ | $26^\circ$ | 235 | 1375 | 132 | 5150 |
| Max. Tur. Vane Fix.  | 7.76 | 4.52        | 0.0  | 0          | $0^\circ$  | 0   | 0    | 200 | 2288 |

Figure 8. Pershing aerodynamic flight controls and loads.

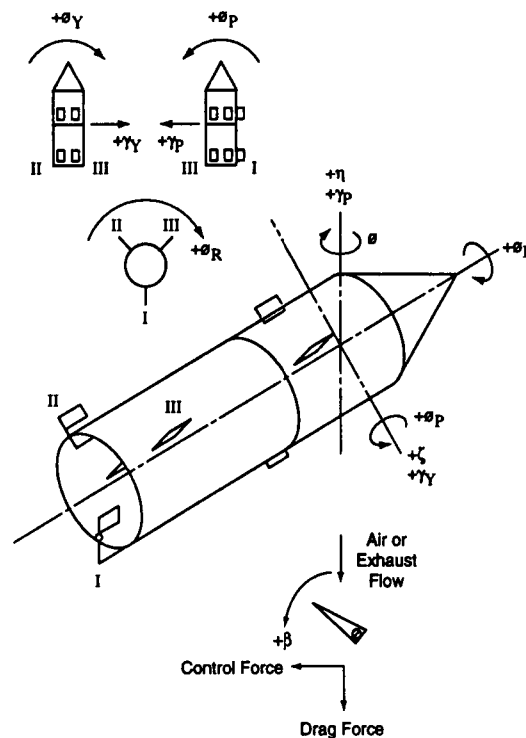


Figure 9. Pershing aerodynamic flight control orientation.

### 2.3 Redstone

Redstone was the first large LV developed by the United States. Redstone was developed from the V-2 and launched the first astronaut. It began operation in 1958, had a launch weight of 62,000 lb, and a range of 175 mi. It was used to launch the Mercury spacecraft. The Redstone was a single-stage LV 69-ft long and 70 inches in diameter. It used liquid propellants, alcohol (19,094 lb) and LOX (25,134 lb), with the heavier LOX located in the aft position. It had a cg travel of 7.3 ft forward (10.5% of body length) (Fig. 10). The thrust was 78,000 lb from the single bipropellant rocket engine (NAA 75-110). [22]

The Redstone was an aerodynamically stable vehicle throughout its entire flight spectrum of Mach numbers. The jet vanes provided primary control; however, as in the V-2, since they were located close to the single engine's longitudinal axis, they provided

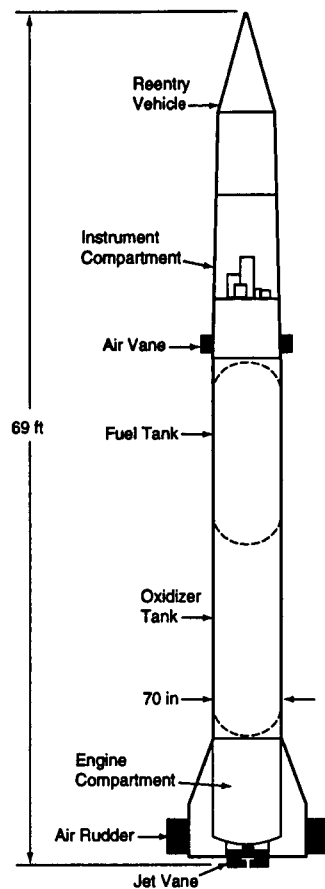


Figure 10. Redstone launch vehicle.

only weak roll control. To assist the TVC (jet vanes) in overcoming moments, small aerodynamic flight controls were added to the tips of the fins. These were coupled with the TVC and driven by the same actuators. Four carbon vanes were located in the exhaust flow for TVC. Four aerodynamic flight controls were located at equidistant points around the exterior fuselage (Figs. 11 and 12). The aerodynamic flight controls had blunt TE's and were constrained to fit the railroad tunnel profiles. The internal controls provided control until the dynamic pressure warranted the use of the aerodynamic flight controls. Cold gas control jets (RCS) were located at the roots of the warhead (forward) flight control surfaces for attitude control during exoatmospheric flight. These were hard coupled to the flight control surfaces. [10]

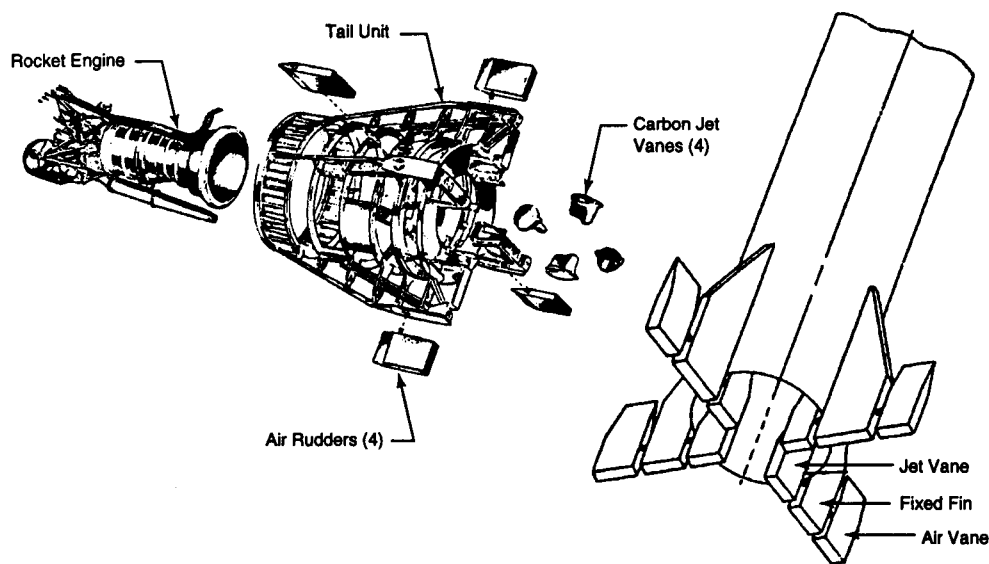


Figure 11. Redstone aerodynamic flight controls and jet vanes.

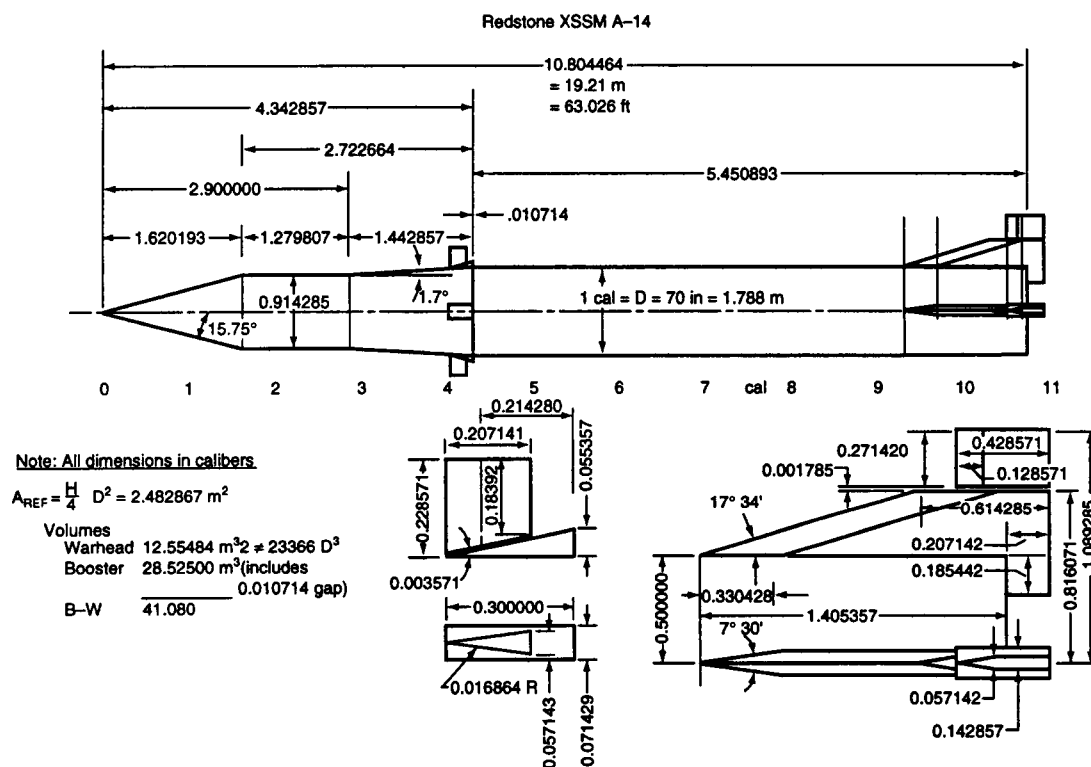


Figure 12. Redstone geometry.

## 2.4 Jupiter-C

The Jupiter-C, shown in Figure 13, was a three-stage LV, 67.3-ft long and 70 inches in diameter, with a lift-off weight of 64,610 lb. It also had the LOX tank located aft and employed an elongated Redstone as a booster for a nose-cone-cluster assembly. The Jupiter-C had two upper stages, developed by the Jet Propulsion Laboratory and carried in a cylindrical tub which could be rotated for stability at a

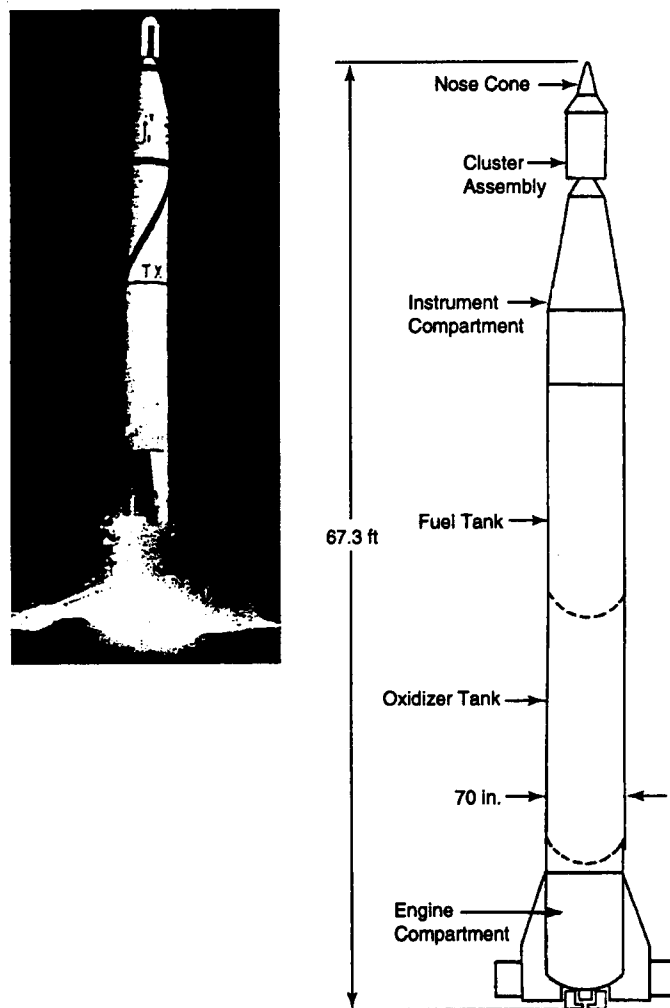


Figure 13. Jupiter-C.

spin speed of 380 r/min. The second stage was a cluster of 11 scaled-down, 6-inch, solid propellant Sergeant rockets; the third stage was a cluster of three. The nose cone was recovered after water impact and carried a deceleration parachute and a balloon to keep it afloat. In 1957, the first nose-cone recovery was made from a Jupiter-C fired to a range of 1,100 nmi. The Jupiter-C used one set of aerodynamic flight controls and one set of internal jet vanes the same dimensions as those of the Redstone. The control system included four cylindrical angle-of-attack meters that stuck out from the surface of the instrument compartment, two each in the pitch and yaw planes. They compensated for the instability of the basic configuration. The system was successful and reliable. Jupiter's role control was accomplished by using the thrust of the engine's turbine exhaust, discharged through a swiveling pipe knee at the flank of the engine compartment. [10, 22]

## **2.5 Saturn I**

The Saturn program, which landed us on the Moon, included three basic vehicles, all of which used aerodynamic surfaces: the Saturn I, Saturn I-B, and Saturn V. There were 32 Saturn launches in all.

Ten Saturn I LV's were launched by NASA from the Kennedy Space Center (KSC) from 1961 to 1965. The first four flights were a Block I test vehicle without aerodynamic surfaces and inert upper stages. The next six flights began the flight operations and sported a corolla of aerodynamic surfaces at the base, a live second stage, and the Block II designation. The Saturn I series of missions performed a multitude of atmospheric tests and meteoroid experiments. The Saturn I launched the Pegasus I, Pegasus II, and Pegasus III meteoroid detection satellites. [23]

The Saturn I Block II stood 173-ft tall with a diameter of 21.4 ft, had a lift-off weight of 1,138,000 lb, and utilized eight aerodynamic surfaces, four full-size fins and four stub fins. The small fins provided support on the launch pad (Figs. 14 and 15).



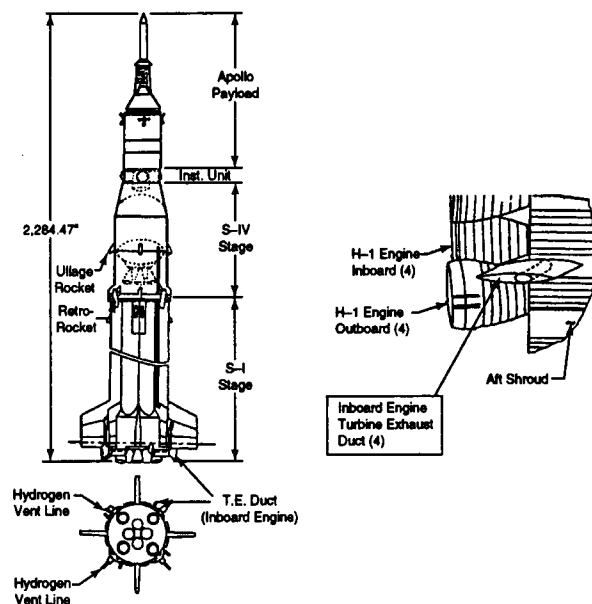


Figure 14. Saturn I, Block II.

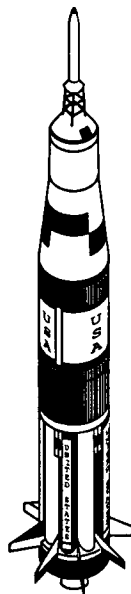


Figure 15. Saturn I-B.

The Saturn I used an eight-engine cluster of H-1 engines. The performance of the Saturn I Block II LV's progressed so well that the Saturn I's were declared fully operational by NASA officials three launches earlier than expected. [23]

The original concept for yaw, pitch, and roll called for hinged outer engines: two hinged for pitch, two hinged for yaw, and all four hinged for roll. But application of adequate control forces required fairly high deflection of the engine thrust vector, and the engine contractor (Rocketdyne) complained that this would put excessive stress on the propellant flex lines. Consequently, gimbaling of all four outer engines was adopted, achieving adequate control force with less engine deflection. The gimbal system for mounting engines permitted each engine in the cluster to swivel about for either pitch or yaw control.

In the process of refining the design of the Saturn I, two major problems emerged: flight stability and base heating. As with most large LV's, the Saturn I was highly unstable, with the overall cg located aft on the heavy, lower-stage booster, while the center-of-pressure in most flight conditions was forward on the upper stages. The low natural frequency was such that engines had to be very carefully gimbaled in order not to resonate with the vehicle bending frequencies. The eight-engine cluster caused considerable base heating. To cope with this, a heavy fire wall was installed across the base of the booster near the throat of the engines, with flexible engine skirts to permit gimbaling. For the four fixed center engines, the fuel-rich exhaust gases were piped to the edge of the booster skirt and dumped overboard into a region of high-velocity air flow. In the later vehicles, the exhaust gases were dumped exactly into the "centerstar" created by the four fixed engines. The gimbaled outboard engines required a different approach. The turbopump was fixed to the gimbaled engines; therefore, an overboard duct for them would have required a flexible coupling that could withstand the high temperature of the turbine exhaust gases. Instead, MSFC devised outboard engine attachments called aspirators, which forced the turbine exhaust into hoods on the stub fins as shown in Figure 14. [10]

## 2.6 Saturn I-B

Nine flights were made on the Saturn I-B between 1966 and 1975 to test Apollo program hardware and to launch three crews to the Earth orbiting space station, Skylab. The Saturn I-B missions included the first manned command/service module (CSM) operations (10 days), seven CMS's total; the lunar module (LM); manned launches to Skylab 4 (28 days, 59 days, and 60 days); and the manned rendezvous and docking with the Russian Soyuz.

The Saturn I-B was a two-stage LV which stood 225-ft tall, had a 21.4-ft diameter, had a lift-off weight of 1,280,000 lb, and used liquid propellants, LOX and RP-1, providing 1,640,000 lb of thrust. The main body stage was a cluster of nine propellant tanks; four fuel tanks and four LOX tanks arranged alternately around a larger center LOX tank. The Saturn I-B also used eight aerodynamic surfaces designed at MSFC, but of a different design than those of the Saturn I, since the Saturn I-B was a longer and heavier vehicle (Figs. 16 through 19). [24, 25]

Eight H-1 engines were used, four stationary inboard and four gimbaled outboard for TVC. The outboard engines were gimbaled by two hydraulic actuators. The upper stage, the S-IVB stage, used a J-2 engine that was gimbaled  $\pm 7^\circ$  for pitch and yaw control and two auxiliary propulsion systems for roll control. The H-1 engines were each canted outward  $6^\circ$ . The outboard engines were also gimbaled in a  $\pm 8^\circ$  square pattern. The gimbal rate was  $15^\circ$  per second in each plane.

The eight aerodynamic surfaces were  $10^\circ$  wedges with blunt TE's of a semimonocoque construction. They provided aerodynamic stability in the midregion of the first stage flight. They were also used to support the LV on the launch pad prior to ignition and during the holddown period after ignition. A heat shield was attached to the TE of each aerodynamic surface to protect them from engine exhaust. The new design of the Saturn I-B aerodynamic surfaces was eight identical surfaces and replaced the

Saturn I's four large and four small surfaces. These aerodynamic surfaces had 2.5 times the lift and weighed 1,800 lb less. [24, 25]

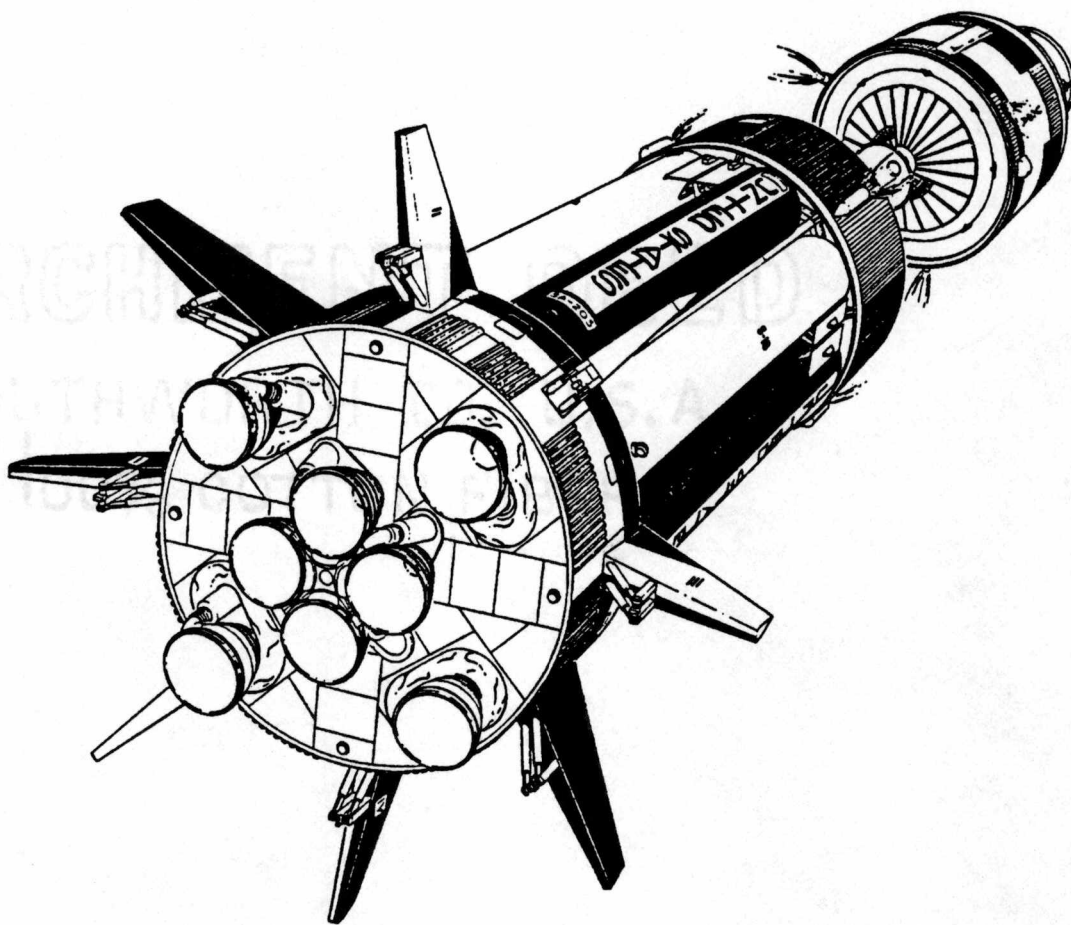


Figure 16. Saturn I-B aerodynamic surfaces.

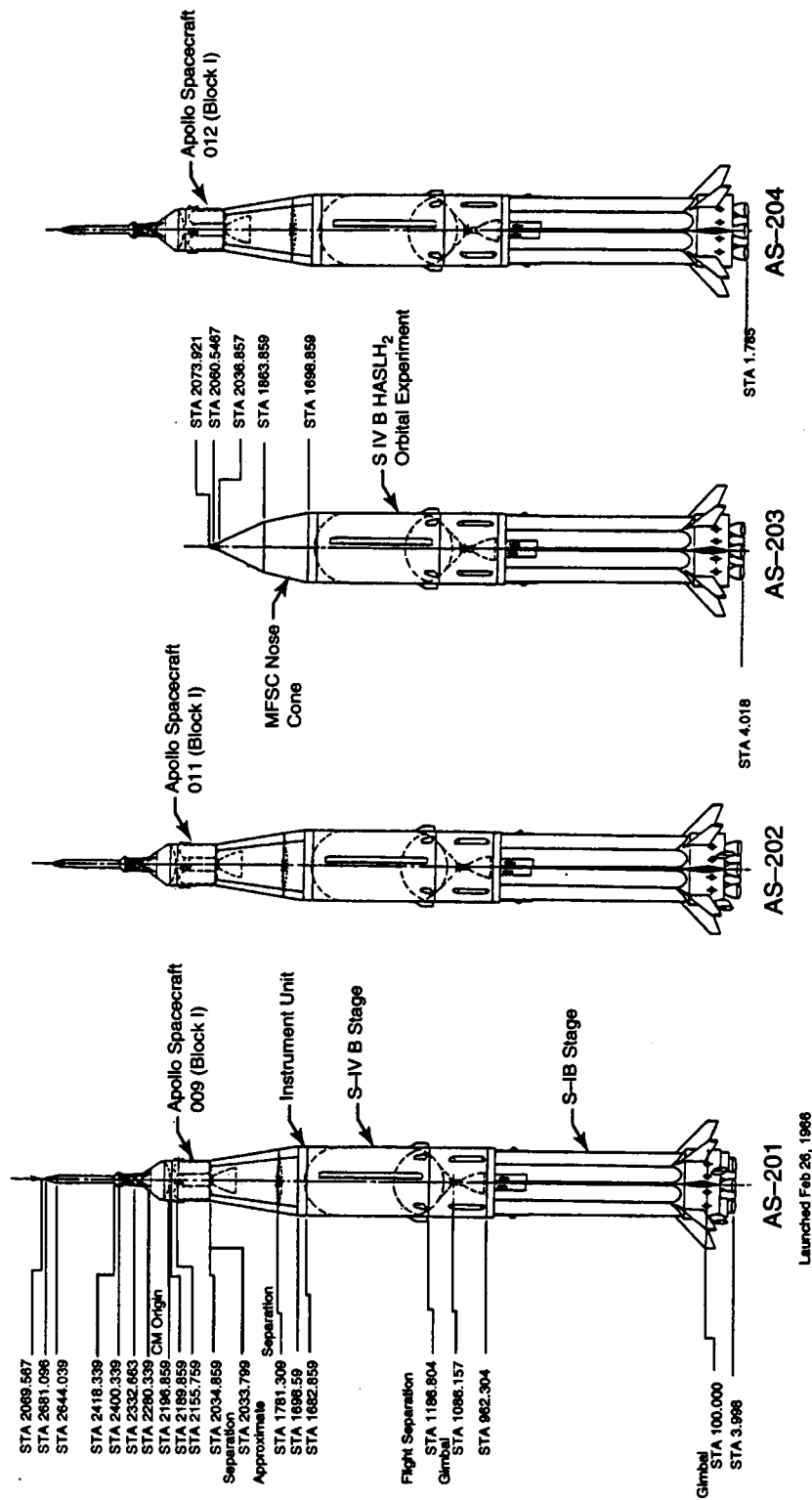


Figure 17. Saturn I-B series.

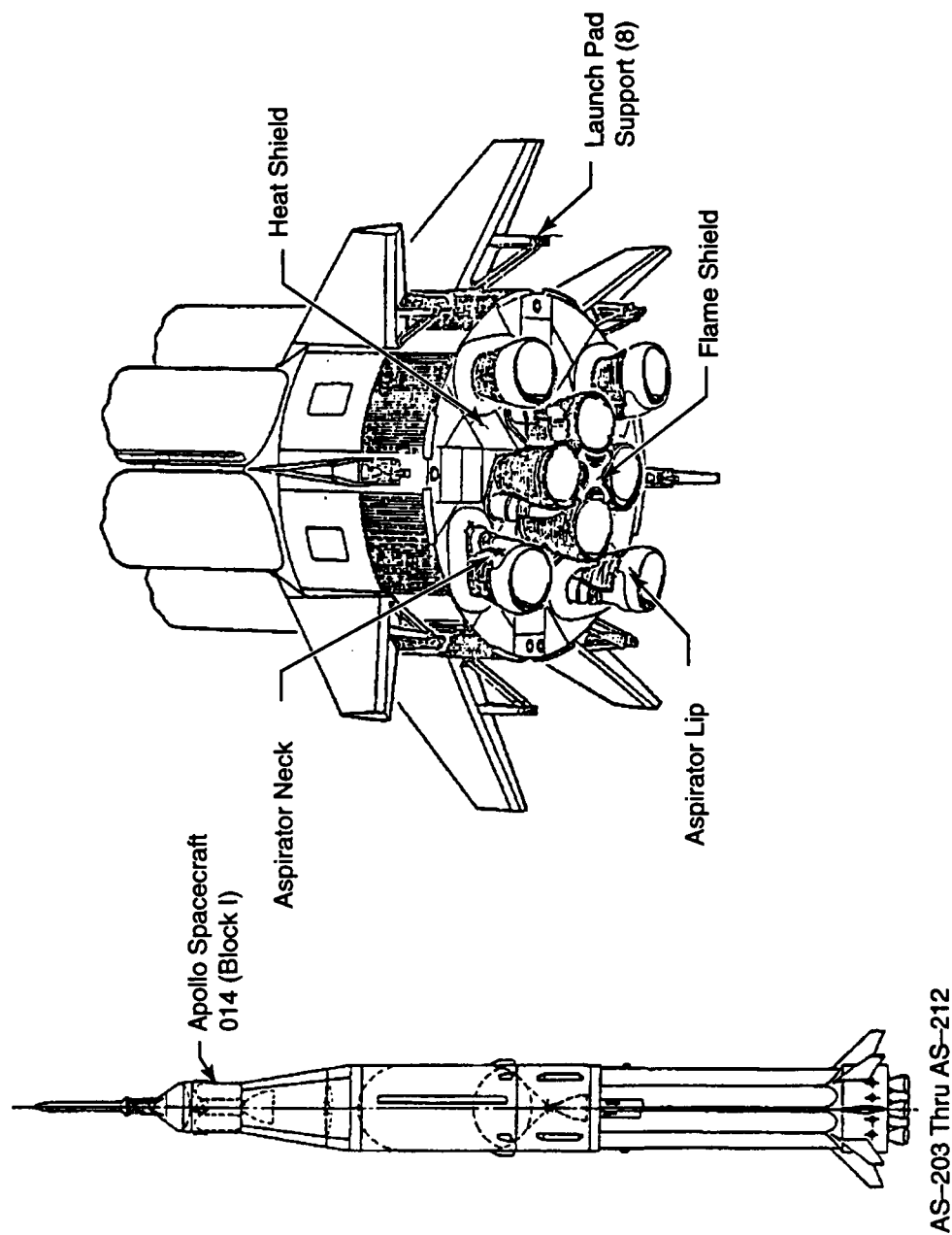


Figure 18. Saturn I-B inner engine turbine exhaust (no air vanes).

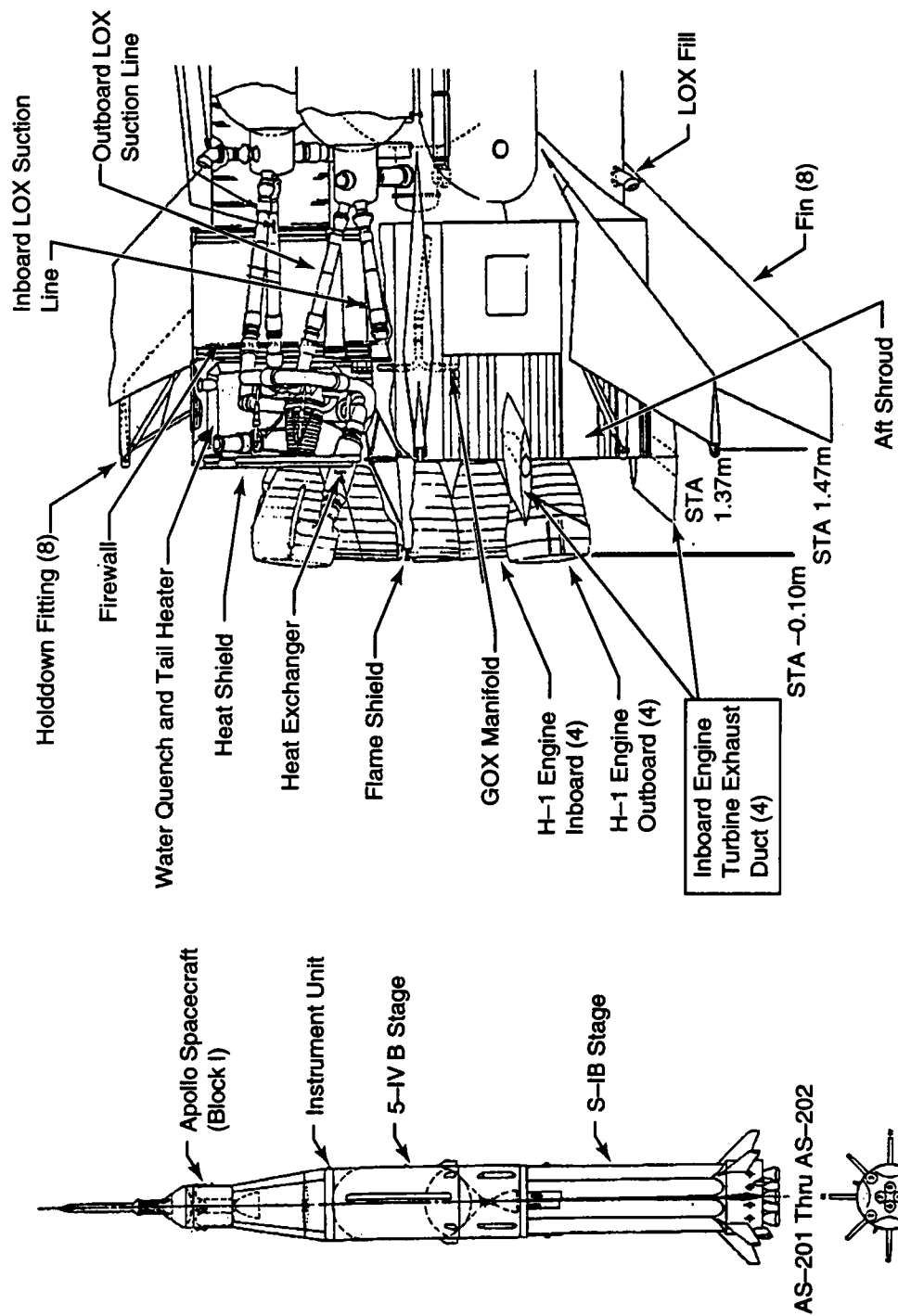


Figure 19. Saturn I-B base exhaust configuration.

## 2.7 Saturn V

The experimental LV of this research is very similar to the Saturn V in many respects. In the design of the 1.5 stage NLS the Saturn V was used for comparisons. Thirteen flights were made on the Saturn V between 1967 and 1973. These missions included Apollo 4 through Apollo 11, the first manned lunar orbital flight (20 h in lunar orbit), the first manned CSM/LM orbit rendezvous (10 days), and the Apollo 11 historic landing on the Moon on July 20, 1969.

Saturn V flights were used for the return of 21 kg of lunar samples from the first landing at the Sea of Tranquillity; the second manned lunar landing in November 1969 and the return of 34 kg from the Ocean of Storms; the third manned lunar landing January 1971 and the return of 43 kg from the Fra Mauro; the fourth manned lunar landing July 1971 and the return of 77 kg from Hadley Appennines; the fifth manned lunar landing April 1972 and the return of 97 kg from Descates; the sixth manned lunar landing December 1972 and the return of 117 kg from Taurus Littrow; and the launch of the Skylab space station in May of 1973. [23]

Figure 18 compares the aerodynamic surface planform shapes and sizes used in three of the Saturn LV's. While these surfaces provided flight stability, they did not move and provided no control. The flight control augmentors designed in this research are all-movable and not only enhance the flight stability but provide the required control augmentation.

The Saturn V stood 365-ft tall, had a diameter of 33 ft, and was a three-stage LV that incorporated 300 ft<sup>2</sup> of aerodynamic surfaces during all of its missions (see Figs. 20 and 21). The Saturn V used five F-1 engines with the center engine fixed and the four outer engines gimbaled. Lift-off weight was 6,228,000 lb and the total thrust was 7,500,000 lb. Propellants for the first stage were 3,307,855 lb of LOX and 1,426,069 lb of RP-1 (kerosene). The second stage was powered by five J-2 engines, each producing



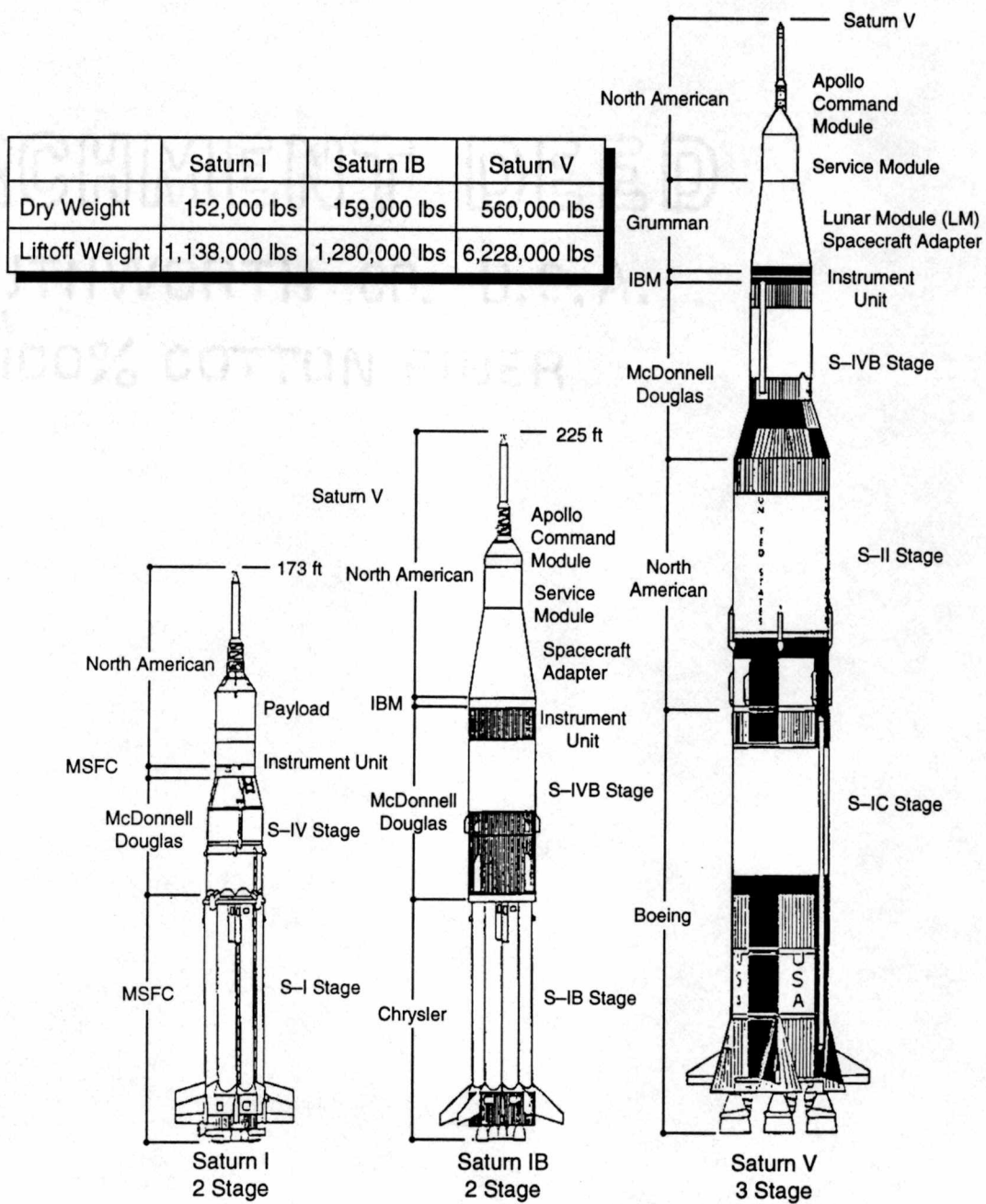


Figure 20. Comparisons of Saturn configurations.

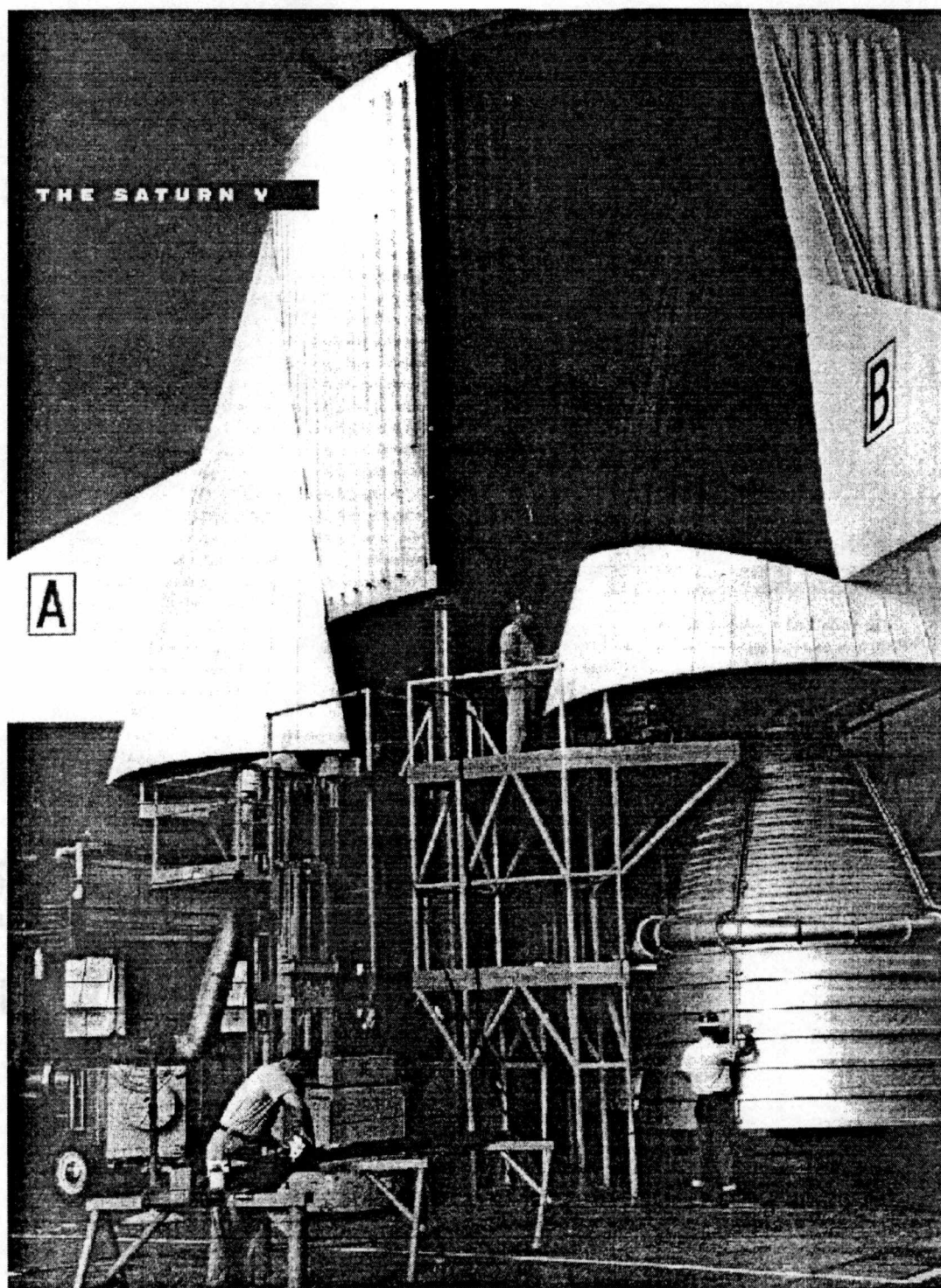


Figure 21. Saturn V aerodynamic surfaces.

250,000 lb of thrust. Propellants for the second stage were 821,000 lb of LOX and 158,000 lb of LH<sub>2</sub>. The third stage was powered by one J-2 engine. The Saturn V payload capability was 250,000 lb. [26]

The first stage of the Saturn V included 24 tons of thrust structure for the five F-1 engines, each of 1,500,000 lb of thrust. Four engine shrouds were attached to the thrust structure and fitted over each of the four outboard engines to smooth the air flow. Four aerodynamic surfaces of 75 ft<sup>2</sup> each were used at the base of the first stage for flight stability and were attached to the engine shrouds. The engine shrouds housed the engine actuator support structures and the eight RCS jets, which generated 86,600 lb of force for  $\frac{2}{3}$  second, to blow off the shrouds and separate the first stage at an altitude of 205,000 ft. The shrouds were 15° cone halves and were constructed of aluminum. The four aerodynamic surfaces were titanium covered. [26]

The gimbled engines of the five-engine cluster of F-1 engines used a spherical universal joint gimbal consisting of a socket-type bearing with a bonded Teflon™-fiberglass insert that provided a low-friction bearing surface. The first stage engines were gimbled 6° in pitch and yaw for TVC. The second-stage outboard engines were gimbled 7° in pitch and yaw. The third stage used a single gimbled J-2 engine for pitch and yaw control and RCS for roll control.

Aerodynamic vibration suppressors can be used to reduce or eliminate wind-induced oscillation on the pad. An unusual LV aerodynamic surface of note on the Saturn V was a set of helical strakes (Fig. 22). These were proposed to control the wind-induced oscillations due to vortex shedding as the vehicle was transported to the launch pad and while waiting on the launch pad. When a circular cylindrical structure, such as a LV on the pad, is exposed to winds, vortices are shed alternately from the sides of the fuselage, which cause alternating lift forces to act upon the sides. Under certain conditions, large oscillations can occur. Increased vehicle slenderness ratio results in increased wind-

induced oscillations on the pad due to vortex shedding. Wind tunnel tests have shown that helical strakes not only prevent wind-induced oscillations, but also damp out vibrations from other sources. The helical strakes attached to the Saturn V fuselage were among the proposed solutions. [27]

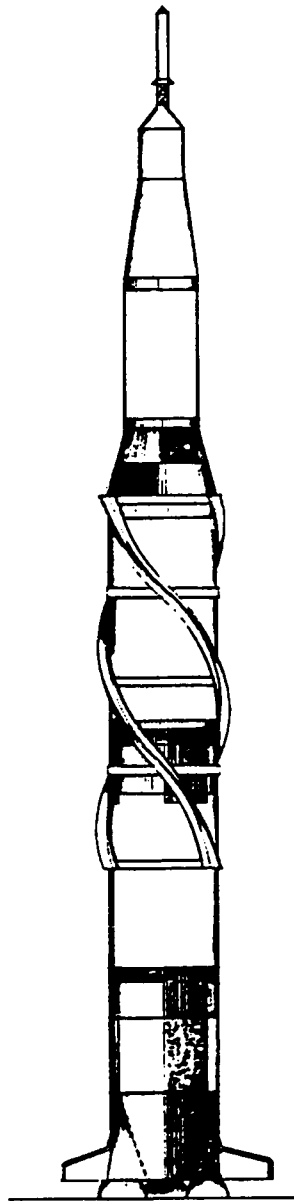


Figure 22. Saturn V helical strakes.

## 2.8 NASA's SCOUT

The SCOUT LV was the first U.S. LV to use exclusively solid propellant. SCOUT is an acronym for solid controlled orbital utility test. It was designed for NASA space probes and orbital and reentry research payloads. Flights began in 1960 and have continued to the present with well over 110 flights. The standard vehicle is a four-stage LV 75.1 ft in length, 45 inches in diameter, and producing 104.5 klb of thrust. SCOUT has launched a variety of payloads from the launch sites of Vandenberg Air Force Base, Wallops Island, and San Marco off the coast of Kenya. Primary operations are low-Earth orbit (LEO) and geosynchronous transfer orbit (GTO) to all inclinations. SCOUT launches up to 600 lb of payload to LEO and 120 lb to GTO. There are five SCOUT LV's in use by NASA and the Department of Defense.

SCOUT employs four aerodynamic surfaces for flight stability, and four aerodynamic flight controls at the tips of these (see Fig. 23). At lift-off, the LV is aerodynamically stable. A proportional control system, featuring a combination of internal jet vanes and aerodynamic tip flight controls operated by hydraulic servo actuators, is used to control the vehicle throughout the entire first-stage burning period. The jet vanes provide the majority of the control torque during thrusting. The aerodynamic flight controls provide all the control after burnout of the first stage. The second and third stages use RCS for control. The second stage uses four 500-lb and four 40-lb thrusters. The third stage uses four 60-lb and four 14-lb thrusters. The fourth stage uses a combination of four impulse spin motors to spin stabilize the payload. [28]

600 lb to LEO:  
Aerodynamic Surfaces;  
Internal Jet Vanes;  
Tip Flight Controls

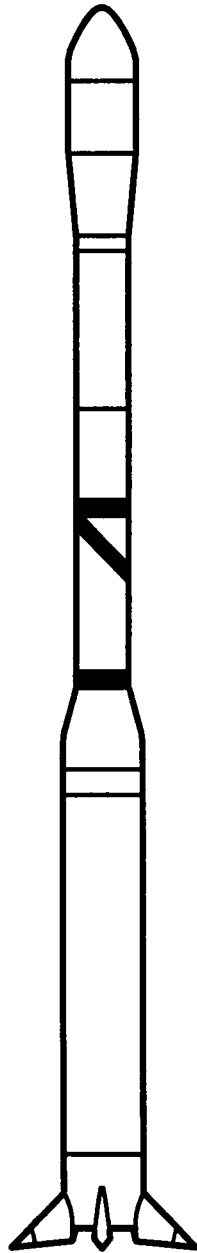


Figure 23. SCOUT.

## 2.9 Enhanced SCOUT

The Enhanced SCOUT is 75.1 ft in length, 45 inches in diameter, uses two strap-on solid rockets of 42 inches in diameter each, and generates 397.9 klb of thrust (see Fig. 24). The Enhanced SCOUT takes 1,160 lb of payload to LEO or 240 lb to GTO. The Enhanced SCOUT also uses four triangular aerodynamic surfaces and four triangular tip flight controls, along with internal jet vanes. In development beyond the Enhanced SCOUT is the SCOUT II, which will use a higher energy fourth stage. The SCOUT II will provide up to 12% more performance for a low polar orbit than the Enhanced SCOUT and may add two additional strap-ons. The SCOUT II will also use four aerodynamic surfaces for flight stability and four tip flight controls and internal jet vanes for control. [28]

1,160 lb to LEO:  
Aerodynamic Surfaces;  
Internal Jet Vanes;  
Tip Flight Controls

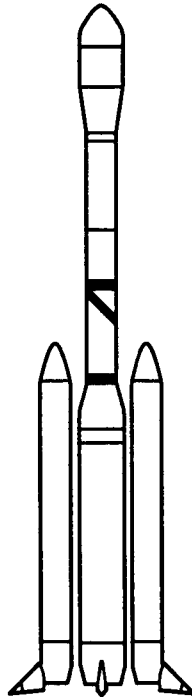


Figure 24. Enhanced SCOUT.

## 2.10 Pegasus

Pegasus is a three-stage, solid-propellant, air-launched vehicle capable of launching 1,000 lb to LEO. It is launched from an altitude of 44,000 ft at Mach 0.8. Pegasus flights began in 1990. It uses a 20-ft span, triangular, double-wedge airfoil wing (Scaled Composites, Inc.) with a LE sweep of 45°. Both aerodynamic surfaces for flight stability and aerodynamic flight controls are used (Fig. 25). [28] The first-stage thrust is 109,420 lb, second-stage thrust is 27,600 lb, and third-stage thrust is 7,770 lb. The first stage uses three hydraulic aerodynamic flight controls for pitch and yaw and three aerodynamic flight controls for roll control. The second and third stages use RCS for flight control.

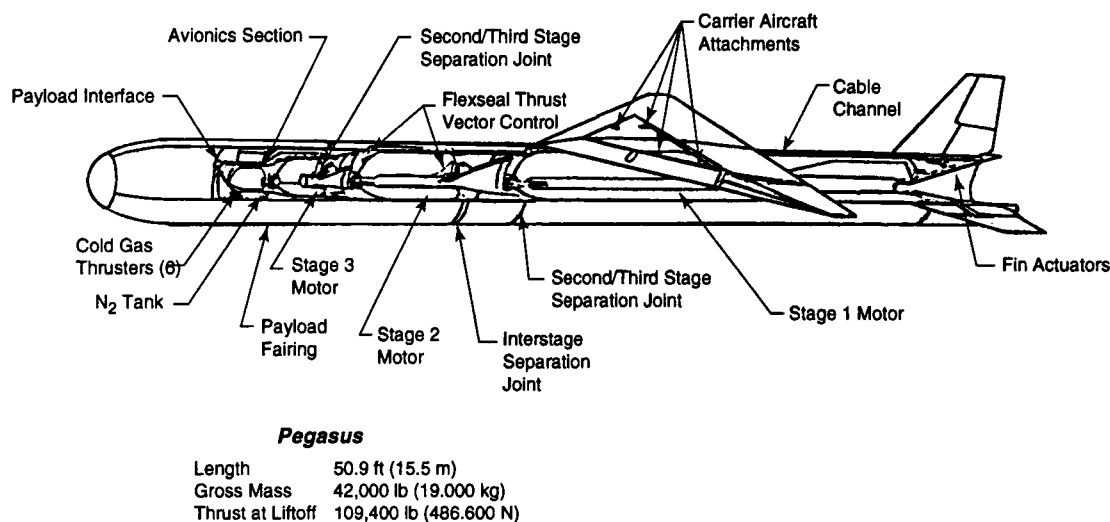


Figure 25. Pegasus.

The aerodynamic surfaces and aerodynamic flight controls are fabricated of lightweight graphite composite material. Thermal protection of the wing during hypervelocity is achieved by selectively applied additional layers of graphite composite, which are allowed to char and ablate. The flight control LE's are protected with Korothersm, while artificial cork is used over less critical areas of the wing and fuselage.



The flight control actuators are electromechanical and part of the “smart” actuator and sensor assemblies. Microprocessors manage the actuators and include initialization, self-test and health reporting via telemetry. The Pegasus wing and set of aerodynamic flight controls are mounted on the first stage. They aerodynamically provide pitch, yaw, and roll control during the first-stage powered flight and the coast period after burnout. Maximum dynamic pressure is  $1,000 \text{ lb/ft}^2$ , after which the first-stage burnout occurs at 200,000 ft of altitude and Mach 8.1. Second-stage ignition is at 213,000 ft. [28]

The second and third stages use engine nozzle gimbaling and RCS for control. The RCS can also provide up to 1,000 lb-in-s impulse for payload spin-up. Pegasus is launched from the same B-52 aircraft that performed the numerous X-15 drops. Pegasus is similar in size and shape to the X-15.

This review has shown that although a small percentage, we have a rich national heritage of LV's that have used aerodynamic surfaces for both flight stability and flight control. The Saturn program, which landed us on the Moon, included three basic vehicles, all of which used aerodynamic surfaces: the Saturn I, Saturn I-B, and Saturn V. There were 32 Saturn launches in all. The Saturn V took us to the Moon with  $300 \text{ ft}^2$  of aerodynamic surfaces during all of its 13 missions. The current day NASA SCOUT series also uses aerodynamic surfaces.

### CHAPTER III. THE X-15 AND SATURN SERIES AERODYNAMIC SURFACE DESIGN CONCEPTS

This chapter further demonstrates that the method of design was to test a great many configurations of aerodynamic surfaces. The most important developments in multiple-aerodynamic-surface flight vehicles made during the period from the end of World War II through the 1960's were based primarily on experience gained through wind tunnel test and flight test. Analytical techniques were not able to account for the multiple interference effects, including shock interference, present in the corolla of eight or twelve aerodynamic surfaces positioned at the base of a LV of the Saturn type such as shown in Figure 16 in the previous chapter. Only in the late 1960's was there enough test data available for preliminary design semiempirical computer codes to make noticeable appearances. [29]

The focus of this chapter is the aerodynamic surface design details of the Saturn series and the X-15. Not only the 32 Saturn flights, but also the 199 X-15 flights are being reevaluated today for the lessons learned, the programs, and the design concepts directly applicable to future LV's. Recent reports on aspects of the X-15 applicable to future LV's and of the X-15 being an important source of hypersonic stability and control data have been published by both NASA Dryden and NASA LaRc. [30, 31] After much wind tunnel testing, the Saturn I flew with eight aerodynamic surfaces of 121.2 ft<sup>2</sup> each, the Saturn IB with eight surfaces of 53.5 ft<sup>2</sup> each, and the Saturn V with four surfaces of 75 ft<sup>2</sup> each. The X-15 diamond airfoils were tested at many thickness ratios and taper ratios. The longitudinal control effectiveness increased by 30% using the all-movable wedge horizontal surfaces over the NACA 66-005 airfoil.

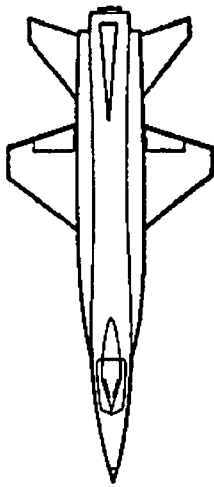
### 3.1 The X-15

Several distinctive aspects of the X-15 which relate it to the work of this research include: the cruciform empennage, the absence of ailerons, the all-movable horizontal empennage surfaces with capability to move both symmetrically and differentially, the blunt TE's, the ejection of an aerodynamic surface after use, the LOX propellant, and the availability of wind tunnel test data on the fuselage and cruciform empennage alone.

As mentioned previously, the current-day Pegasus LV is similar in size and shape to the X-15 and is launched from the very same B-52 aircraft that performed numerous X-15 drops. The X-15 flew to an altitude of 354,200 ft, reached Mach 6.7, and made almost 200 flights over a 9-year period. The X-15 was part LV, with an RCS and a thermal protection system, and was designed to explore the hypersonic speed regime and near-space environment. It was actually the second stage of a two-stage system. The first-stage was the B-52 which released it at an altitude of 45,000 ft and Mach 0.8. (The Dyna-Soar X-20 follow-on was to reach Mach 18 but was canceled.) [32]

The X-15 rocket engine had almost as much thrust as the Redstone LV and was also comparable in performance to the Redstone. The X-15 had 60,000 lb of thrust and the Redstone had 76,000 lb. The Redstone reached a maximum speed of 5,180 mi/h and 116 mi altitude on the Mercury flight, while the X-15 achieved 4,520 mi/h and 67 mi altitude. (The X-15 theoretically had the performance to reach 100-mi altitudes but could not safely reenter from there.) The maximum dynamic pressure recorded was 2,027 lb/ft<sup>2</sup> and the maximum temperature measured on the vehicle was 1,323 °F. The X-15 used an XLR99 rocket engine and 33,000 lb of propellants (1,000 gal of LOX and 1,400 gal of anhydrous ammonia). The outer skin of the fuselage was the exterior wall of the propellant tanks. Figure 26 [32] shows the configuration of the three X-15's that were built and Figure 27 [32] shows the proposed delta-wing X-15 that is similar to current reusable LV configurations.

# **X-15** **Three View** **Specifications**



Length.....50 ft

Wing Span.....22 ft

Height.....13 ft

Wing Area.....200 Square ft

Sweep  $C_{1/4}$ .....25°

Weight at Launching.....31,275 lbs

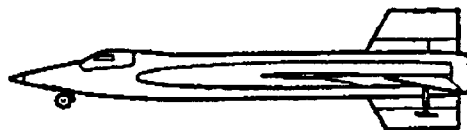
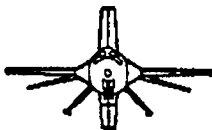


Figure 26. The X-15.

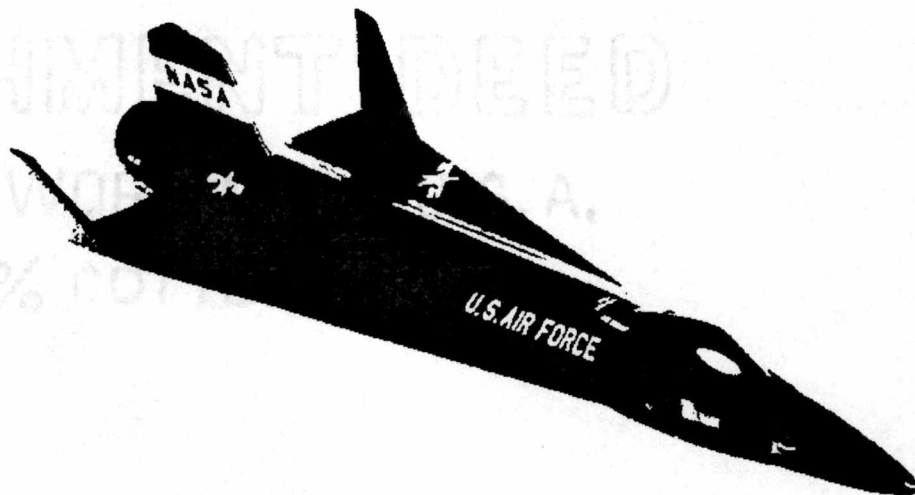


Figure 27. Delta-winged X-15.

The empennage was unusual, with four aerodynamic flight control surfaces that provided both stability and control, two all-movable NACA 66-005 airfoil horizontal stabilizers with  $15^\circ$  anhedral that could be operated differentially, and two  $10^\circ$  wedge airfoil section vertical stabilizer-rudders, one above and one below, with 55% for the dorsal fin and 45% for the ventral fin. The lower rudder extended below the landing gear and had to be jettisoned before landing. The vertical wedge-shaped surfaces had blunt TE's. This was a solution to the hypersonic directional stability problems. (Chuck Yeager in the X-1 program found that the all-movable horizontal tail was needed to fly through Mach 1.) The X-15 empennage resembled cruciform missile fins and also the empennage of the experimental LV of this research. Wind tunnel test data are available for the X-15 horizontal stabilizer deflections without its trapezoidal wing and are compared to the LV of this research. The flight control surfaces were hydraulically actuated. No ailerons were used on the small wings that gave the vehicle an aspect ratio (A) of 2.5. Roll control was achieved by differential deflection of the two horizontal stabilizers. There were TE

flaps on the wing and four speed brake segments located on the rear portions of the upper and lower vertical fins. They opened in a V-shape to produce drag and some directional control.

The control system was a high-gain, high-authority, stability augmentation system (SAS) with advanced command augmentation. (A notch filter was chosen to stabilize the SAS loop for the 12- to 14-Hz resonance of the horizontal surface.) The vehicle was neutrally stable in ballistic flight. RCS jets were located in the nose and on the stub wings. Pitch and yaw RCS were located behind the nose sensor. There were two nozzles on the top of the nose pointing up and two on the bottom pointing down. Two yaw jets were located on each side of the nose. The roll jets were in the outboard wing panels, two on each wing.

The original vehicle configuration suffered at high angle-of-attack ( $\alpha$ ) at hypersonic speeds from loss of effectiveness of the upper vertical stabilizer and a large increase in effectiveness of the lower vertical stabilizer, leading to an undesirable high anhedral effect. The large ventral fin was necessary for controllability to cope with thrust misalignments. The vehicle could be controlled without SAS at high  $\alpha$  in spite of the anhedral, as shown in Figure 28. [31] Thus, the symmetrical tail was used. In flight, it was found unflyable at  $\alpha > 8^\circ$  without dampers due to the influence of secondary aerodynamic effects, such as trim on the stability derivatives, effects which were not included in the simulation. Space attitude control along with threshold aerodynamics were demonstrated. Transitions from RCS to aerodynamic controls were made, and a control mode in which the two systems were blended was developed. The X-15-3 had integrated aerodynamic controls and RCS, and automated three-axis stabilization. [31]

A great many different configurations of the X-15 aerodynamic surfaces and flight controls were wind tunnel tested at NASA LaRC. The airfoil sections tested

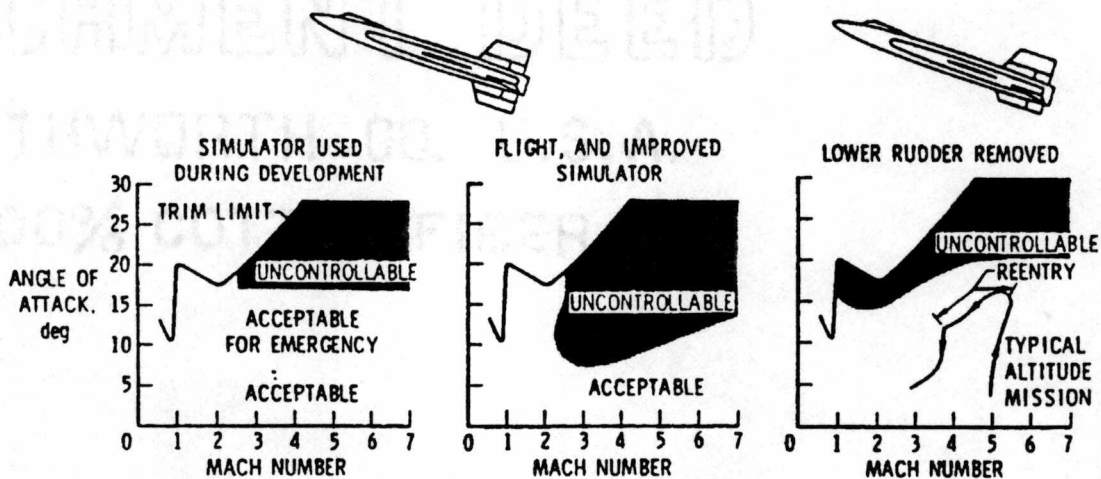


Figure 28. Handling characteristics of X-15 with dampers inoperative.

included symmetrical double wedges with thickness ratios of 5, 10, and 15% (Jordan, G.H., "X-15 Project," NASA LaRC, 1961). The taper ratios were varied on the all-movable horizontal stabilizers, which had a sweep angle of  $45^\circ$ . Three different designs were wind tunnel tested for flutter at the Mach ranges of 0.72 to 1.32 and 0.79 to 1.47. Full and semispan models were tested (Fig. 29 and 30), one of which used a 1% chord thickness at the TE. Required flutter margins were met. [33]

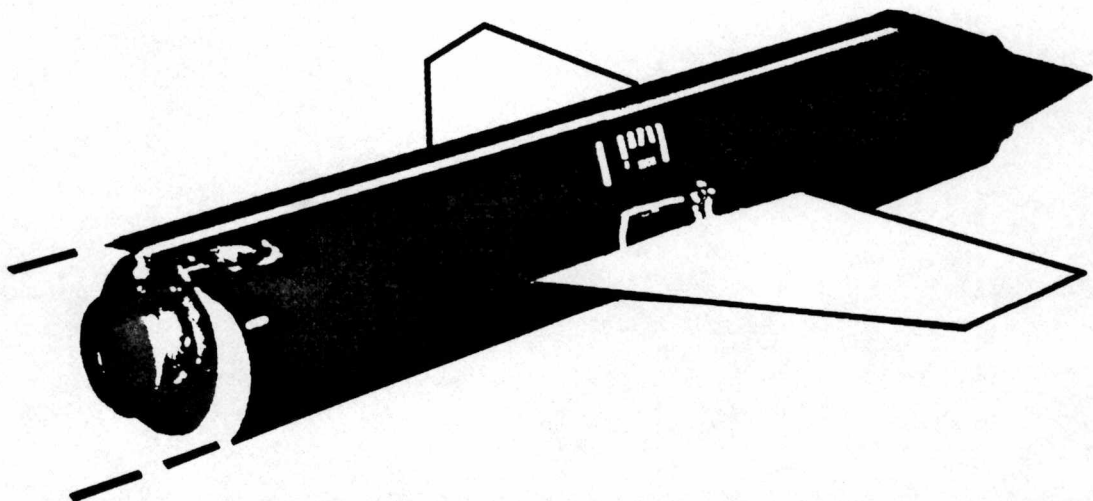


Figure 29. X-15 horizontal aerodynamic surface being wind tunnel tested.

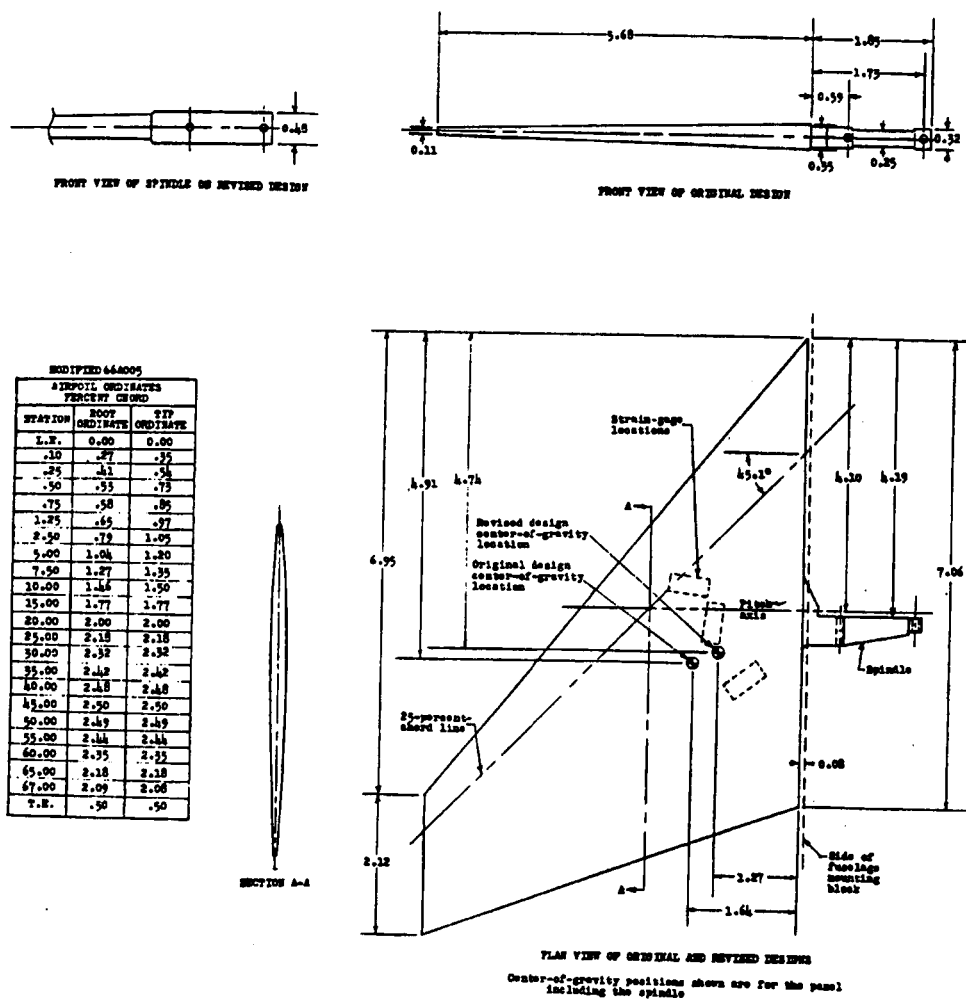


Figure 30. X-15 horizontal aerodynamic surface geometry.

Control characteristics of two preliminary X-15 models with two different vertical fins and two different horizontal fins were investigated at Mach numbers of 2.98 and 4.01 and Reynolds numbers ( $Re$ ) of  $2 \times 10^6$  to  $4 \times 10^6$  based on wing mean aerodynamic chord ( $\bar{c}$ ). Six-component data were obtained for  $\alpha$  from  $-4^\circ$  to  $22^\circ$  and for angle-of-sideslip ( $\beta$ ) from  $-5^\circ$  to  $1^\circ$ . When the NACA 66-005 airfoil section horizontal fins were replaced with  $10^\circ$ -wedge horizontal fins, the longitudinal control effectiveness ( $C_{m_\delta}$ ) increased by 30% (Fig. 31 and 32). [34] The all-movable controls led to increased interest



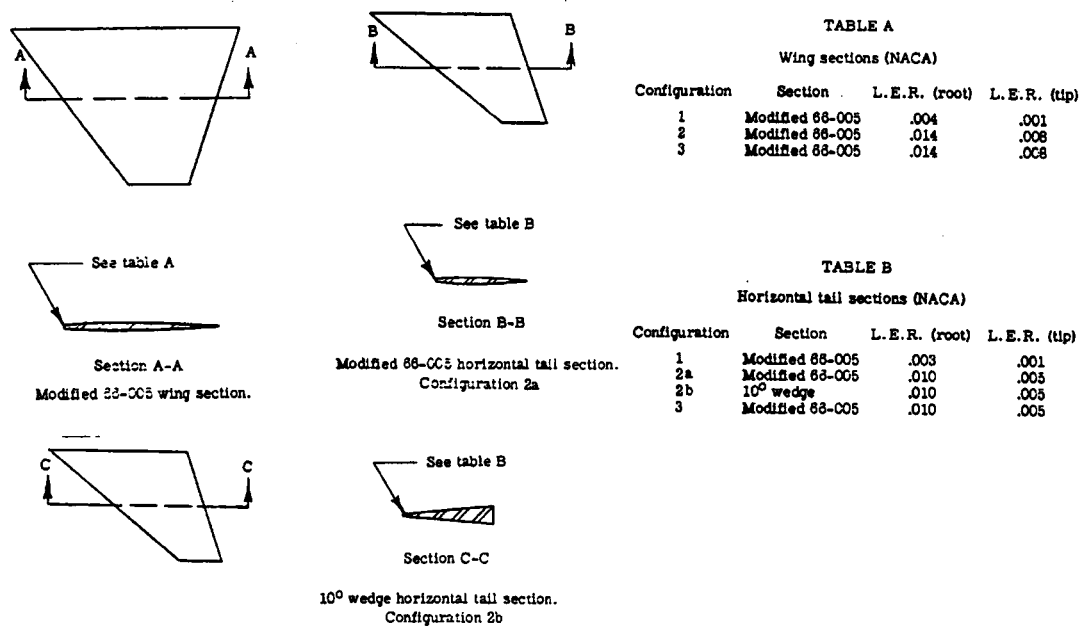


Figure 31. X-15 horizontal aerodynamic surfaces tested.

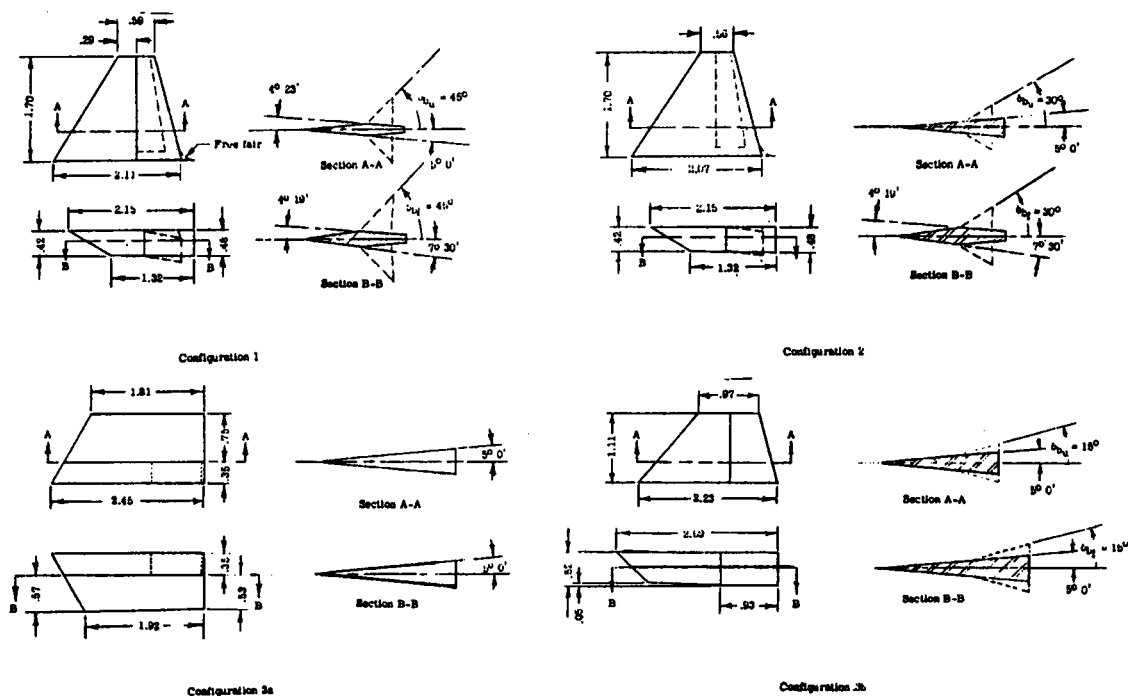


Figure 32. X-15 vertical aerodynamic surfaces tested.

in wind tunnel flutter tests. Several configurations, including a rectangular planform, were flutter tested on  $1/12$ -size dynamically and elastically scaled models of the all-movable horizontal surfaces at Mach 6.86 at the NASA LaRC 11-inch hypersonic tunnel. Five horizontal surface configurations and two vertical surface configurations were tested at Mach numbers of 1.64, 2.0, 3.0, and 4.0. No flutter problems were encountered. [35] Static longitudinal, lateral, and directional stability and control were investigated on many stabilizer configurations including both a sharp TE and a  $5^\circ$ -wedge TE, dihedral of  $0^\circ$  and  $-15^\circ$ , 60% dorsal fin and 40% ventral fin, vertical fin deflections of  $0^\circ$  and  $-5^\circ$ , and speed-brake deflections of  $20^\circ$ , at Mach 6.86 and Re of 640,000 based on  $\bar{c}$ , through  $\alpha$  ranges of  $-4^\circ$  to  $24^\circ$  and  $\beta$  ranges of  $0^\circ$  to  $-5^\circ$  (Figs. 33–35). [36] Nine alterations were made on the vertical stabilizer. In final configuration tests, the horizontal surface was deflected from  $-35^\circ$  to  $15^\circ$ , the vertical surface from  $-7.5^\circ$  to  $0^\circ$ , and the speed brake from  $0^\circ$  to  $50^\circ$  (Penland, NASA LaRC; 1960).

### 3.2 Saturn I

The Saturn I Block II finally flew with eight aerodynamic surfaces consisting of four large fins and four stub fins. Before this design was chosen, many combinations and configurations were wind tunnel tested. Figure 36 shows the eight fins of 121.2 ft<sup>2</sup>. Figure 37 shows the eight stub fins. Figure 38 shows the eight fins of 181.9 ft<sup>2</sup>. Figure 39 shows the dimensions of the 181.9 ft<sup>2</sup> fins. Figure 40 shows the dimensions of the stub fins. Figures 41 and 42 show the full fin. Figure 43 shows a combination of Saturn I aerodynamic surfaces being tested in the wind tunnel at NASA MSFC. [37]

The final combination and planform area that flew was: four large fins of 121.2 ft<sup>2</sup> each, and four stub fins of 53.3 ft<sup>2</sup> each. Total planform area of these surfaces was 698 ft<sup>2</sup>, span was 42.833 ft, and fin  $\bar{c}$  was 14.2778 ft. All the fins had blunt TE's. [37]

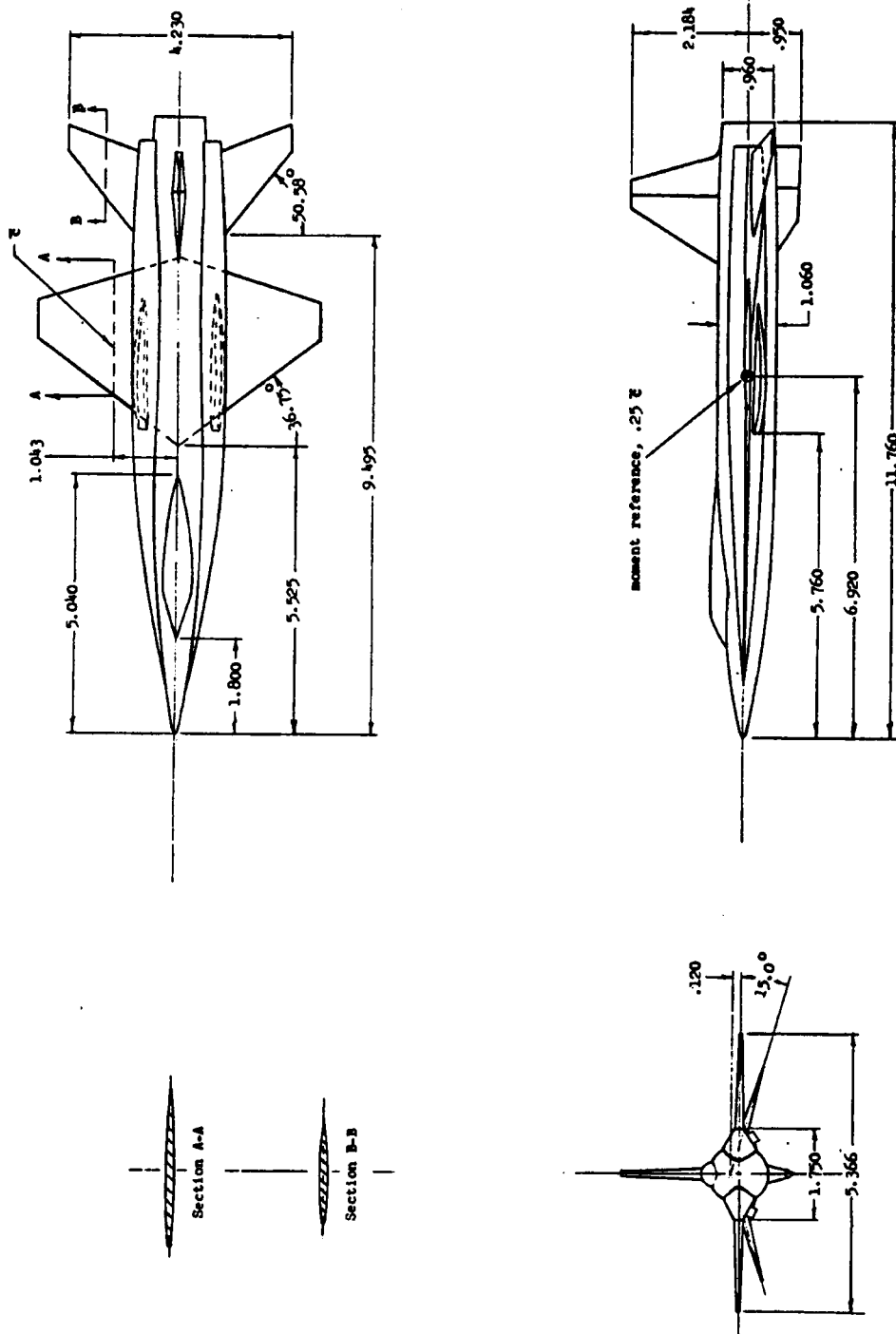


Figure 33. Details of X-15 configuration 1 tested (BWXXHV<sub>0</sub>V<sub>L</sub>), all dimensions are in inches.

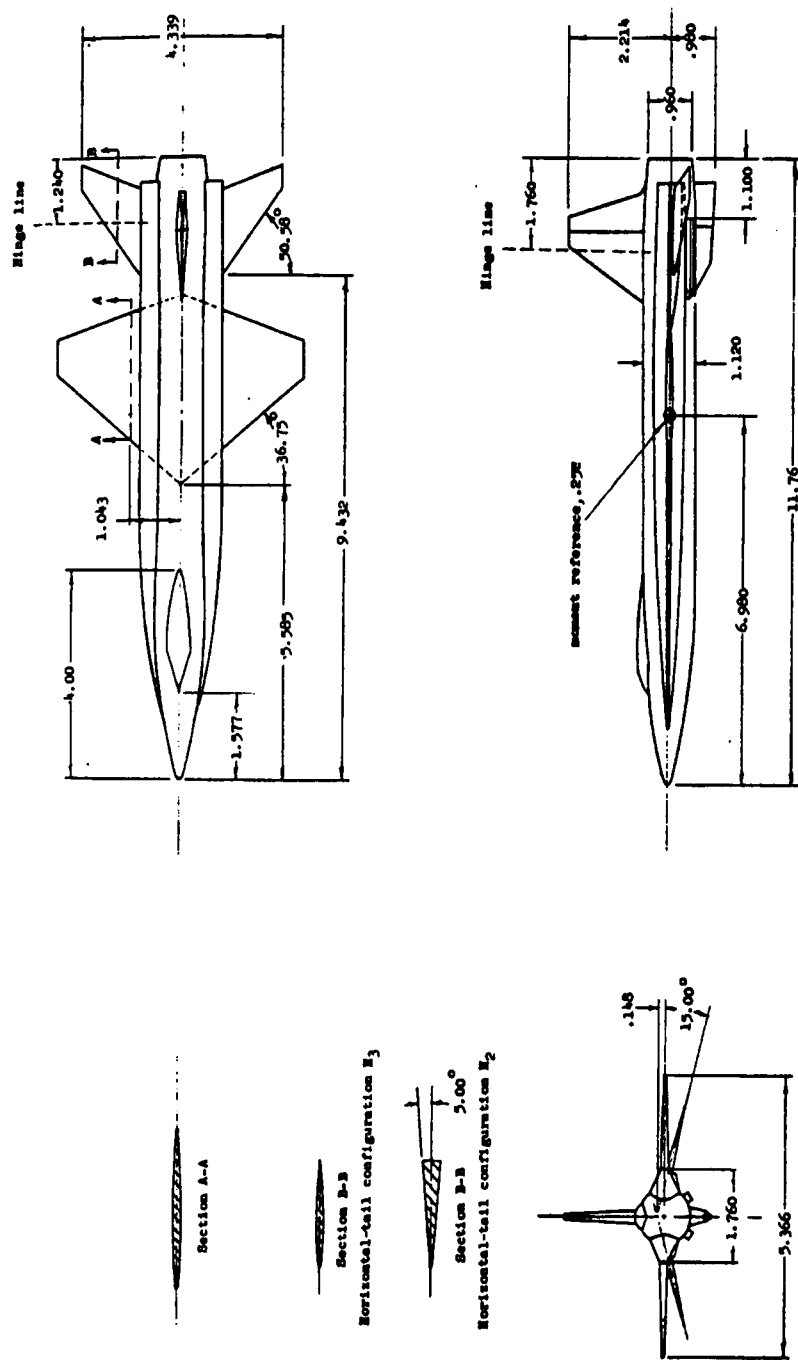


Figure 34. Details of X-15 configuration 2 tested ( $B_2 W_2 X_3 H_3 V_{U2} V_L$ ), all dimensions are in inches.

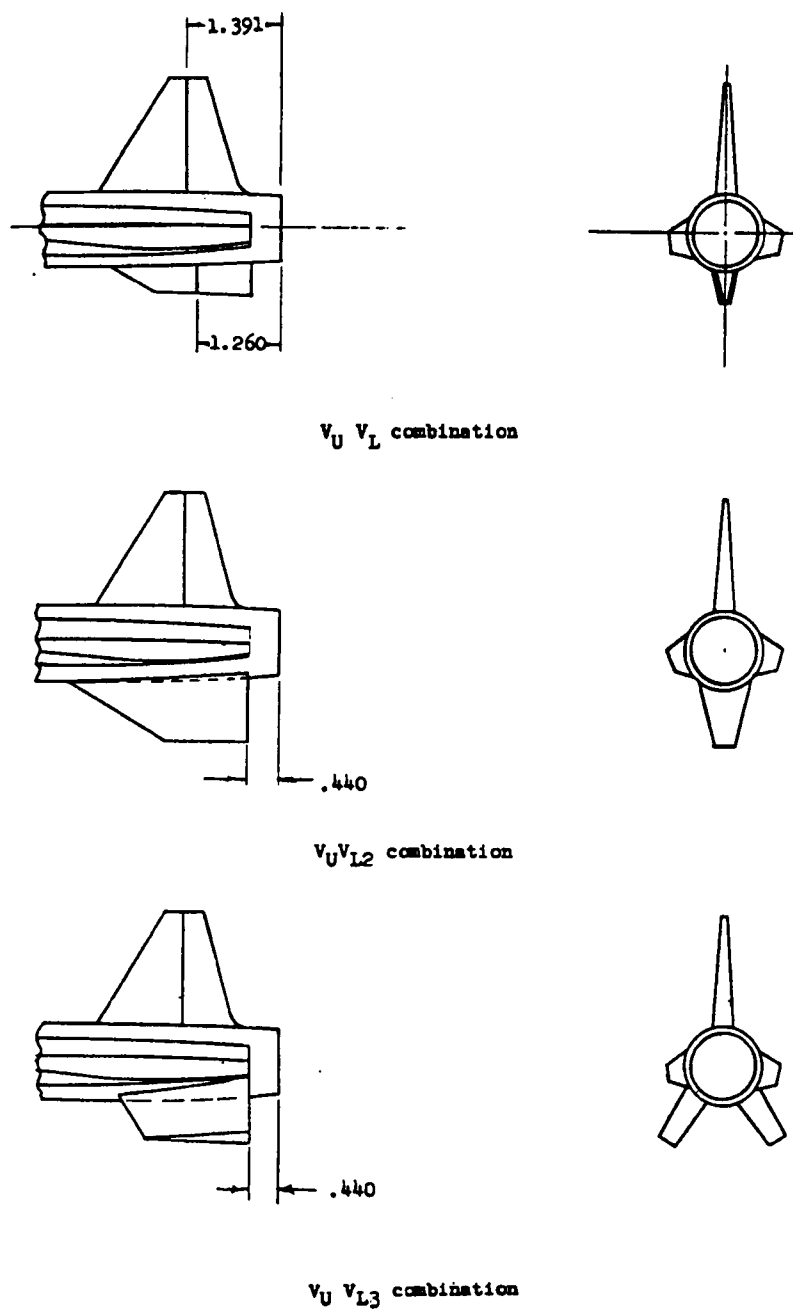


Figure 35. X-15 vertical aerodynamic surface positions.

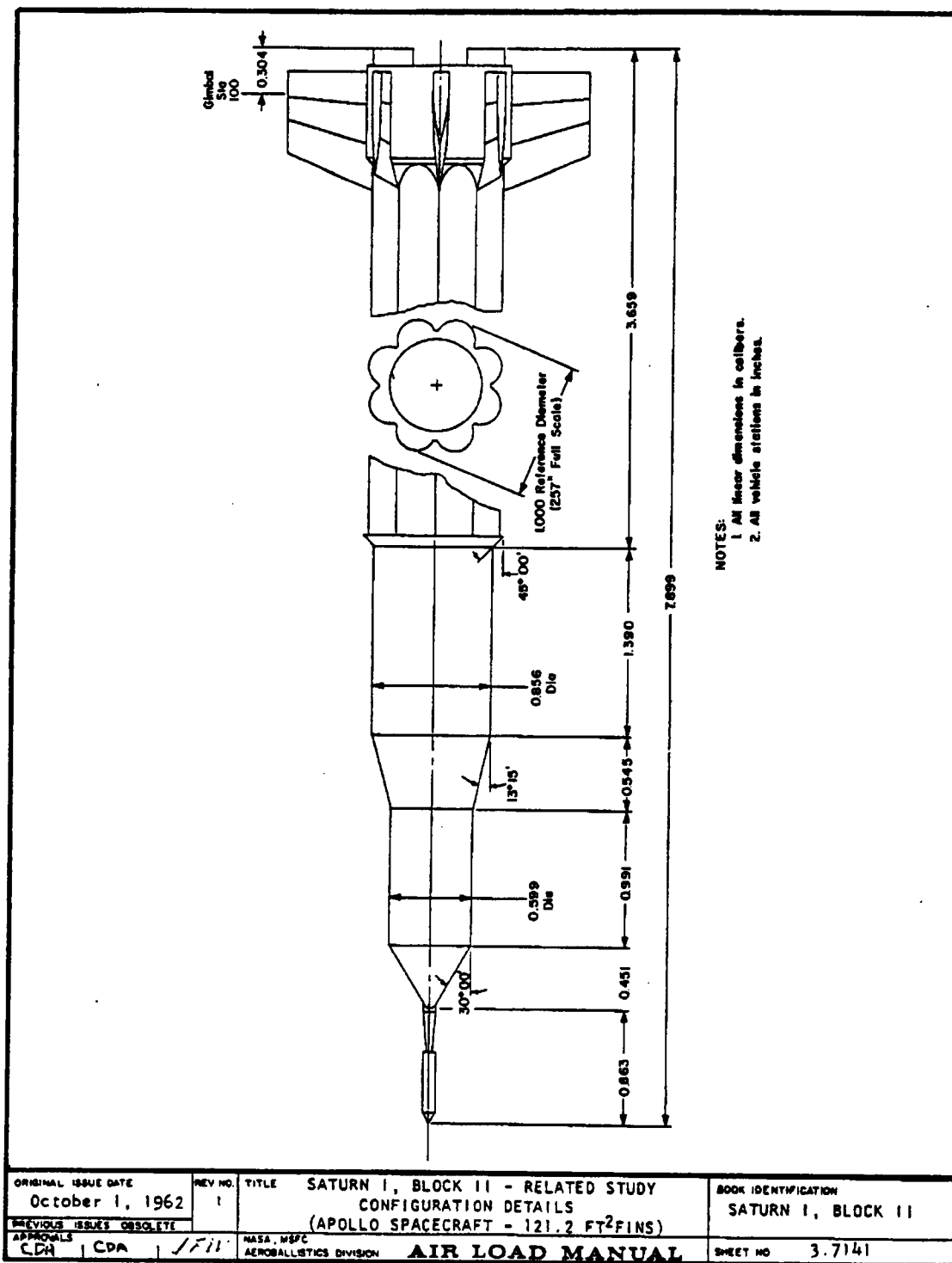


Figure 36. Saturn I geometry with 121.2 ft<sup>2</sup> aerodynamic surfaces.

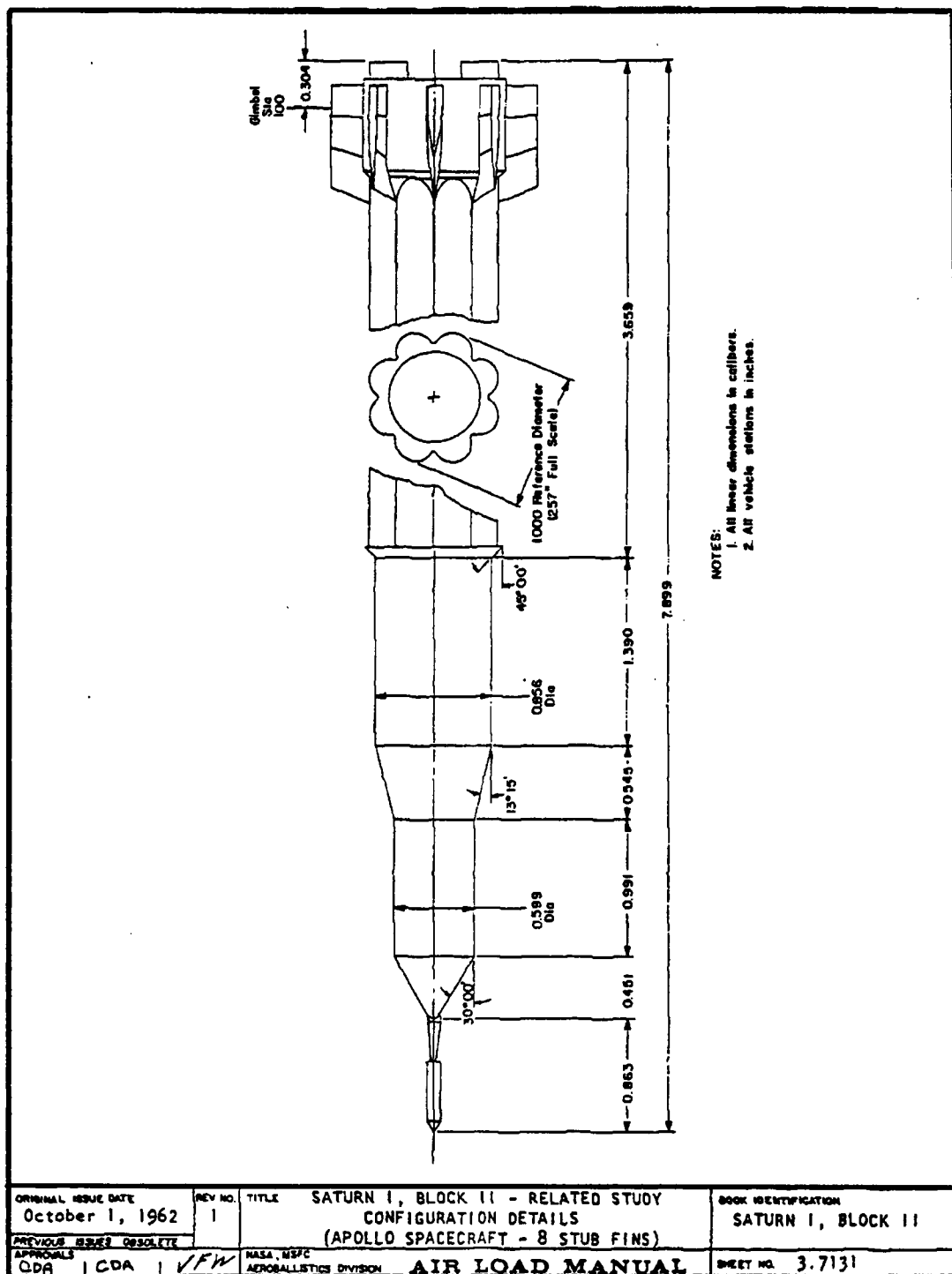


Figure 37. Saturn I geometry with eight stub fins.

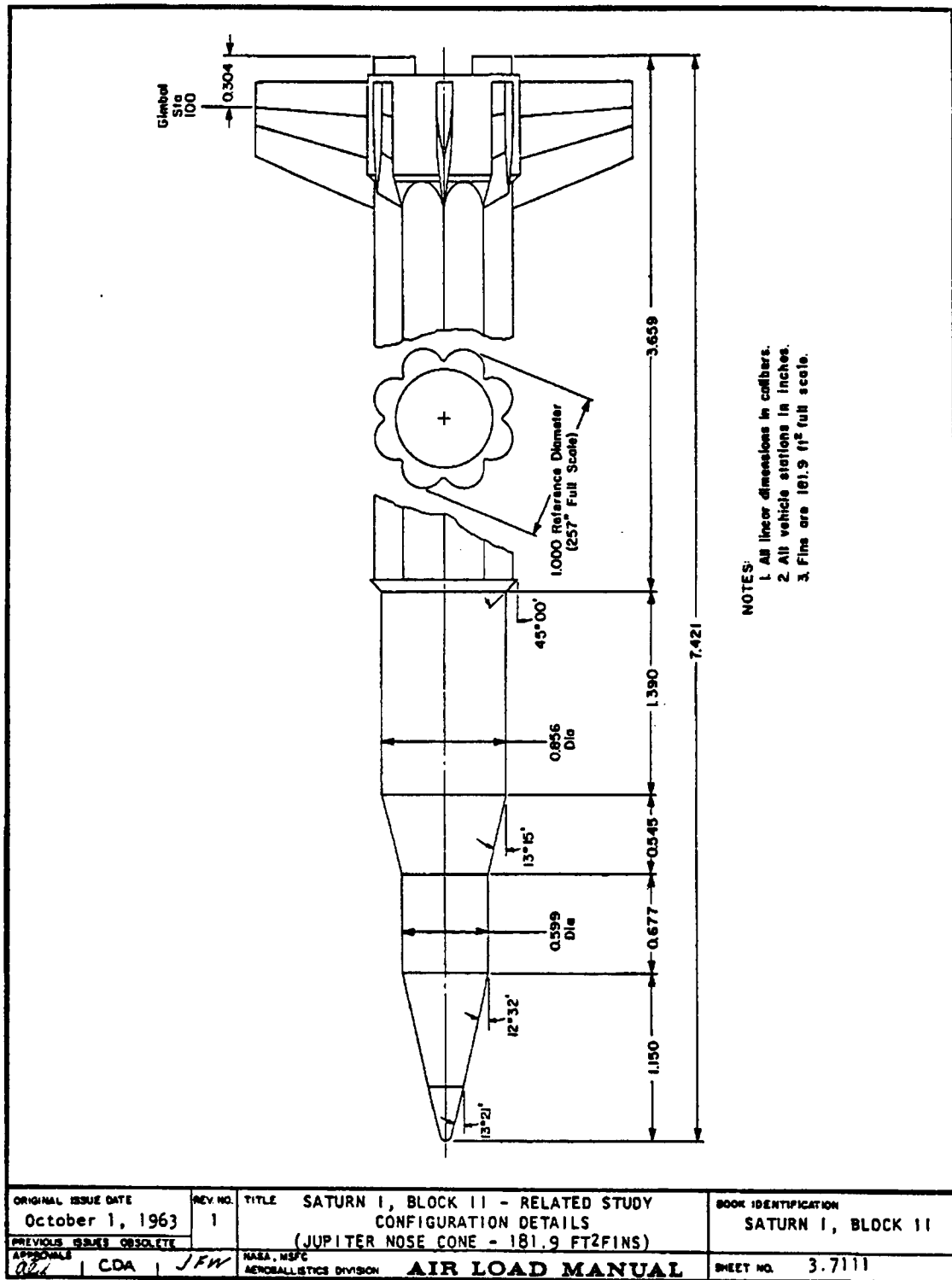


Figure 38. Saturn I geometry with 181.9 ft<sup>2</sup> aerodynamic surfaces.



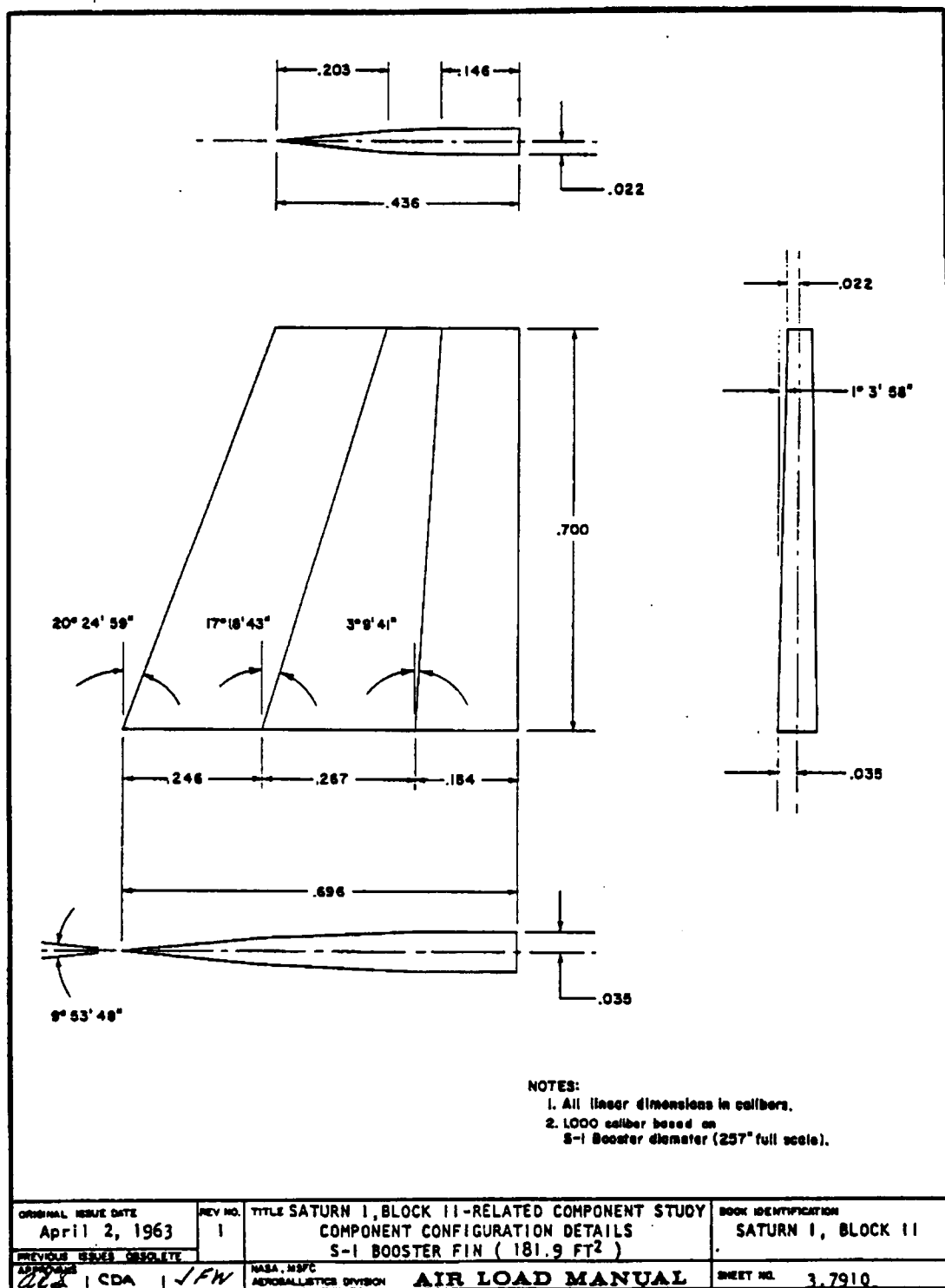


Figure 39. Saturn I 181.9 ft<sup>2</sup> fin detail.

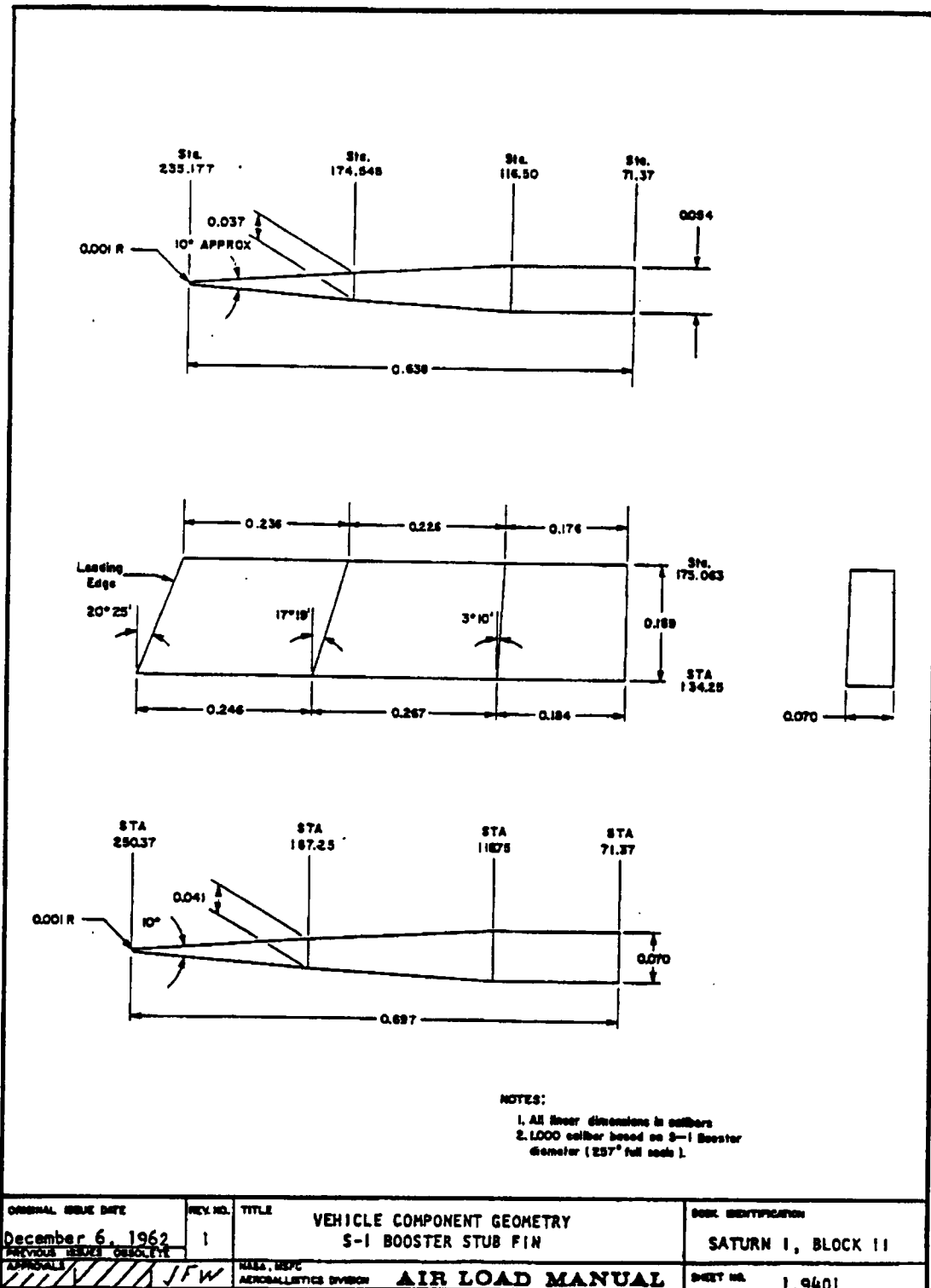


Figure 40. Saturn I stub fin detail.

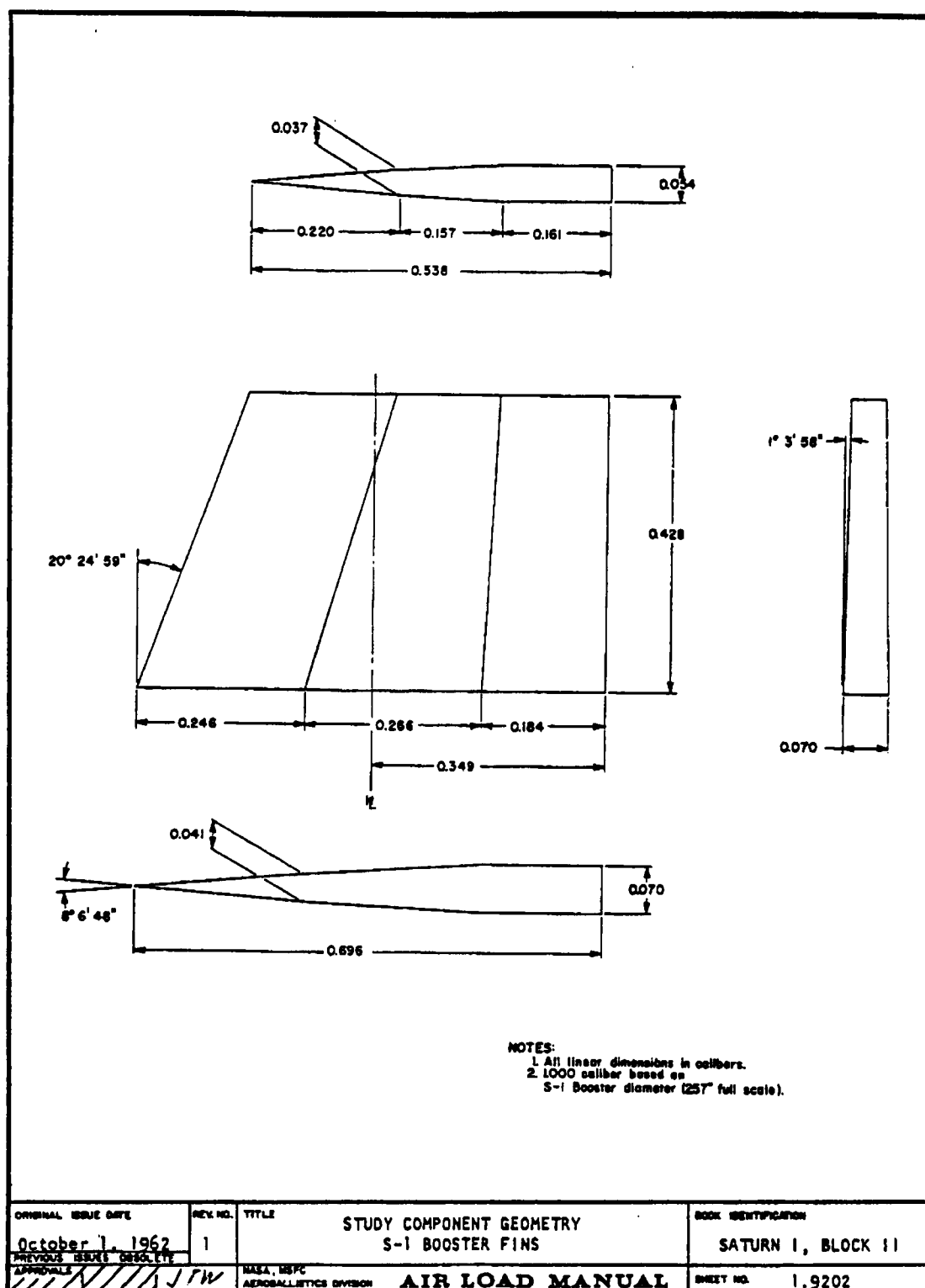


Figure 41. Saturn I geometry with 121.2 ft<sup>2</sup> fin detail.

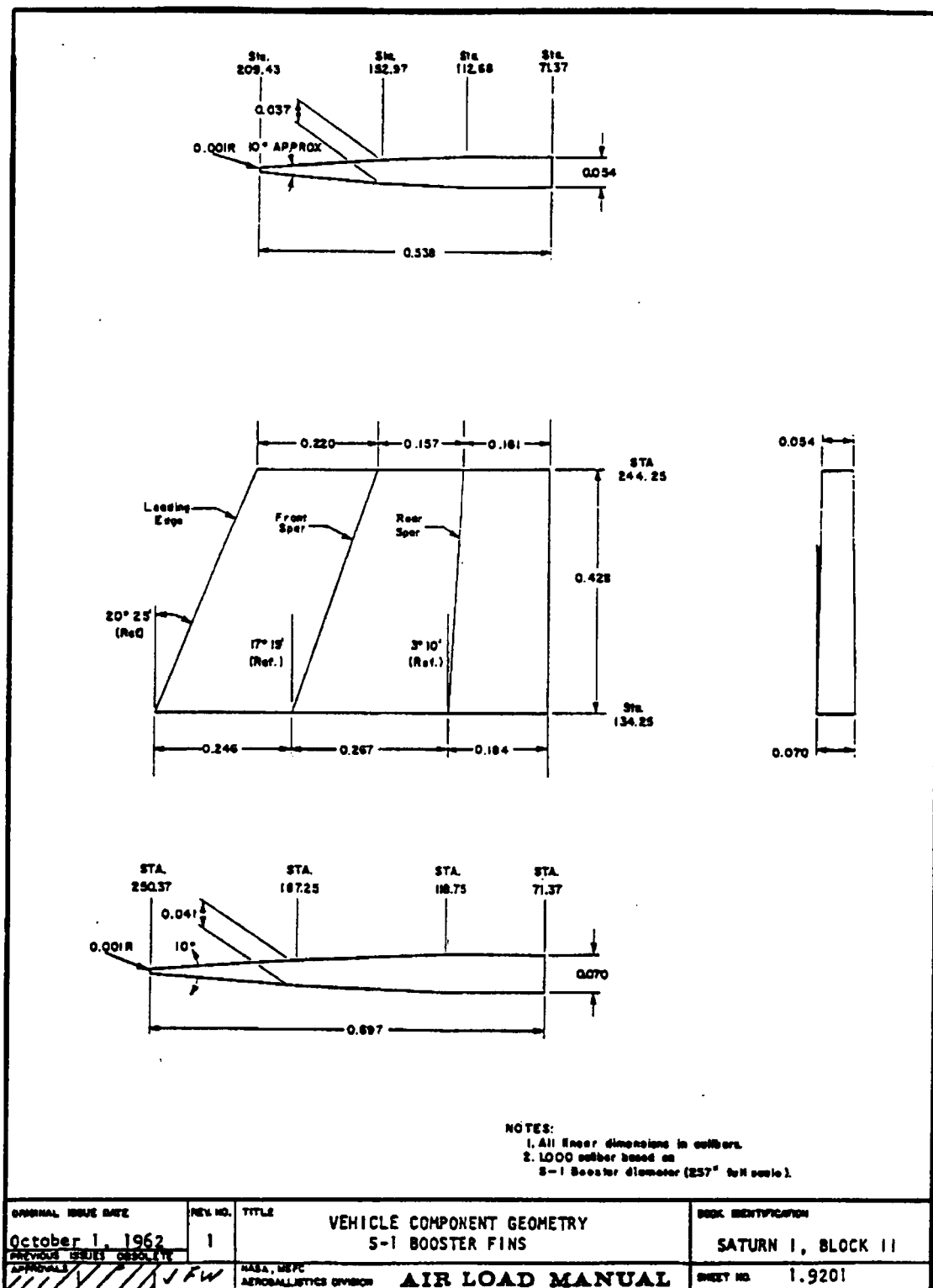


Figure 42. Station numbers of 121.2 ft<sup>2</sup> fin.



Figure 43. Saturn I being wind tunnel tested (AEDC Von Karman facility hypersonic tunnel B).

### 3.3 Saturn I-B

The Saturn I-B aerodynamic surfaces underwent extensive redesign and produced nearly 2.5 times the lift as the eight stubs of the previous designs, and weighed 1,800 lb less. Because of the long chord and short span, the eight stub fins on the Saturn I were very inefficient aerodynamic surfaces. The goal in the Saturn I-B fin redesign was to achieve the same lift with eight uniform fins of 53.3-ft<sup>2</sup> area each, i.e., the same lift with less planform area. The same thickness as used on the Saturn I was retained in order to use the same attach points on the LV as the old stub planforms, thereby keeping the surface redesign from affecting the basic LV design. The Saturn I-B fins also had blunt TE's. Figure 44 shows the 53.5-ft<sup>2</sup> area. [38,39] Figure 45 shows a 5% scale Saturn I-B being wind tunnel tested with the fin redesign.

The final design produced more lift and provided more flight stability transonically, where it was needed most, and less lift supersonically than the four full fins and four stub fins of the Saturn I. The eight uniform fins of the Saturn I-B had a 45° sweep of the LE's and a 24.4° sweep of the TE's. Figure 46 shows the internal structure of the Saturn I-B aerodynamic surface, and Table 1 [24] lists the pertinent detail

Table 1. Saturn I-B fin redesign.

| Item                    | Value                       |
|-------------------------|-----------------------------|
| Length, M.A.C.          | $\bar{c} = 74$ inches       |
| Location, M.A.C.        | $y = 47$ inches             |
| Leading Edge Sweep      | $\Lambda_{LE} = 45^\circ$   |
| Trailing Edge Sweep     | $\Lambda_{TE} = 24.5^\circ$ |
| Thickness Ratio         | $t/c = 17.25\%$             |
| Aspect Ratio            | $A = 3.14$                  |
| Planform Area           | $S = 53.5$ ft <sup>2</sup>  |
| Taper Ratio             | $c_{root}/c_{tip} = 2.50$   |
| Half Angle Leading Edge | $\theta = 5.5$              |
| Spanwise Taper Angle    | $\lambda = 2.88$            |

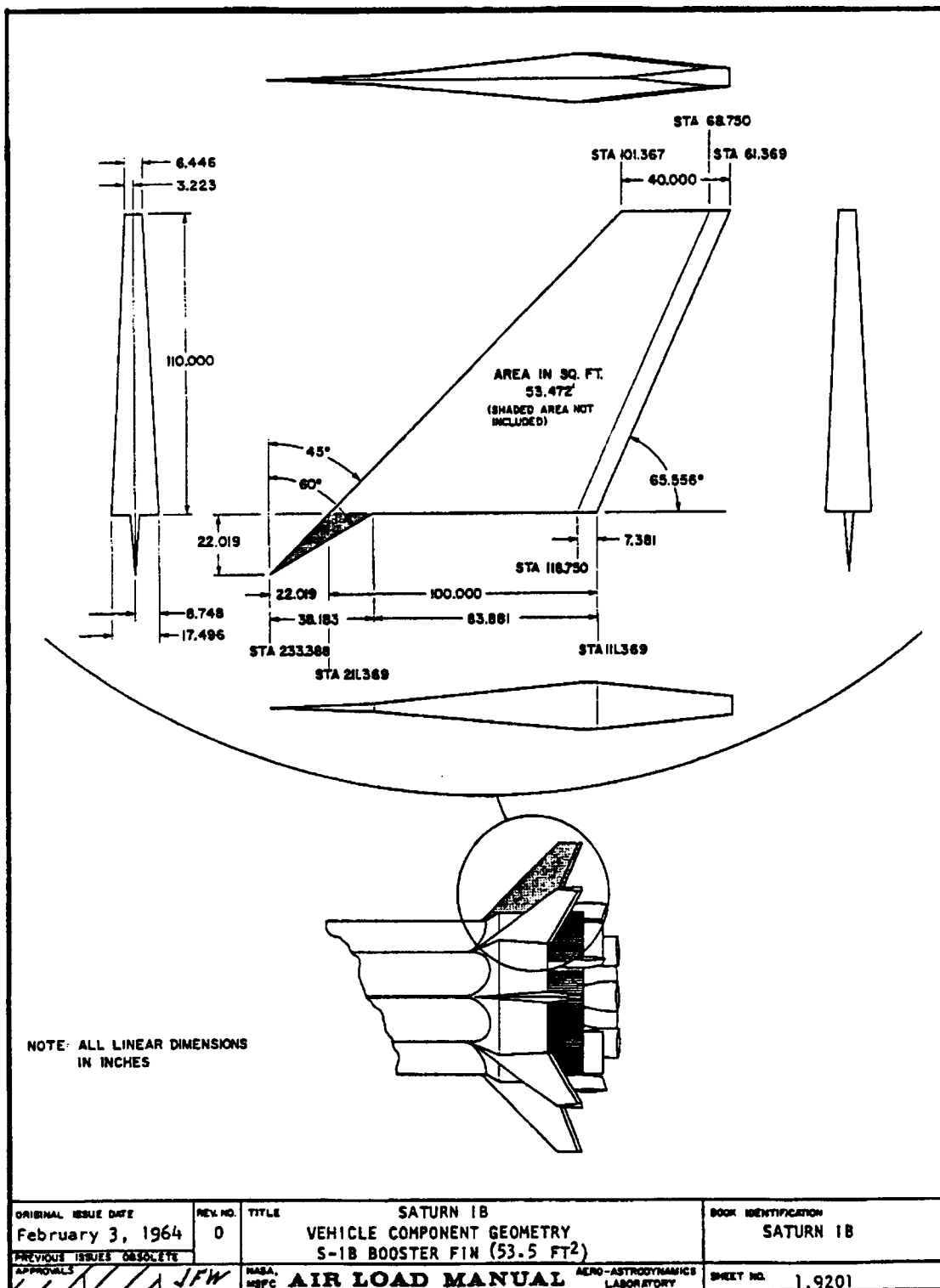


Figure 44. Saturn I-B aerodynamic surface geometry.

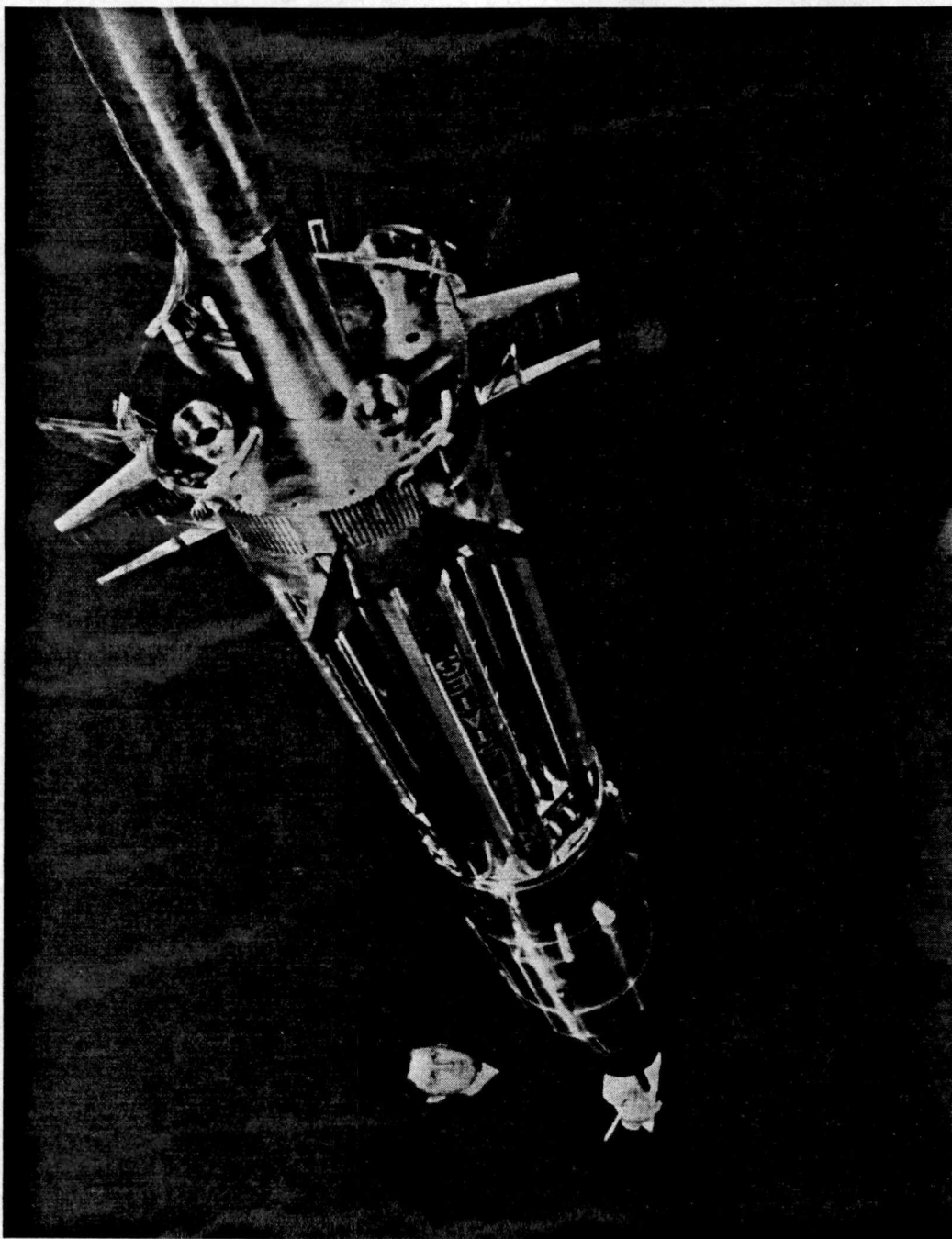


Figure 45. Saturn I-B (5% scale) in wind tunnel test.



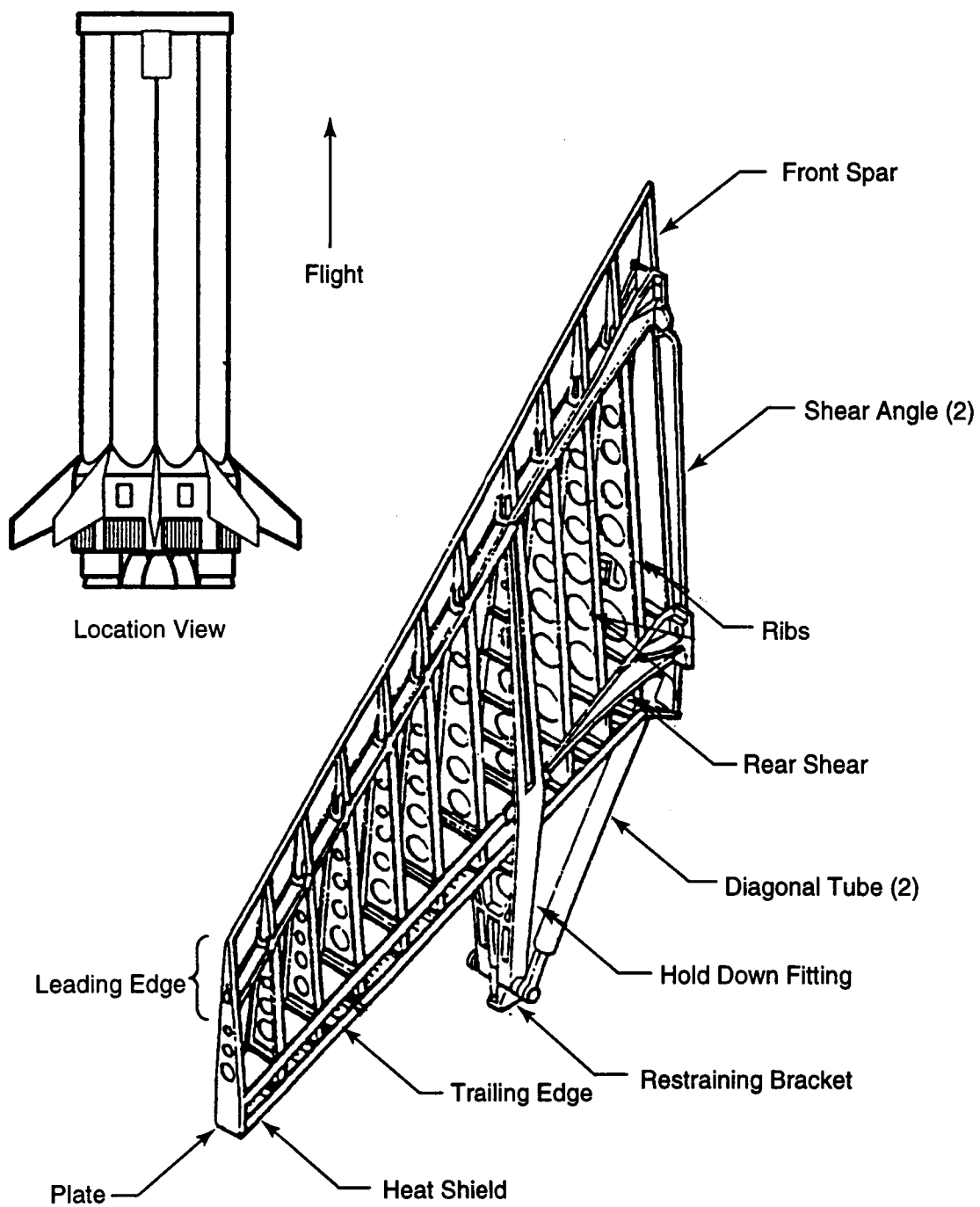


Figure 46. Saturn I-B aerodynamic surface internal structure.

geometry. The profile was a triangular tapered-wedge with a blunt TE. The redesigned Saturn I-B surfaces significantly increased the aerodynamic static flight stability of the vehicle and also increased the dynamic flight stability of the vehicle in pitch and yaw.

### **3.4 Saturn V**

The Saturn V used four aerodynamic surfaces of 75 ft<sup>2</sup> each to enhance flight stability (Fig. 47). [40] Each fin had a span of 139 inches, a root chord of 138 inches, a tip chord of 55 inches, an outboard chord depth of 4 inches, a root chord depth of 15 inches, a LE sweep angle of 30°, and a wedge angle of 10°. The surface was designed for a maximum dynamic pressure ( $Q_{max}$ ) of 1,440 lb/ft<sup>2</sup>. The fin assembly was mechanically attached to the engine fairing assembly (Fig. 48). The Saturn V aerodynamic surfaces were constructed of 7075-T6 aluminum and were titanium coated to withstand 2,000 °F heat from the engine exhausts. Figures 49, 50, and 51 show the engine shrouds to which the surfaces were attached. Total aerodynamic surface weight was 2,035.2 lb. [40, 41]

The fins and shrouds of the Saturn V were shown in the wind tunnel test to be more efficient when the LV was rolled 45°. Since the lift on the fin-shroud combination increased, the total vehicle center of pressure was moved farther aft, which further increased the Saturn V flight stability.

This coverage of the detailed aerodynamic surface design developments of the Saturn and X-15 series shows the considerable amount of design evolution.

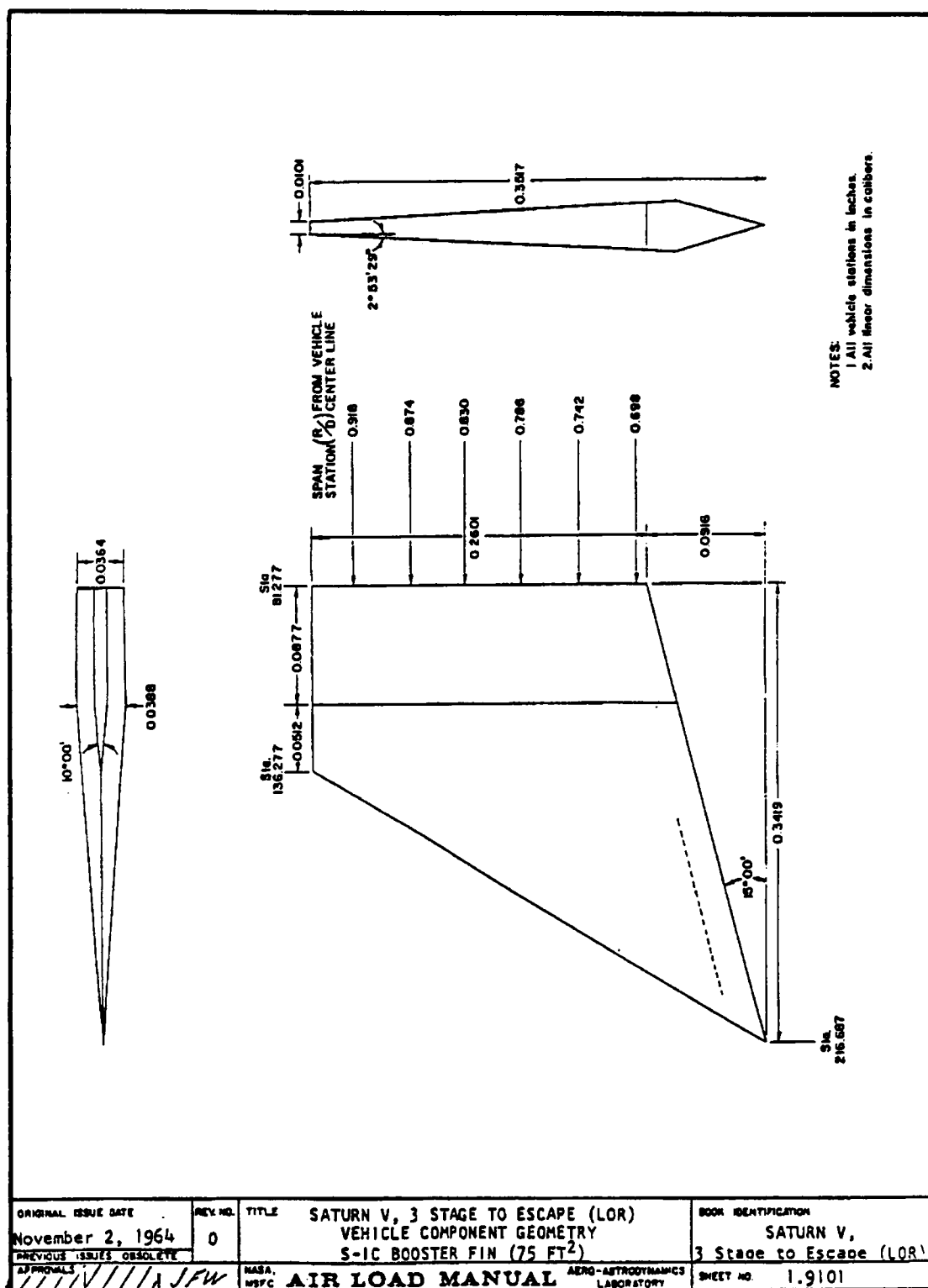
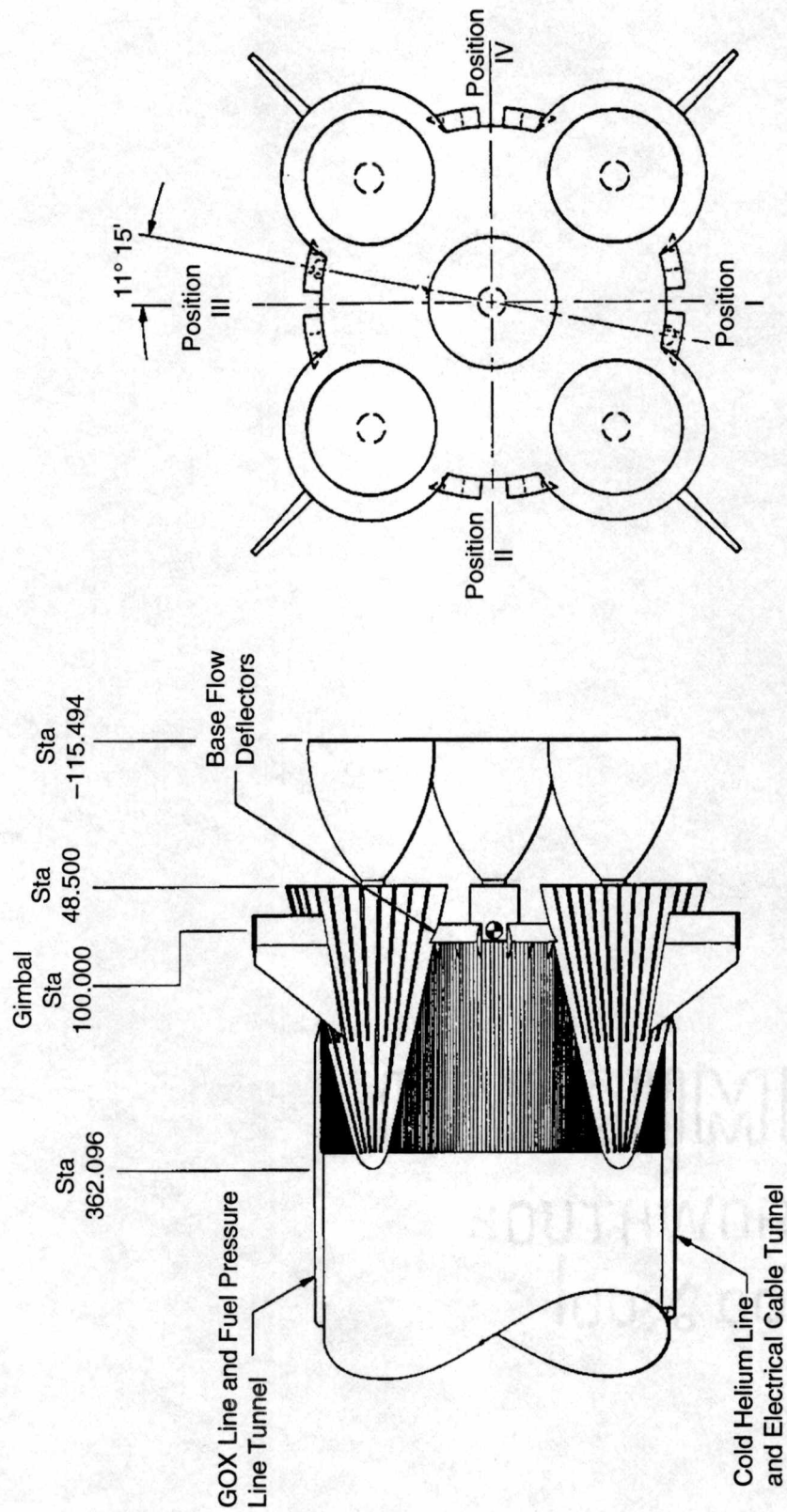


Figure 47. Saturn V aerodynamic surface geometry.



1. All Vehicle Stations in Inches.

Figure 48. Saturn V, S-IC stage, base area geometry.



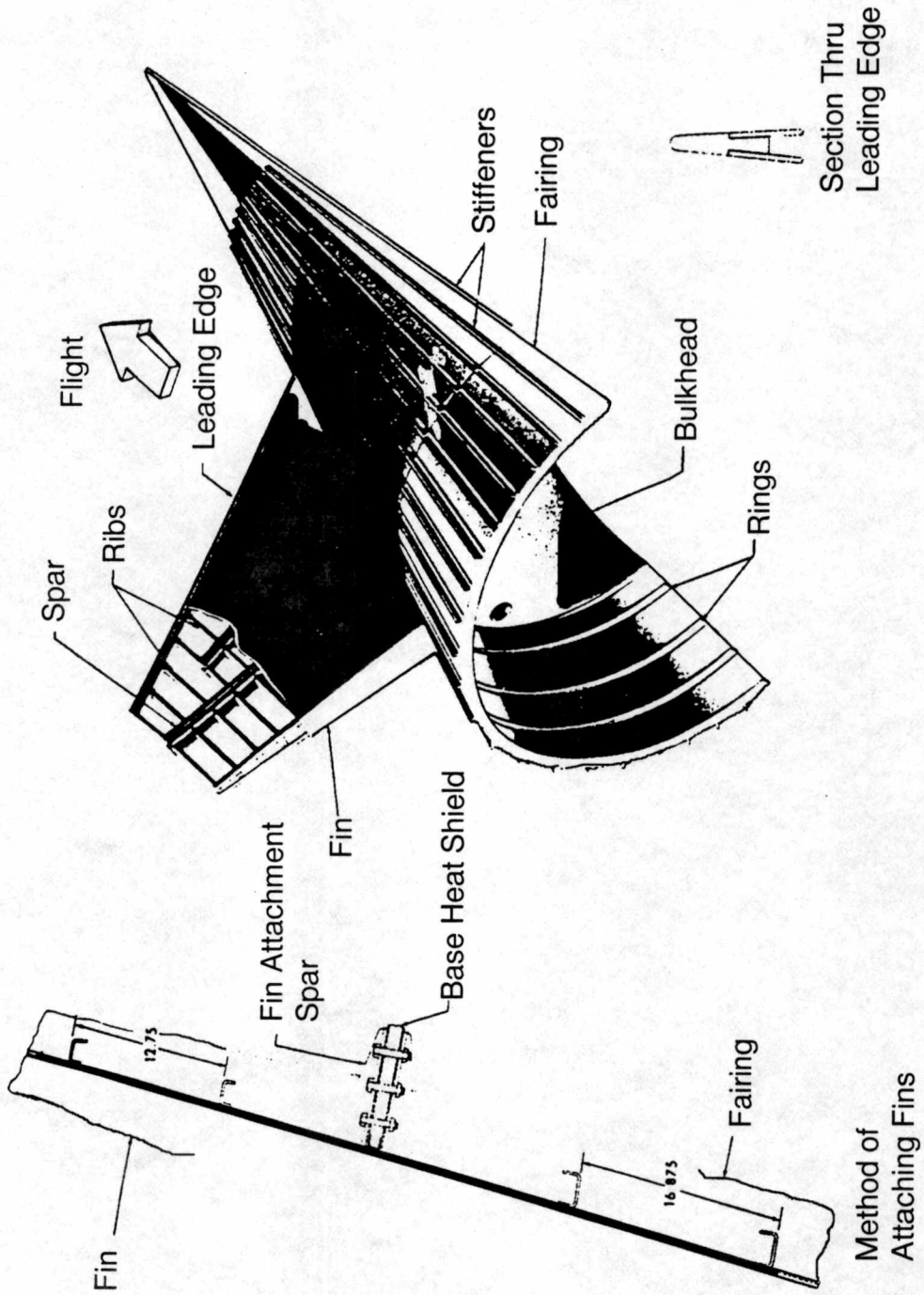


Figure 50. Saturn V aerodynamic surface and shroud assembly.

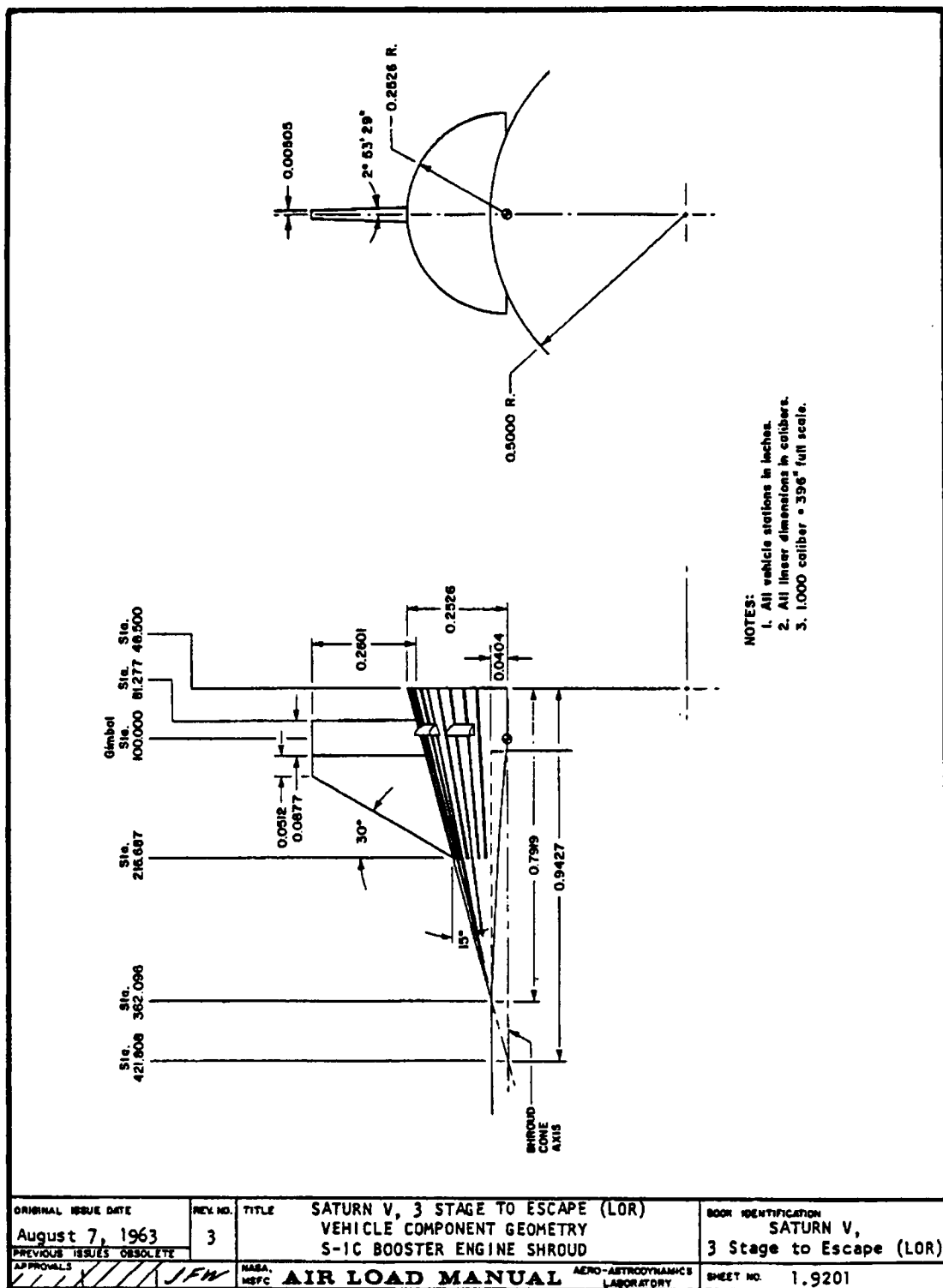


Figure 51. Saturn V shroud geometry.



## **CHAPTER IV. SUMMARY OF CURRENT USE OF AERODYNAMIC SURFACES BY FRENCH, RUSSIAN, CHINESE, AND JAPANESE LAUNCH VEHICLES**

While the last two chapters addressed what has been done before, this chapter addresses what is being done now. Although no longer used in the U.S., aerodynamic surfaces are currently being used on large LV's by many other nations. Today there are a myriad of LV's actively launched from nearly two dozen geographic launch sites around the globe. Space News recently listed (February 4, 1996) all of the world commercial heavy-lift LV's launched in 1995. Of this list 39% use aerodynamic surfaces. This chapter demonstrates the wide variety of foreign LV aerodynamic surface planform shapes in current use.

The ESA's Ariane 4 is one of the most active commercial LV's using aerodynamic surfaces today. It has launched 13 satellites in 1995 and hopes to surpass that record in 1996. President Clinton has approved three U.S. commercial satellites to be launched in Chinese Long March LV's which use aerodynamic surfaces. The Japanese M-5 is a 1995 LV which has been specially designed with aerodynamic surfaces that guarantee aerodynamic flight stability of the vehicle through Mach 2. Solid RCS are also attached to the tips of the fins for roll control. In the FCA designs of this research, a blunt TE flight control augmentor design could easily accommodate solid RCS in the same fashion for even more control. Only the most significant foreign usages of LV aerodynamic surfaces are summarized here.

### **4.1 French (ESA)**

Although many countries have provided funding, France has provided the largest funding for the Ariane series of LV's. The Ariane 1, Ariane 2, Ariane 3, Ariane 4 (or 40), Ariane 42L, Ariane 44L, and Hermes all use aerodynamic surfaces for flight stability. Hermes uses aerodynamic surfaces for both flight stability and flight control (Fig. 52). [42]



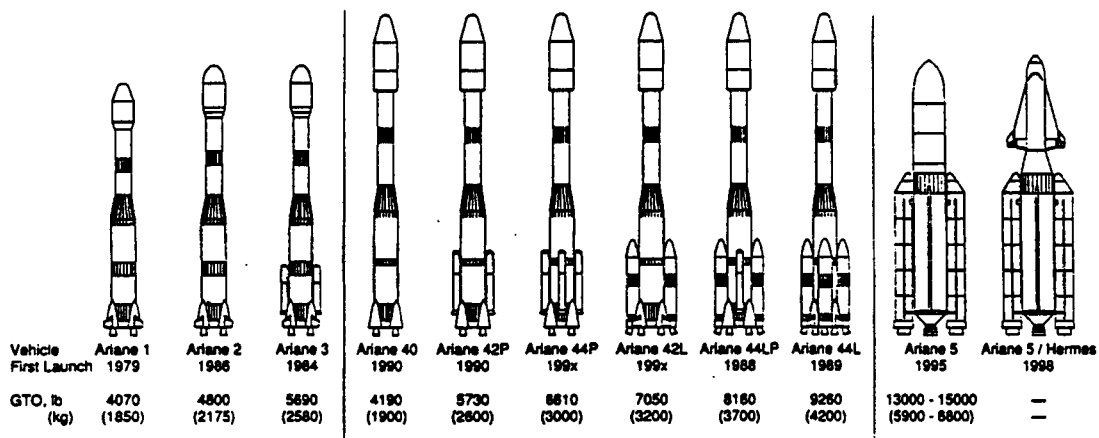


Figure 52. Ariane LV's.

The Ariane 4, first launched in 1988, uses four fins, 21 ft<sup>2</sup> each, to improve flight stability during atmospheric flight (Fig. 53). [43] Ariane 4 can launch 21,100 lb to LEO or 9,260 lb to GTO. Ariane 4 configurations include: Ariane 40 with no strap-on boosters, Ariane 42P with two solid propellant strap-on boosters, Ariane 44P with four solid strap-on boosters, Ariane 42L with two liquid propellant strap-on boosters, Ariane 44L with four liquid propellant strap-on boosters, and Ariane 44LP with two liquid and two solid propellant strap-on boosters. [42]

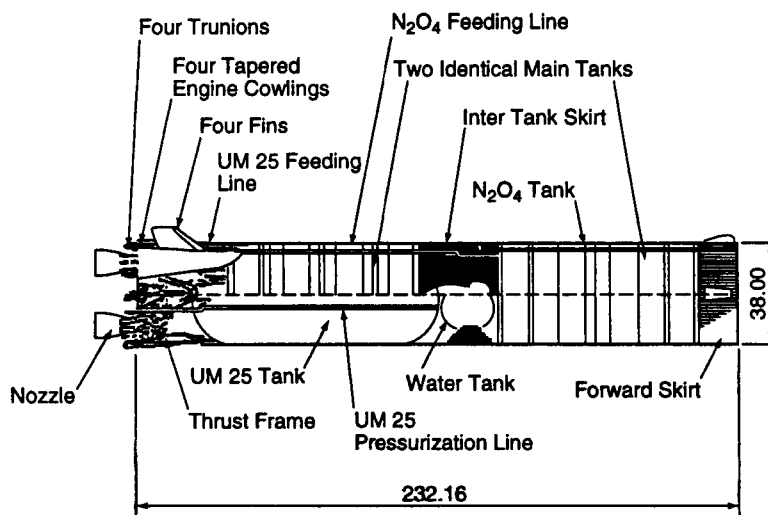


Figure 53. Ariane 4 aerodynamic surfaces.

Figure 52 shows the conical shrouds which are used on the Ariane 1, Ariane 2, Ariane 3, Ariane 40, Ariane 42P, Ariane 44P, Ariane 42L, Ariane 44LP, Ariane 44P, and Ariane 44L to enhance the aerodynamic flight stability of the vehicles. The Ariane 5 uses aft skirts on the solid boosters to the same advantage. [43] These LV's launch payloads of 4,070 to 15,000 lb to GEO. Hermes takes 22,000 lb to GEO.

## **4.2 Russian**

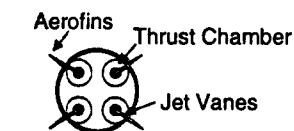
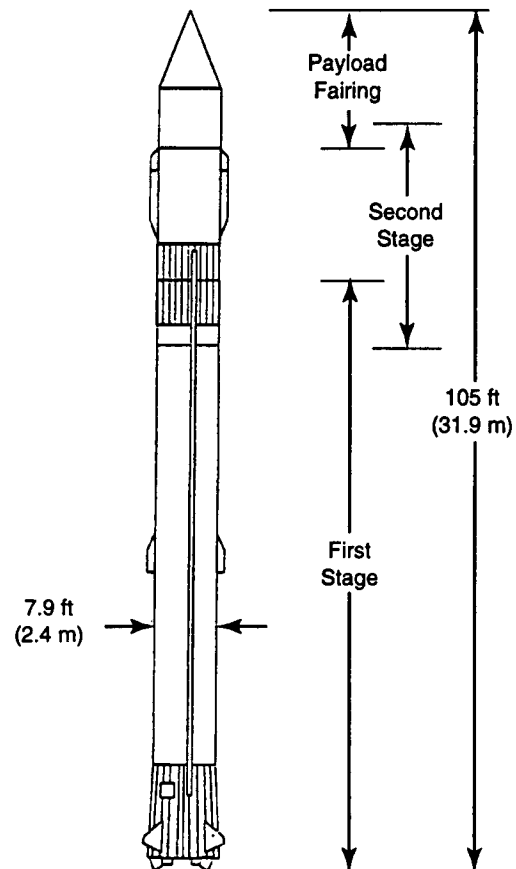
The Russians used aerodynamic surfaces of 4 ft<sup>2</sup> each on the SL-1, SL-2, and the SL-5 of the 1960's. In current production, the two-stage Kosmos uses aerodynamic surfaces for flight stability and jet vanes for control. Kosmos launches 3,000 lb to GEO (Fig. 54). [42] The liquid propellant first stage has a diameter of 7.9 ft with a base section that flares to a diameter of 9.35 ft. The four stabilization fins have a fin-tip to fin-tip diameter of 14.4 ft. The Kosmos has 400 successful launches to its credit.

The Tsyklon, which means cyclone, also uses aerodynamic fins for flight stability. Tsyklon launches 8,800 lb to LEO and accounts for over 10% of all Russian launches with a success rate of 99%. Both the first and second stages have four vernier RCS for three-axis control. The eight vernier engines are positioned within aerodynamic shrouds which enhance the flight stability (Fig. 55). [43]

The Vostok, Soyuz, and Molniya all use aerodynamic fins for flight stability. Sometimes they do not show up well in diagrams, but Figure 55 (an actual photo) verifies their existence. In addition, the Vostok, Soyuz, and Molniya all use a similar set of four wide-angle, conical strap-on boosters. [43] The pronounced shape of the conical set of strap-ons has significant aerodynamic stability advantages. The Vostok is a 2<sup>1</sup>/<sub>2</sub>-stage LV that launches 10,400 lb to LEO. The Soyuz has a more powerful core second stage and launches 15,400 lb to LEO. The Molniya has an additional third-stage core and its primary missions are the Molniya, highly elliptical, orbits. The attitude control system for

### **Overall**

|                   |                          |
|-------------------|--------------------------|
| Length            | 105 ft (31.9m)           |
| Gross Mass        | 264,000 lb (120,000 kg)  |
| Thrust at Liftoff | 330,000 lb (1,470,000 N) |

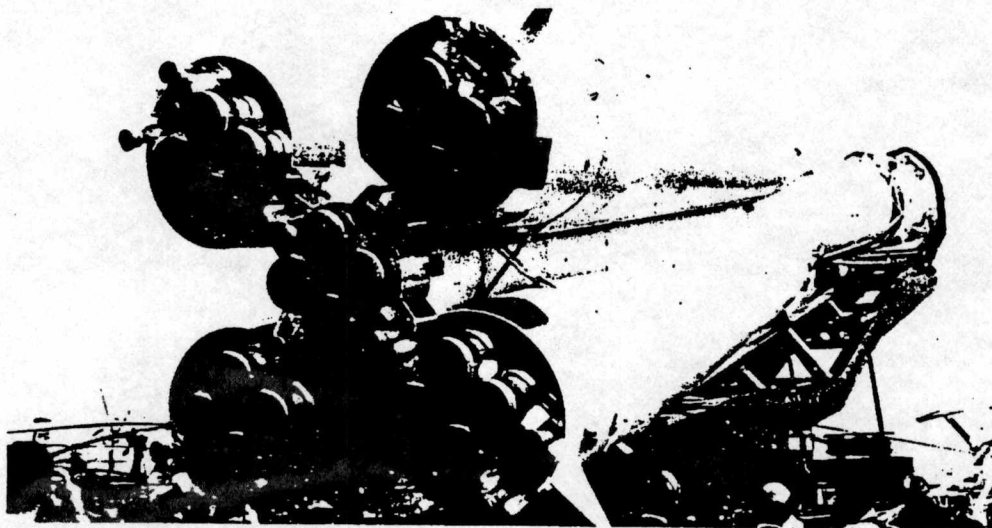
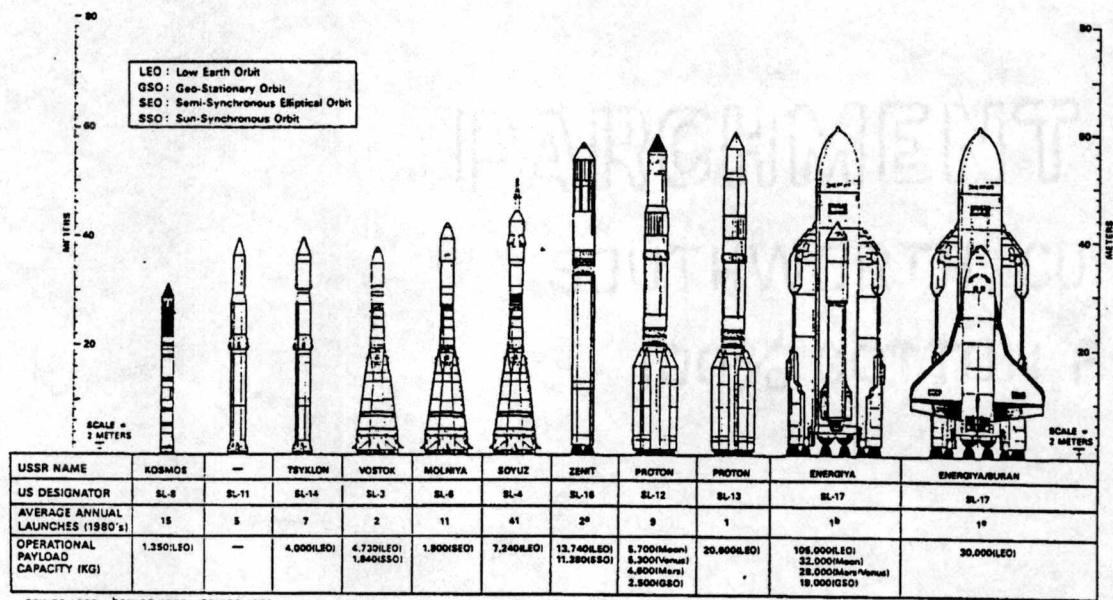


Base view of two  
RD-216 engines on  
first stage

### **Attitude Control System**

The first stage is controlled by aerodynamic fins and jet vanes under the engine exhaust.

Figure 54. Kosmos launch vehicle.



Vostok: not shown publicly until 6yr after placing Gagarin in orbit, its cluster configuration is still in use (Reginald Turnill)

Figure 55. Russian LV's and close-up of Vostok fins.

all three consists of two gimbaling vernier engines on each strap-on, and four vernier engines on the core. The core second stage is controlled by four vernier engines.

Buran launches 66,000 lb to GEO and uses aerodynamic flight control of elevons, rudders, and a body flap.

### 4.3 Chinese

Most all of China's Long March family of LV's incorporate aerodynamic surfaces. A variety of fin planform and sizes are used (Fig. 56). [43] The Long March 1 (LM-1), also designated CZ-1 for Chang Zheng, launched China's first satellite in 1970 using aerodynamic surfaces for flight stability and was controlled in the first and second stages by jet vanes. The fin surfaces were of a trapezoidal planform, about 25 ft<sup>2</sup> each, and had both LE and TE sweep. The LM-1 (CZ-1) was a three-stage LV which launched 660 lb to LEO. An enhanced version, the LM-1D (CZ-1D), also uses similar trapezoidal fins for stability and uses jet vanes in the first and second stages for vehicle control. RCS is used in the ballistic coast phase. It launches 1,650 lb to LEO or 440 lb to GTO (Fig. 57). [43]

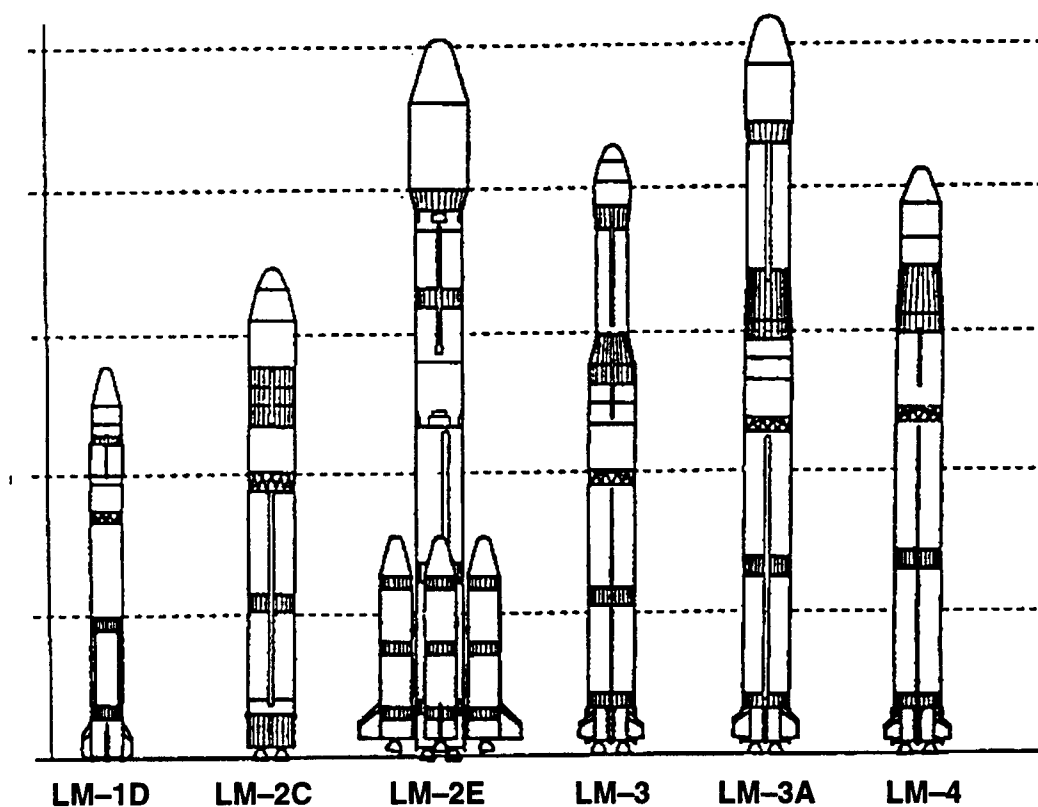


Figure 56. Chinese Long March LV's.

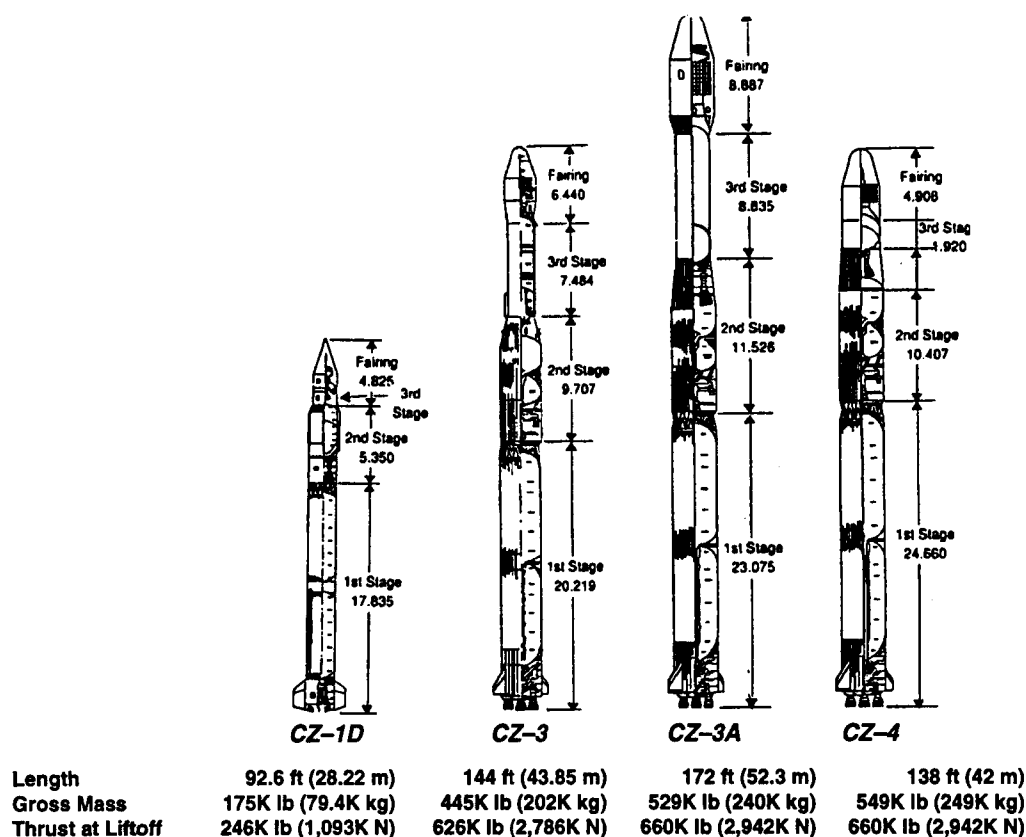


Figure 57. Comparison of aerodynamic surfaces on Long March vehicles.

The CZ-3 was first launched in 1984 and uses another planform of aerodynamic surface with an unswept TE. It is a three-stage LH<sub>2</sub>/LOX propelled LV and is controlled by hydraulic engine gimballing of  $\pm 10^\circ$ . The CZ-3 launches 11,000 lb to LEO or 3,300 lb to GTO (Fig. 57).

The LM-4 (CZ-4) is also a three-stage LV and was first launched in 1988. It uses trapezoidal fins about 22 ft<sup>2</sup> each with a LE sweep and no TE sweep. The CZ-4 launches 8,800 lb to LEO or 2,430 lb to GTO. In 1990, China launched its first commercial satellite aboard a CZ-3 using aerodynamic surfaces. Control is by hydraulic engine gimballing  $\pm 10^\circ$  (Figs. 57 and 58). [43]

The CZ-3A began in 1992 as a commercial three-stage LV for the international market. It also uses the trapezoidal fins with swept LE's and unswept TE's. The planform area of each fin is about 22 ft<sup>2</sup>. The CZ-3A launched 15,800 lb to LEO and 5,500 lb to GTO (Fig. 57). Control is provided by hydraulic engine gimbaling  $\pm 10^\circ$ .

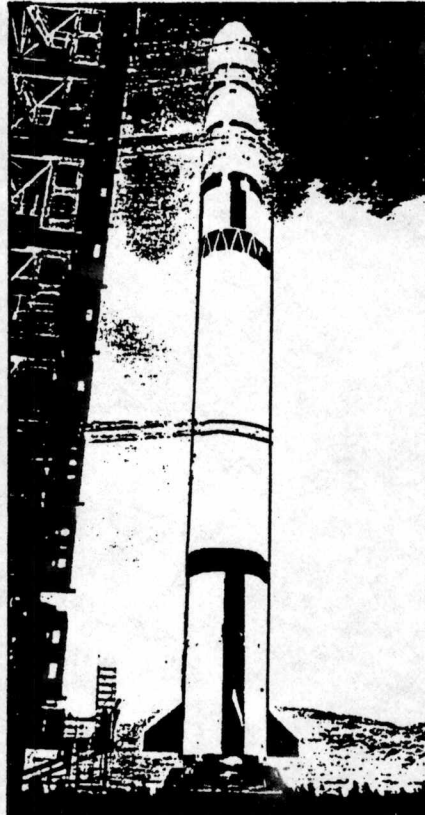


Figure 58. China's CZ-4A aerodynamic surfaces.

The CZ-2E also uses aerodynamic surfaces for stability and engine gimbaling of  $\pm 10^\circ$  for control. The four aerodynamic fins are about 25 ft<sup>2</sup> each. CZ-2E began service in 1990 and uses liquid propellant strap-on boosters which enable it to launch 20,403 lb to LEO and 7,430 lb to GTO (Fig. 56).

#### 4.4 Japanese

The L-4S began flights in 1966 and is a four-stage LV which uses two sets of aerodynamic surfaces for flight stability and flight control. The TR-1 began flights in 1985 and uses aerodynamic surfaces for flight stability. The M-3C is a three-stage LV using aerodynamic surfaces and launches 430 lb to LEO. The M-3H and M-3S began flights in 1980 and are both three-stage LV's using aerodynamic surfaces and launch 640 lb to LEO. The M-4S began flights in 1970, and it is a larger four-stage LV built by Nissan Motor Company. It uses aerodynamic surfaces for flight stability and launches 400 lb to LEO.

The M-3SII began flights in 1985 and was built for the Halley's comet mission. It is a three-stage, solid propellant LV with two strap-on boosters. It uses four tapered aerodynamic surfaces about 17 ft<sup>2</sup> each with swept LE's and straight TE's for flight stability, with four small RCS attached to the tips for roll control. It launches 1,720 lb to GEO (Fig. 59). [42]

The M-5 is a 1995 LV with launch capability of 4,300 lb. The M-5 has been specially designed with aerodynamic surfaces that guarantee aerodynamic flight stability of the vehicle up to Mach 2.0. Thereafter, the vehicle is only slightly unstable until stage separation. Solid RCS are also attached to the tips of the fins for roll control (Fig. 59). Three rate gyros on the first stage provide damping signals for first-stage pitch, yaw, and roll control. The M-5 uses INS and fiber optic gyros on the third stage. The H-2, also a 1995 LV, uses a conical flare shroud. [42]



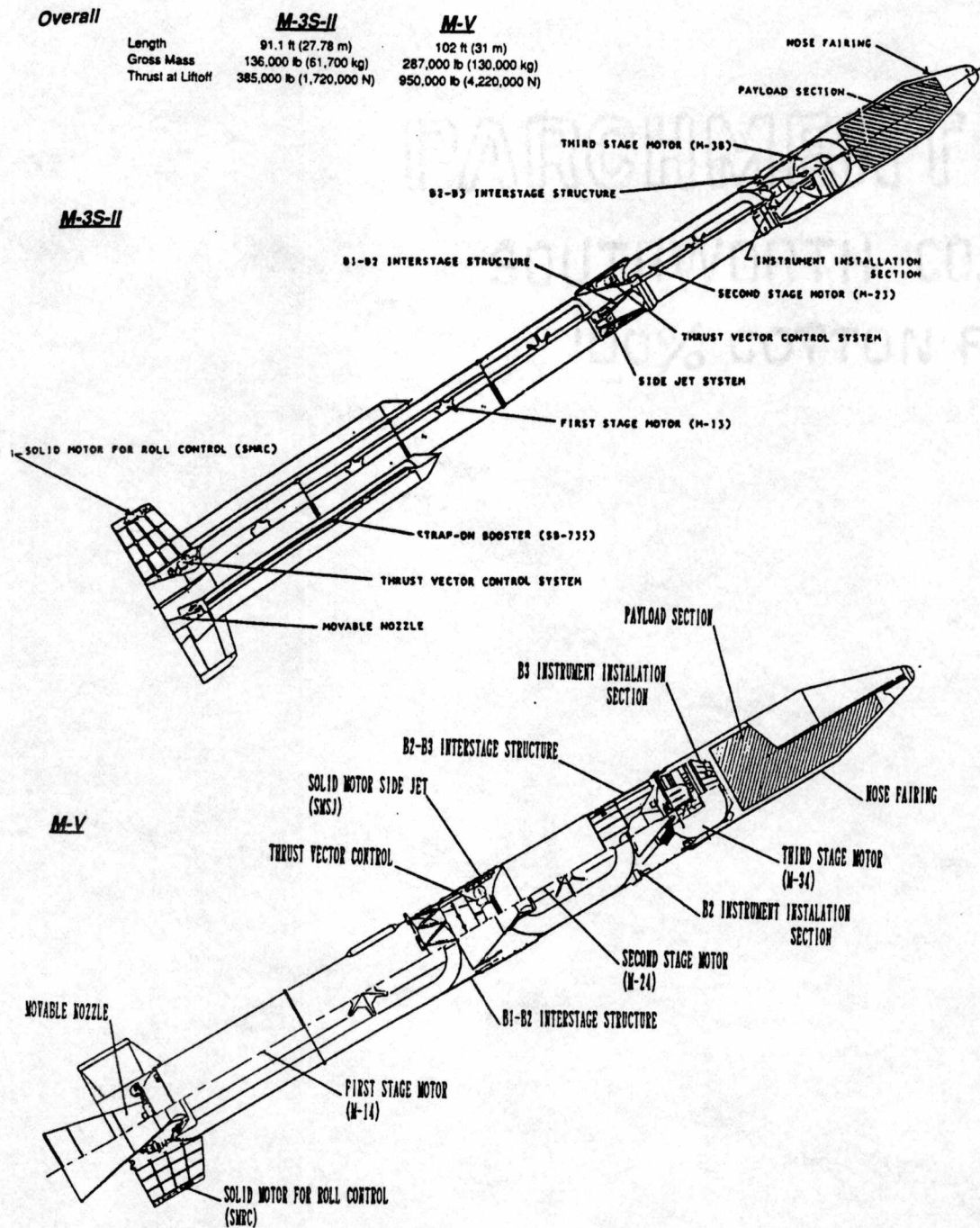


Figure 59. Japanese M-3SII and M-5 LV's.

#### 4.5 Other

Brazil's VLS began flights in 1992, launching satellites to LEO and using aerodynamic surfaces (fins) for flight stability (Fig. 60). [42] India's four-stage SLV-3 was first launched in 1979 and uses four aerodynamic surfaces (fins) for flight stability and aerodynamic flight controls at the tips of the fins. The uprated version is the ASLV, which uses two strap-on boosters and launches 330 lb to LEO (Fig. 61). [42] Israel's Shavit was first launched in 1988, taking 350 lb to LEO. It is a three-stage LV using four aerodynamic surfaces, trapezoidal fins with swept LE's and TE's (Fig. 62). [42] Shavit launched Israel's third Ofeq satellite in April of 1995. This launch took 600 lb to LEO (Space News; April 16, 1995). The Clinton administration is considering a U.S. Defense Department payload to be launched on a Shavit LV. (Aviation Week, December 18, 1995)

This survey of current usage of aerodynamic surfaces by other nations demonstrates the wide variety of LV aerodynamic surface planform shapes in current use along with their ability to provide increased flight stability and significant flight control augmentation.

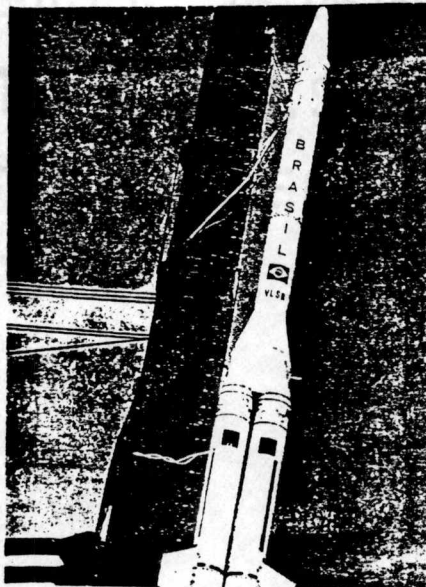


Figure 60. Brazil's VLS aerodynamic surfaces.

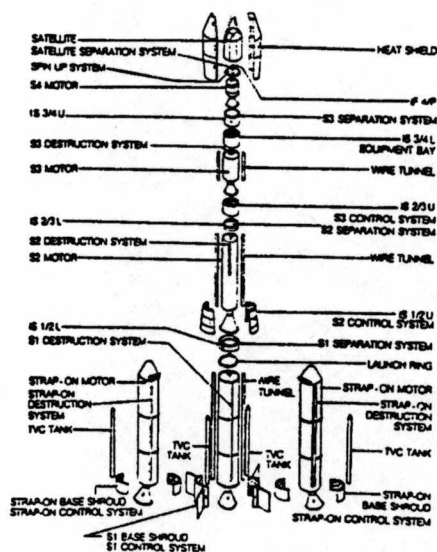
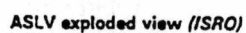


Figure 61. India's SLV-3 and ASLV aerodynamic surfaces.

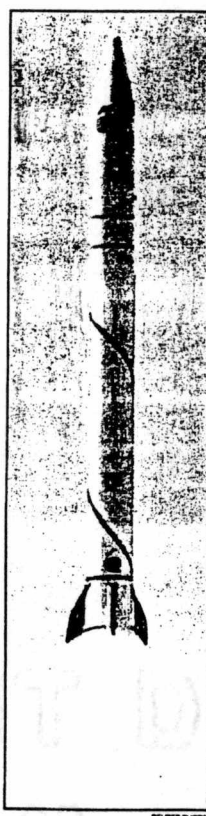


Figure 62. Israel's Shavit LV.

## **CHAPTER V. CONTROL REQUIREMENTS ANALYSES AND DESIGN OF THE CANDIDATE FLIGHT CONTROL AUGMENTORS**

### **5.1 Experimental Launch Vehicle**

For this research effort, a LV has been selected from the NLS family. This is an expendable vehicle with a length of 269.3 ft, a LOX tank forward and a LH<sub>2</sub> tank aft in its original configuration, large LOX feedlines which run the length of the vehicle on the exterior fuselage, a biconic nose, and propelled by six bipropellant rocket engines each providing 583,000 lb of vacuum thrust. This experimental LV is very similar in many respects to the Saturn V (described in Chapters II and III) and is 74% the length of the Saturn V. The Saturn V has been used as the reference for this LV throughout its design program.

This basic vehicle has been redesigned to become the experimental LV of this research. First of all, the position of the LOX tank and the LH<sub>2</sub> tank have been interchanged in order to delete approximately 10,000 lb of structural weight, and thus be able to carry more payload to orbit and reduce the launch cost/lb. Of course, locating the heavy LOX tank aft gives the vehicle an aft cg which in turn requires control augmentation, which is the problem being solved in this research. Having done this, there is no need for the large LOX feedlines running the length of the vehicle on the exterior fuselage, as the LOX is now in close proximity to the engines. The LOX feedlines are removed. An added benefit of this configuration is the much reduced fueling time. What the vehicle looks like at this point is depicted in Figure 63.

Next, the biconic nose has been redesigned to a 1.2 caliber ogive nose in order to better accommodate the forward located flight control augmentors (FCA's) by providing a cleaner and more uniform flow. And finally, four cruciform, all-movable, FCA's have been added at the aft end of the vehicle on the four engine shrouds of the four outer

engines. The cruciform controls allow immediate turns in any plane. This is the experimental LV of this research, which is still very similar to the Saturn V.

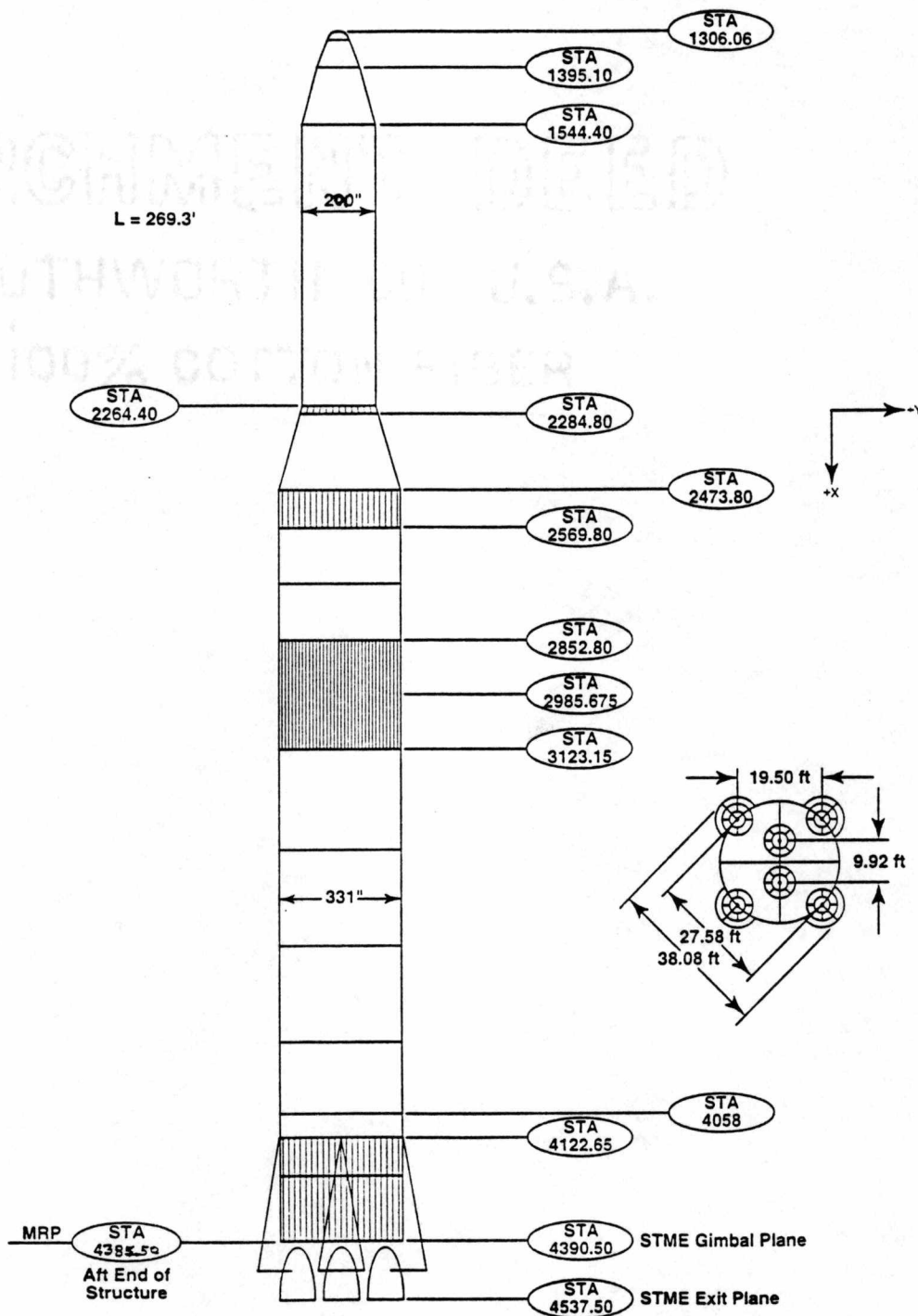


Figure 63. Untreated, unadorned experimental launch vehicle.

Then based on the problem statement in Chapter I, the review of our national heritage of LV's using aerodynamic surfaces in Chapter II, the review of the X-15 and Saturn series aerodynamic surface design concepts in Chapter III, the survey of current use of aerodynamic surfaces by other nations in Chapter IV, the generated mass properties for the experimental LV (NASA MSFC Systems Engineering Division), the generated trajectory data for the experimental LV (NASA MSFC Guidance and Control Systems Division), the basic vehicle aerodynamic data, and the U.S. Air Force Stability and Control DATCOM as a preliminary design tool, a flight control requirement analysis was conducted and a set of candidate FCA's were designed.

## **5.2 Control Requirements and Analyses**

In flight vehicle structural design, 1.5 has been chosen as the traditional factor of safety. It is desired to have the vehicle be able to withstand 1.5 times the limit loads. This structural requirement comes from a materials stress/strain curve and originated with the Federal Aviation Administration for the safety of passenger-carrying aircraft. This same structural factor of safety is used in the LV industry to protect both passengers and payloads (which are sometimes considered as valuable as passengers). In Chapter I it was stated that when the LV is on the launch pad, the environmental disturbances are also multiplied by 1.5 to account for von Kármán vortex shedding effects. That is, 1.5 happens to be used as a safety factor for many parameters in LV design.

NASA MSFC is the NASA Center where LV's have been originally designed and built in the U.S. Consequently, MSFC is the Federal Agency which specifies requirements for NASA LV's. In addition, the LV design business at MSFC has developed some LV "control jargon" that the author has been immersed in for the past ten years and begs the readers indulgence, e.g., "control torque" is the traditional expression for a control moment. In the NASA MSFC Guidance and Control Systems Division,

(where the author has done this research) the traditional LV control requirement is to have the ratio of the control torque to the aerodynamic torque be greater than or equal to 1.5. This simply means that the LV is to have at least 1.5 times as much control authority over the vehicle as the pitch up destabilizing aerodynamic moment. This requirement, established at the Federal level, has been in use for many decades and is used throughout the LV industry to ensure adequate vehicle control. It can be thought of in the same vein as a safety factor for controllability.

Therefore, the basic control requirement is for the LV to have at least 1.5 times as much control authority over the vehicle as the anticipated pitch up destabilizing aerodynamic moment. That is,

$$CR \equiv \frac{CT}{AT} \geq 1.5 \quad (1)$$

where      CR = Control Ratio  
              CT = Control Torque  
              AT = Aerodynamic Torque

This control requirement is used to size the designed LV FCA's. From Figure 64 using force and moment equations, it can be seen that about the cg,

$$CT = T \sin \delta_E (E) \ell_E \quad (2)$$

$$AT = N \ell_{cp} = C_{N_\alpha} \alpha Q S \ell_{cp} = C_N Q S \ell_{cp} \quad (3)$$

$$\ell_{cp} = X_{cp} - X_{cg} \quad (4)$$

where

$T$  = thrust per engine

$\delta_E$  = engine gimbal deflection angle in degrees

$E$  = number of engines

$\ell_E$  = moment arm from cg to engine gimbal plane

$N$  = normal force

$C_N$  = coefficient of normal force

$C_{N_\alpha} = \frac{\partial C_N}{\partial \alpha}$  = gradient of coefficient of normal force/degree

$\alpha$  = angle-of-attack in degrees

$Q$  = dynamic pressure

$S$  = reference area, which for a LV is the cross sectional area of the major length of fuselage

$\ell_{cp}$  = moment arm from cg to cp

$X_{cg}$  = distance from aft end to cg

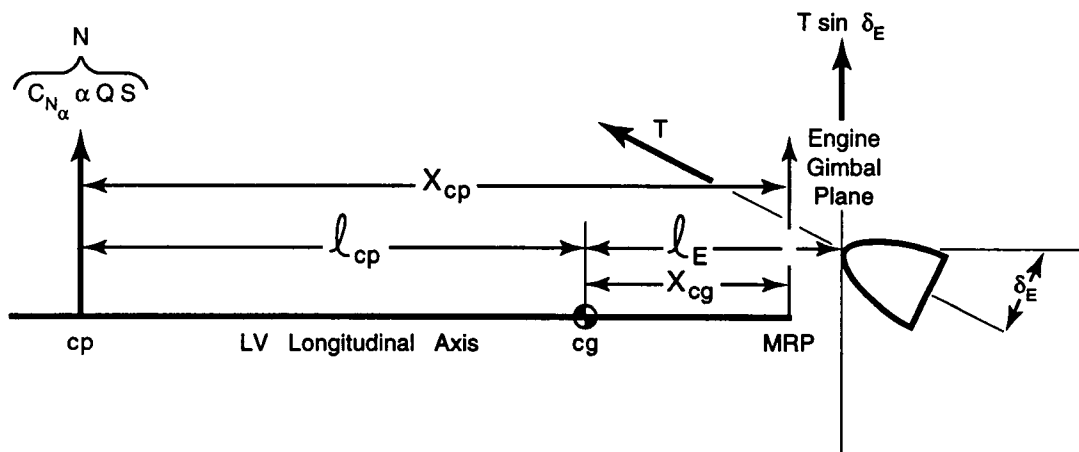


Figure 64. Aerodynamic torque vs. engine gimbal torque.



Thus, equation (1) becomes,

$$CR = \frac{T \sin \delta_E (E) \ell_E}{C_{N_\alpha} \alpha Q S \ell_{cp}} \geq 1.5 \quad (5)$$

From Figure 64, it can be seen that about the moment reference point (MRP),

$$AT = C_{N_\alpha} \alpha Q S X_{cp} = C_m Q S d = C_{m_\alpha} \alpha Q S d \Rightarrow$$

$$\frac{C_{m_\alpha}}{C_{N_\alpha}} = \frac{X_{cp}}{d} \quad \text{for a symmetric vehicle} \quad (6)$$

where  $X_{cp}$  = distance from aft end to cp

$C_m$  = coefficient of pitching moment

$C_{m_\alpha} = \frac{\partial C_m}{\partial \alpha}$  = gradient of coefficient of pitching moment/degree

$d$  = reference length, which for a LV is the diameter of the major portion of the fuselage and is called a caliber.

Therefore,

$$AT = C_{N_\alpha} \alpha Q S \frac{C_{m_\alpha}}{C_{N_\alpha}} d = (C_{m_\alpha} Q) \alpha \frac{\pi d^3}{4} \quad (7)$$

This shows that the maximum AT, for fixed  $\alpha$ , occurs when  $(C_{m_\alpha} Q)$  is a maximum.

When a LV flies through its ascent trajectory, the traditional critical point from an aerodynamic loads standpoint is near maximum dynamic pressure (max Q). LV control systems and structures are designed around this critical point and LV trajectories are “shaped” accordingly. For fixed Mach numbers (M), and azimuthal winds and gusts, dynamic loading plots called “squatcheloids” routinely plot  $Q\alpha$  vs.  $Q\beta$  during preliminary design of a LV where  $Q\alpha$  and  $Q\beta$  are the normal and lateral load parameters,

and  $\beta$  is the angle-of-sideslip. Many times in preliminary design, for a fixed aerodynamic angle, max  $Q$  is used as the critical point, since it is close to the maximum of  $(C_{m_\alpha} Q)$  as a function of time in the ascent trajectory. Therefore, the mass properties and trajectory data always denote max  $Q$  as can be seen in Appendix A. For the LOX tank aft experimental LV, Figures 65 to 68 show the variation of  $(C_{m_\alpha} Q)$  and  $Q$  with  $M$ , and the variation of  $(C_{m_\alpha} Q)$  and  $C_{m_\alpha}$  with  $Q$ . It can be seen that the maximum of  $(C_{m_\alpha} Q)$  is very close to that of max  $Q$  as a function of time in the trajectory. Therefore, for preliminary FCA sizing the traditional and more conservative trajectory time of max  $Q$  is used. Figures 69 to 71 show variation of  $X_{cp}$ ,  $X_{cg}$ , and  $T$  with  $Q$  during ascent.

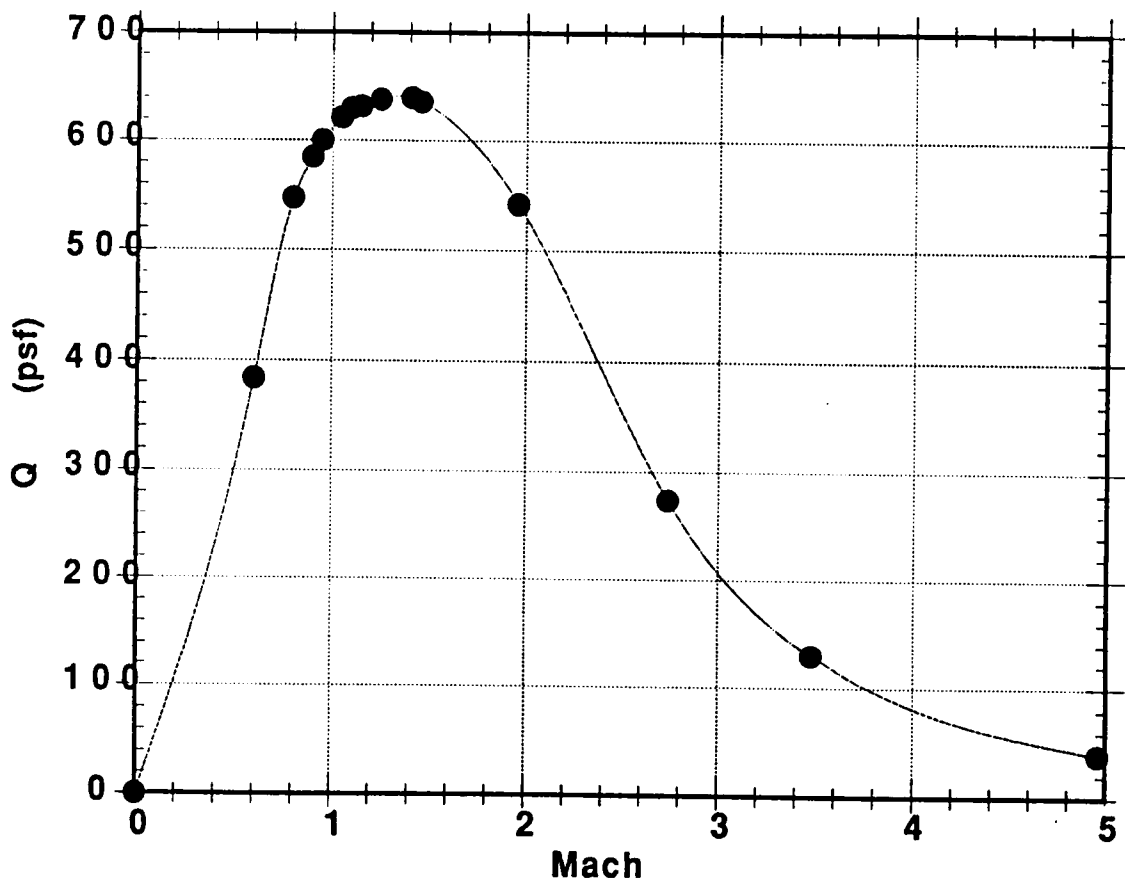


Figure 65. Variation of dynamic pressure during ascent.

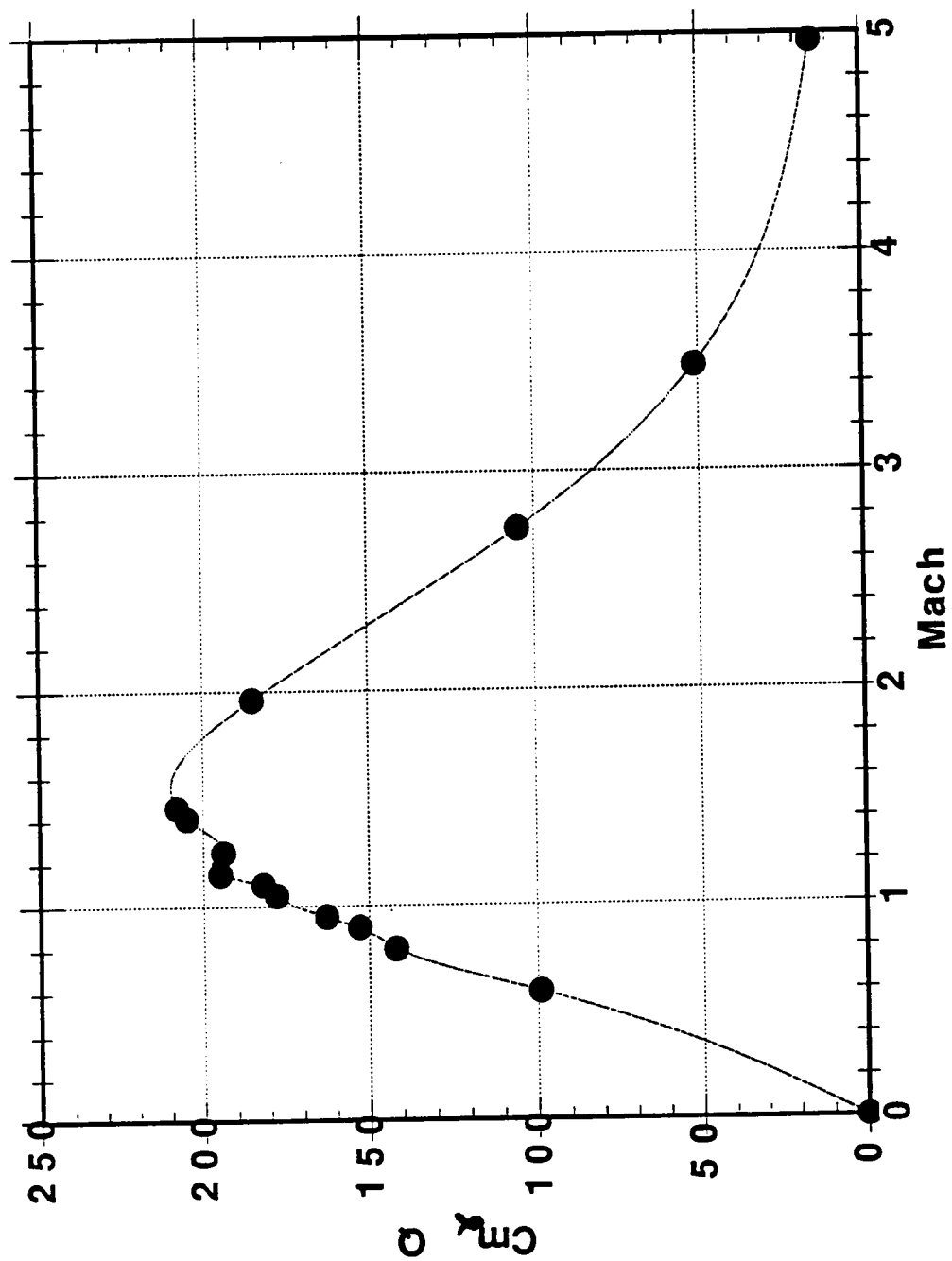


Figure 66. Variation of pitching moment gradient times dynamic pressure during ascent.

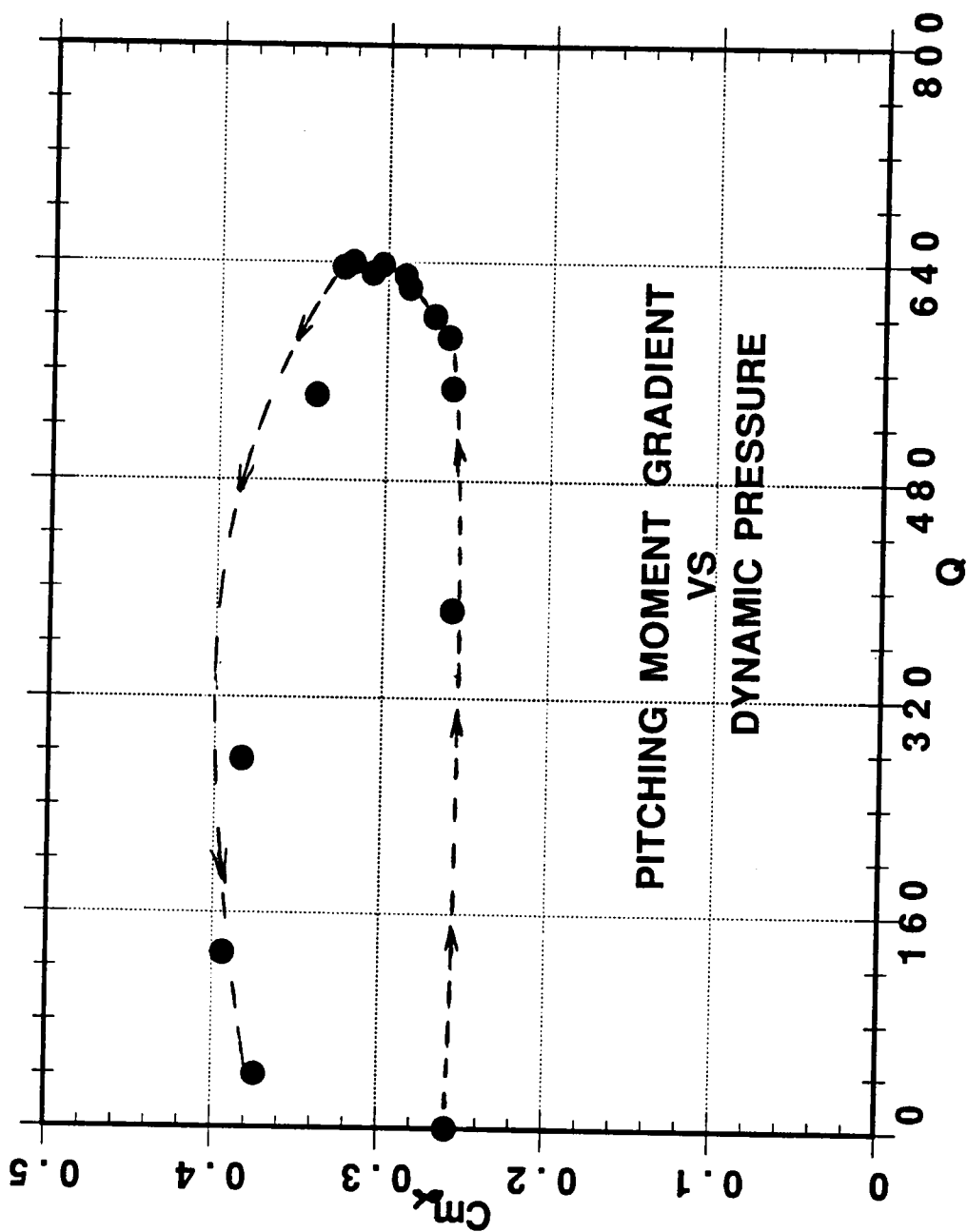


Figure 67. Variation of pitching moment gradient with dynamic pressure.

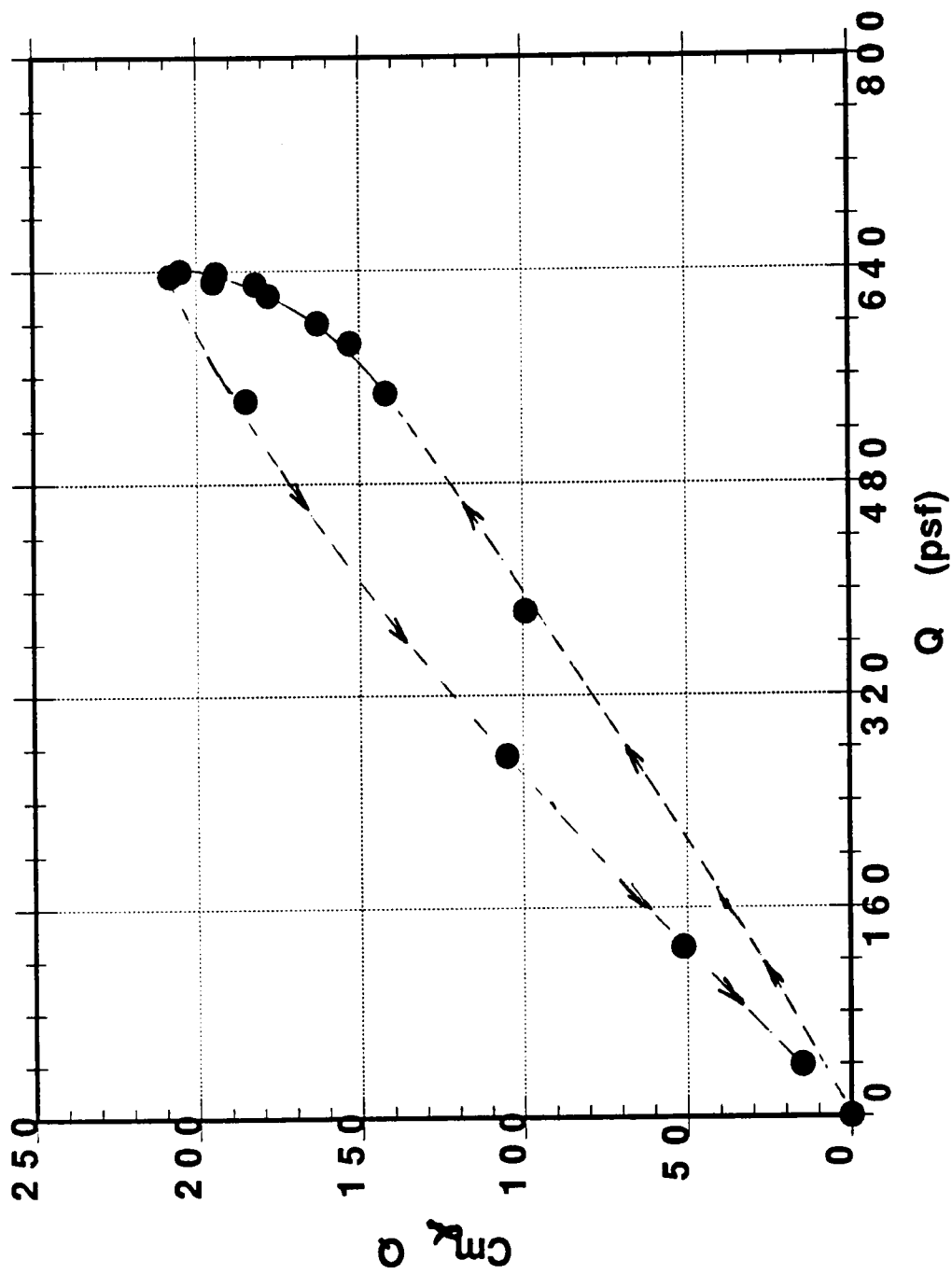


Figure 68. Variation of pitching moment gradient times dynamic pressure with dynamic pressure.

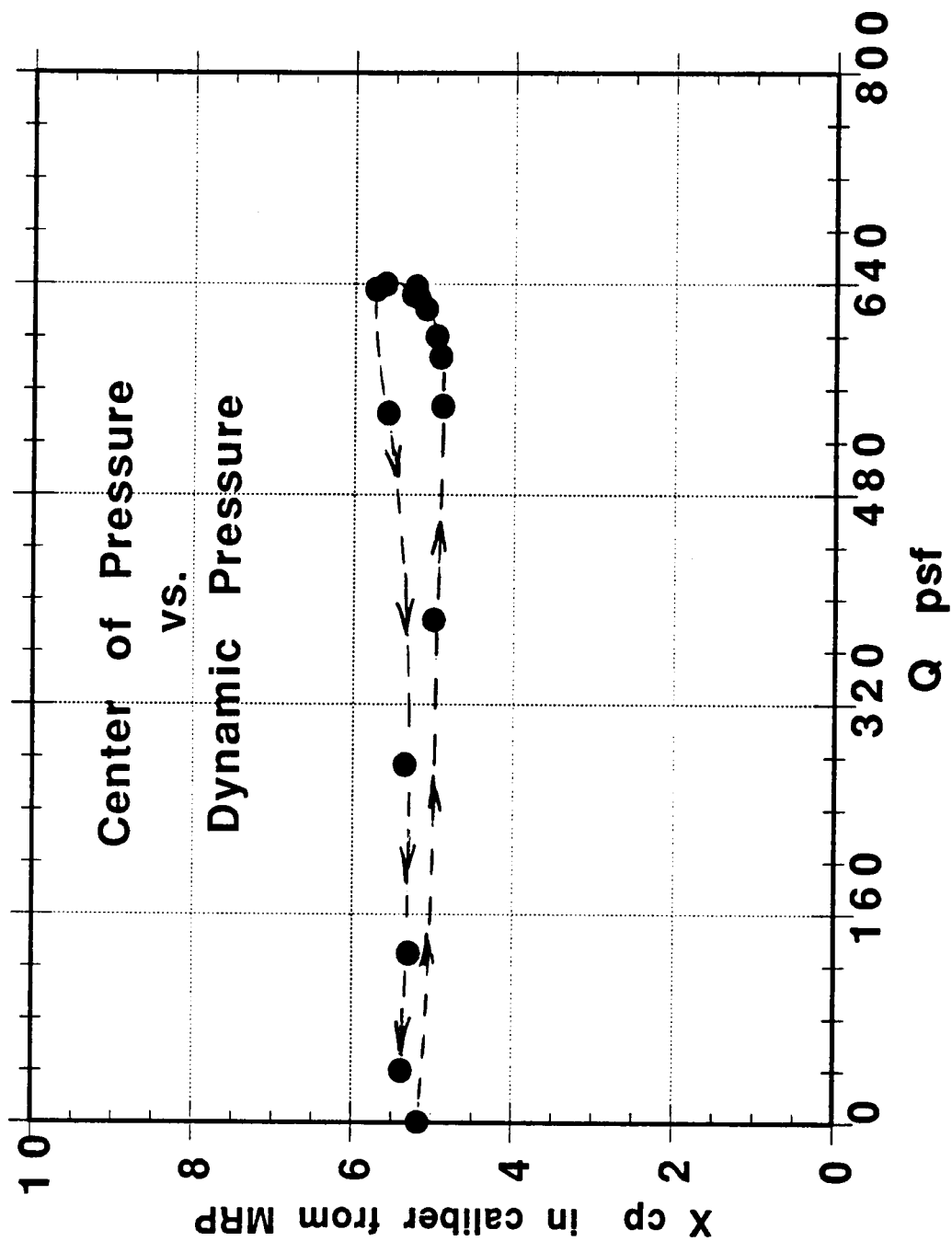


Figure 69. Center of pressure location vs. dynamic pressure.

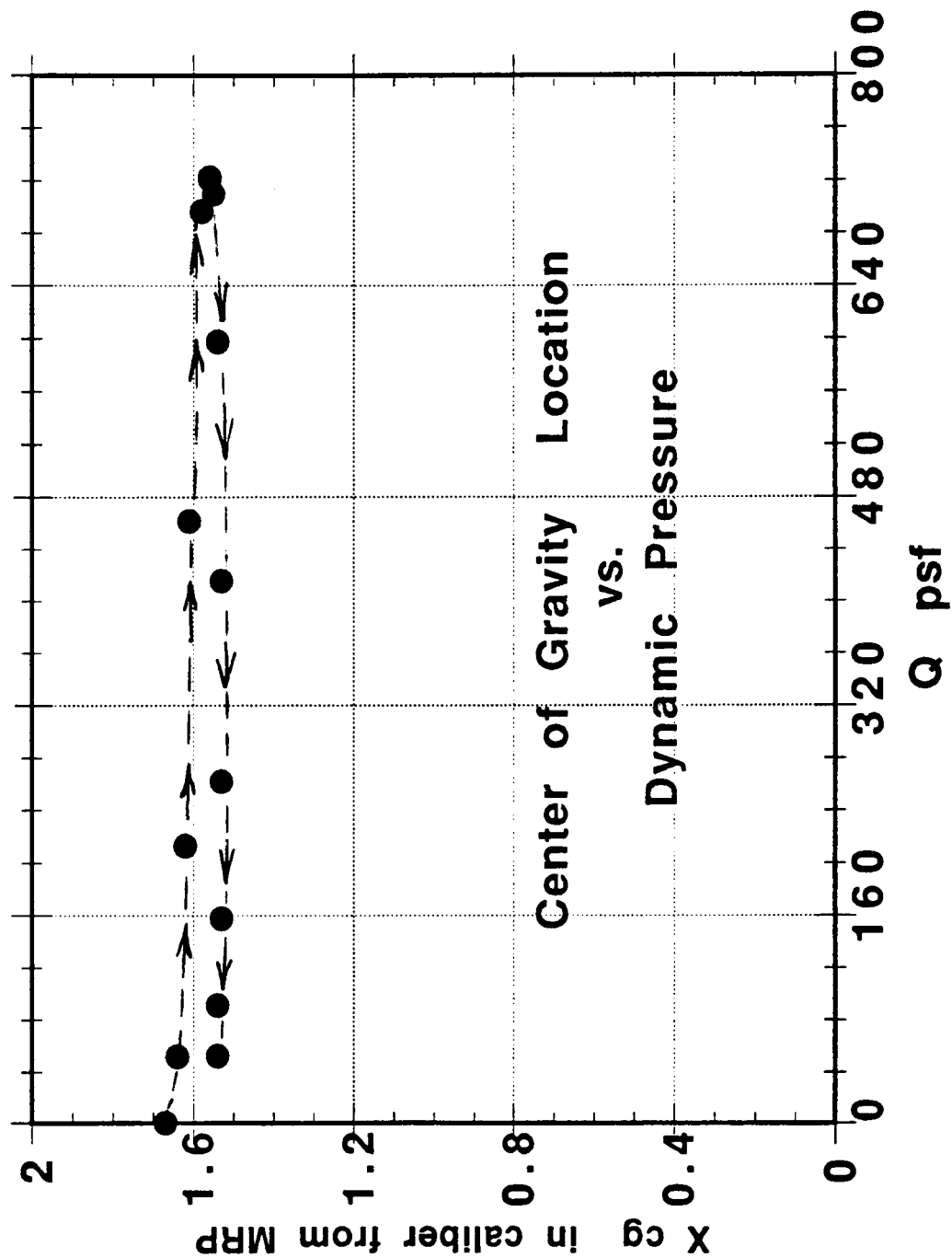


Figure 70. Center of gravity location vs. dynamic pressure.

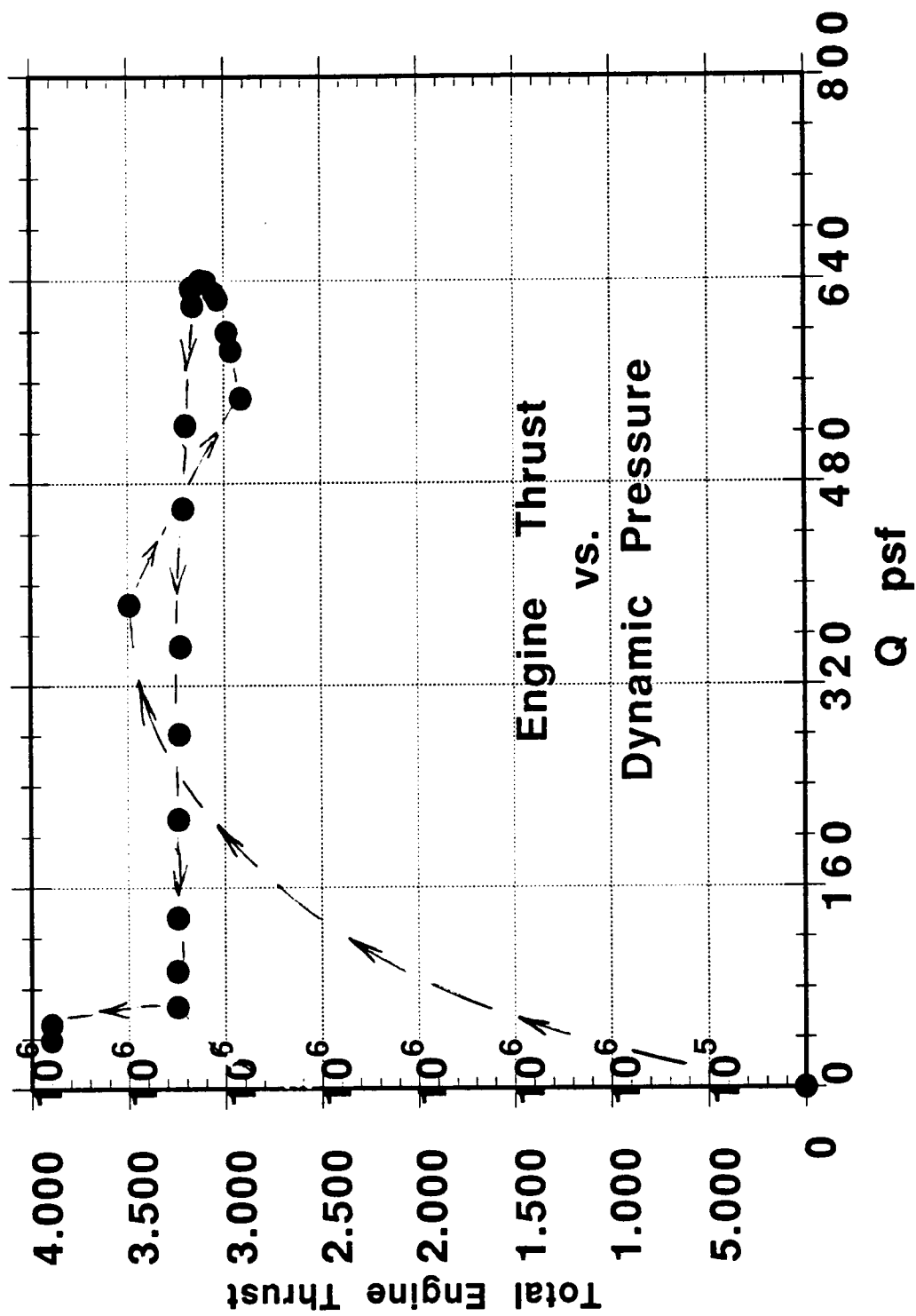


Figure 71. Engine thrust vs. dynamic pressure.



To evaluate equations (5) and (6) the experimental LV preliminary mass properties and trajectory data, along with the 1.5 stage NLS aerodynamic data, are utilized. The relevant mass properties data give the cg location in station numbers, and the moments of inertia, and the products of inertia as a function of time (Appendix A). The relevant trajectory data give thrust, altitude, and dynamic pressure as a function of Mach number and Mach number as a function of time (Appendix A). The 1.5 stage NLS aerodynamic data as a function of Mach number are in their standard linearized gradient/degree format with respect to the MRP at the aft end of the LV (Appendix A). This is the standard format that MSFC uses when aerodynamic data are shipped out to other Centers or industry contractors so that any user-supplied aerodynamic angles or moment centers can be used. From these and Figure 63, max Q = 639.9 psf; at that point T = 520,513 lb/engine; E = 6; M = 1.41; S = 597.56 ft<sup>2</sup>; d = 27.58 ft = 1 caliber; C<sub>N<sub>α</sub></sub> = 0.057153; C<sub>m<sub>α</sub></sub> = 0.321070; X<sub>cg</sub> = 52.98 ft; and ℓ<sub>E</sub> = 53.4 ft. Using δ<sub>E</sub> = 4° (although 8° are possible, 4° is the usual number allotted to routine control), and α = 4° (based on the maximum used during Saturn V flights), equation (6) gives X<sub>cp</sub> = 5.618 calibers = 154.9 ft and ℓ<sub>cp</sub> = 102 ft. Thus, equation (5) becomes,

$$CR = \frac{CT}{AT} = \frac{11,633,452 \text{ ft-lb}}{8,916,467 \text{ ft-lb}} = 1.30$$

This is inadequate controllability, since the minimum requirement is 1.5.

Using equation (1) to size the all-movable FCA planform which will provide the required controllability,

$$CR = \frac{CT + N_F \ell_F}{AT} = \frac{T \sin \delta_E (E) \ell_E + C_{N_{\alpha F}} (\alpha + \delta_F) Q (2) S_F \ell_F}{C_{N_{\alpha}} \alpha Q S \ell_{cp}} \geq 1.5 \quad (8)$$

where  $N_F$  = normal force provided by FCA

$$C_{N_{\alpha F}} = \frac{\partial C_{N_F}}{\partial \alpha} = \text{FCA gradient of coefficient of normal force/degree}$$

$S_F$  = planform area of one FCA in a horizontal position

$\ell_F$  = moment arm from cg to ac of FCA

$\delta_F$  = deflection angle in degrees of the all-movable FCA

as shown in Figure 72. It is assumed here that the ratio of local dynamic pressures at the FCA location is very close to 1, that the ac of the FCA is near the quarter-chord location (which is a conservative assumption), and that the moment about the FCA ( $C_{m_{acf}}$ ) is near zero, i.e., that the FCA is symmetrical and without camber.

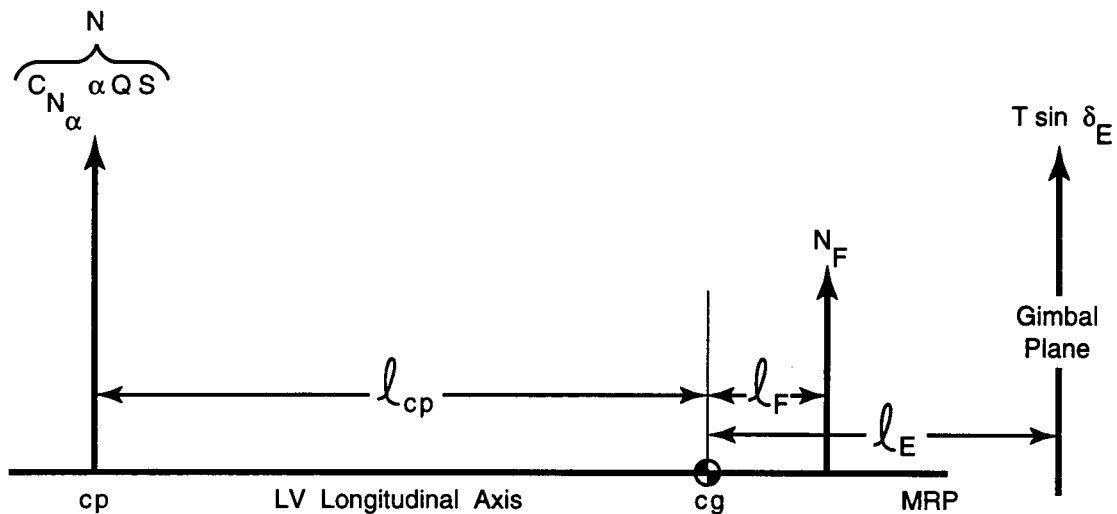


Figure 72. Control augmentation provided by FCA's.

Equation (8) shows that the FCA's augment the control torque by shifting the LV cp aft, thereby reducing the moment arm of the resultant normal force on the vehicle. The cp is defined as the point of no moment, whereas the ac is defined as the point where the moment is independent of  $\alpha$ , i.e., where  $C_{m_{\alpha}}$  equals zero. For a symmetrical vehicle

$cp = ac$ . Of course, the normal force on the LV is actually a continuous pressure distribution over the entire fuselage.  $X_{cp}$  is the location of the  $cp$  for the entire LV. The new  $cp$  is the point of application of a single resultant normal force which produces the same moment about the MRP as does the continuous pressure distribution over the entire fuselage as shown schematically in Figure 73. The  $cp$  location is of vital concern in flight stability and control, since it determines the aerodynamic moment arm, hence the magnitude of the anticipated aerodynamic torque that the engines must overpower by gimbaling, i.e., TVC.

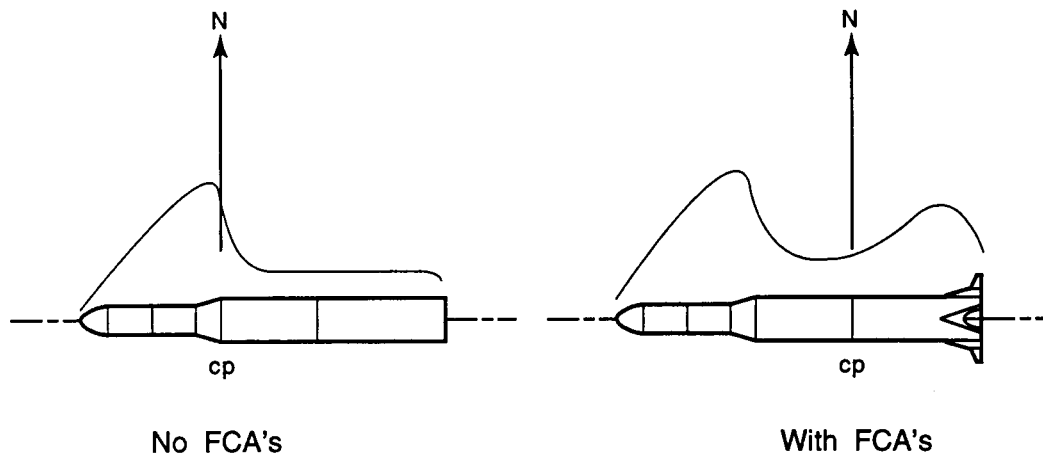


Figure 73. Center of pressure shift provided by FCA's.

To solve equation (8), values for  $C_{N_{\alpha_F}}$  and  $\ell_F$  are needed. Two available options include: utilization of Saturn V data, or use of the U.S. Air Force Stability and Control DATCOM preliminary design tool. From previous work with Saturn V aerodynamic data the value of  $C_{N_{\alpha_F}}$  as a function of  $M$  has been isolated for a 100 ft<sup>2</sup> aerodynamic surface.  $C_{N_{\alpha_F}} = 0.1102/\text{deg}$ . From Figures 21, 47, and 51 shown in Chapters II and III, the geometry of the Saturn V 75 ft<sup>2</sup> surface and its location with respect to the engine shrouds are shown. Figure 74 shows this scaled up to 100 ft<sup>2</sup> for the calculation of  $\ell_F$ . The

calculation of  $\ell_F$  requires the value of the mean aerodynamic chord ( $\bar{c}$ ) of the surface.

The following can be used,

$$\bar{c} = \frac{1}{S} \int_{-b/2}^{b/2} c^2 dy \quad (9)$$

or for a linear taper ratio,

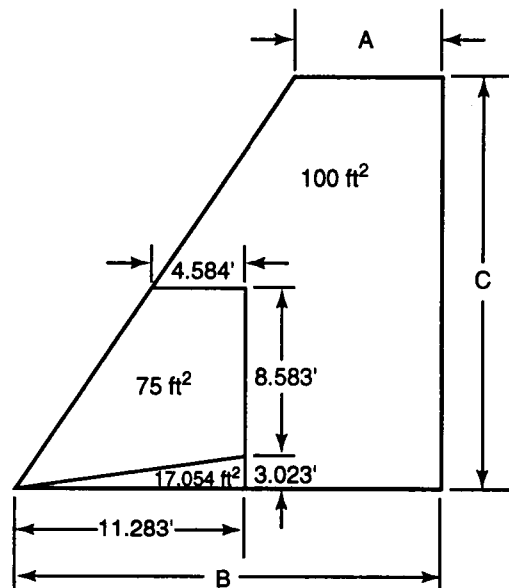
$$\bar{c} = \frac{2 c_0}{3} \frac{1 + \tau + \tau^2}{1 + \tau} \quad (10)$$

where

$b$  = span of the aerodynamic surface

$c_0$  = root chord

$\tau$  = taper ratio =  $c_{TIP} / c_0$ .



$$A = 5.221'; \quad B = 12.85'; \quad C = 13.217'$$

$$\bar{c} = 10.31'$$

$\Rightarrow$

$$\boxed{\ell_F = 45'}$$

Figure 74. Calculation of  $\bar{c}$  for scaled up Saturn V surface.

For aircraft, the span ( $b$ ) of aerodynamic surfaces usually far exceeds the width of the fuselage, therefore  $b$  includes the width of the fuselage. For LV's the reverse is true. If aerodynamic surfaces are used, their span is far less than the width of the LV fuselage, therefore  $b$  does not include the width of the fuselage as shown in Figure 75.

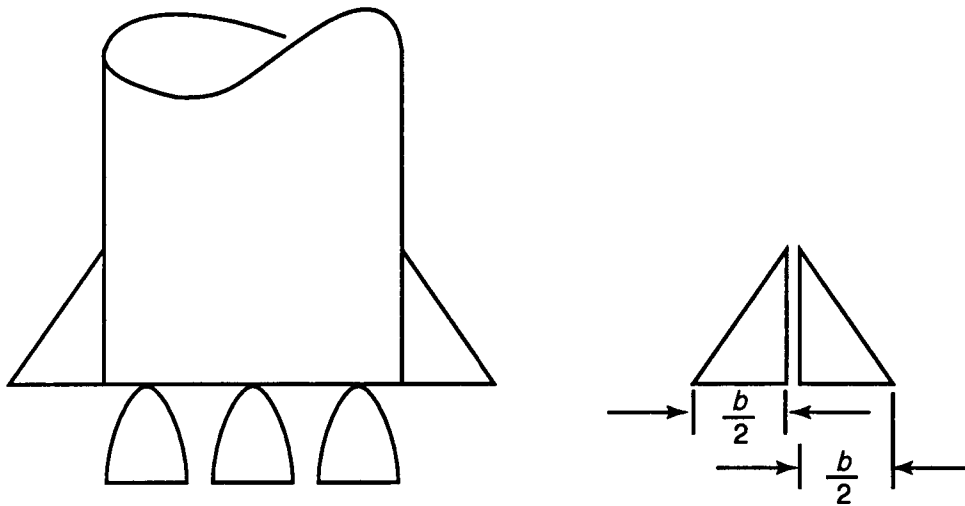


Figure 75. Span of LV aerodynamic surface.

When these values are substituted into equation (8) for the experimental LV,

| $\delta_F^\circ$ | $S_F (\text{ft}^2)$ | Engine Out $S_F$ |
|------------------|---------------------|------------------|
| 2                | 44                  | 95               |
|                  |                     | Choose 80        |
| 4                | 33                  | 71               |

A planform area of  $80 \text{ ft}^2$  is chosen to not only move the  $c_p$  aft and thereby improve stability and control, but also to be able to accommodate a very far aft  $c_g$  and an engine out. Choosing  $80 \text{ ft}^2$  means that in an engine out event the LV control requirements

would be met with less than  $4^\circ$  of FCA deflection. The FCA together with the TVC then provide the required control authority.

### 5.3 Design Considerations

After an aerodynamic surface has been sized, the next step is to decide on a planform shape. Sometimes curved planforms are desirable, e.g., an elliptical planform is desirable for subsonic speeds. Since the experimental LV is expendable, simple planform designs are desired for ease of fabrication. Therefore, for the aft FCA's that have been sized, the only other significant design issues to address are the operational speed regimes, the aspect ratio ( $A$ ), the LE sweep angle ( $\Lambda$ ), the taper ratio ( $\tau$ ), the airfoil, and the airfoil thickness-to-chord ( $t/c$ ) ratio.

A LV flight is normally composed of three distinct phases: ascent, on-orbit, and reentry. Since the experimental LV is an expendable vehicle, there is only ascent to consider. During its ascent flight, the experimental LV flies through three speed regimes: subsonic, transonic, and supersonic. Figure 76 [44] shows the pressure variations of a typical airfoil in these flow regimes, where  $C_p$  is the coefficient of pressure. Figure 77 [44] shows the details of transonic flow over an airfoil. Airfoil characteristics vary with speed ranges. It has been established that the FCA's will provide the greatest assistance to the LV during the time when  $(C_{m_\alpha} Q)$  is maximum. Figure 66 has shown that the  $(C_{m_\alpha} Q)$  maximum occurs when  $M = 1.46$ . Since this is a supersonic Mach number, the FCA's are designed for the supersonic speed regime.

All-movable FCA's are chosen since they are known to have an operational advantage in the supersonic regime. In subsonic flow, a TE effect of a partially movable control surface is felt upstream. In supersonic flow, on the other hand, the forward part of a surface does not feel the effect of a deflected aft portion of a control surface (except through the boundary layer). Thus, if large lift is desired, the entire surface should move.

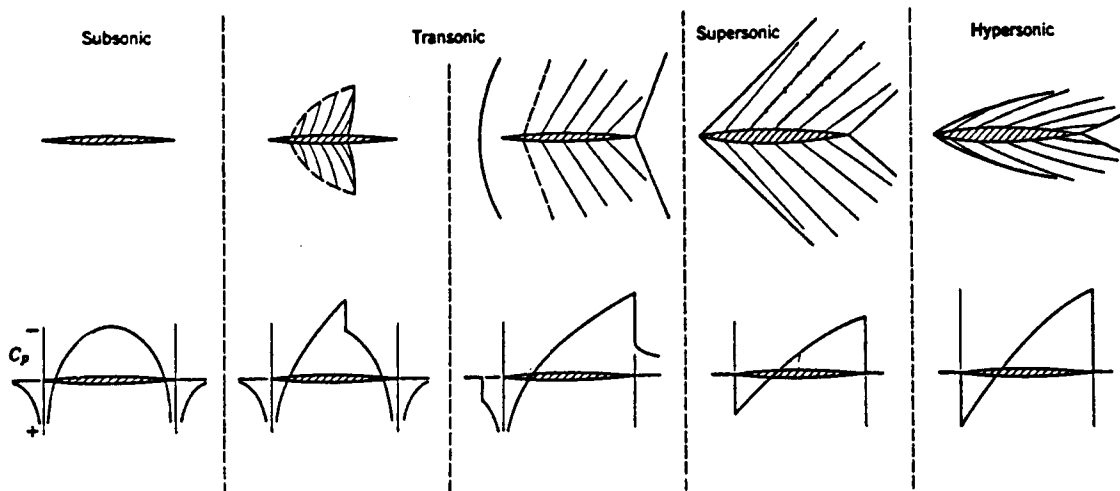


Figure 76. Flow regimes.

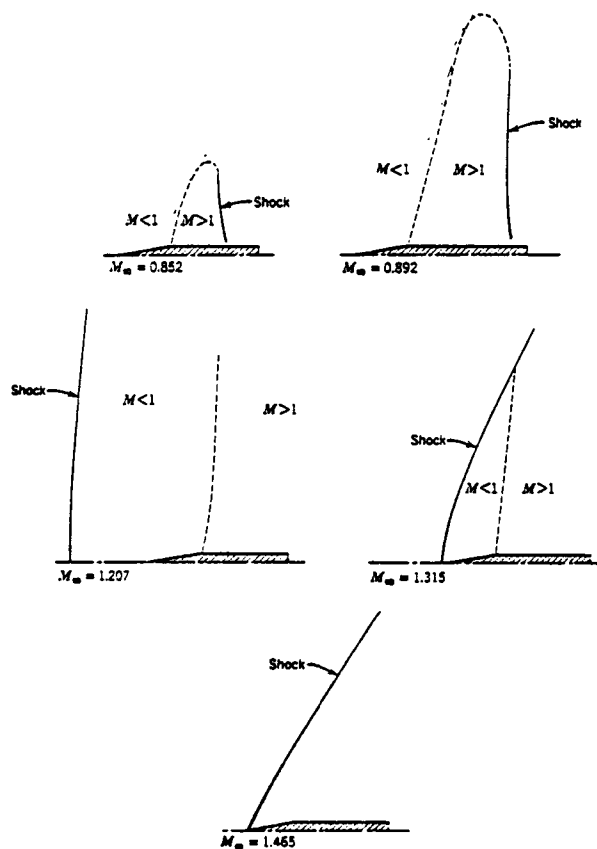


Figure 77. Development of flow over a wedge through the transonic range.

A set of four cruciform FCA's are chosen based on Saturn V success (Chapters II and III), and also on the fact that the experimental LV has four of its six engines located in line with the aft fuselage circumference, hence four engine shrouds which provide suitable location for four FCA's.

The preliminary design considerations here are based on the research of Chapters II, III, and IV, experimental LV calculations, and the U.S. Air Force Stability and Control DATCOM. The U.S. Air Force Stability and Control DATCOM, referred to in this chapter as DATCOM (DATCOM is short for data compendium) has been widely used for stability and control work of flight vehicles in all speed regimes since its inception in 1960 and has been updated many times. The DATCOM is intended to be used as a preliminary design tool before the acquisition of test data. The DATCOM provides stability and control data and methods for the design of flight vehicles in all speed regimes. See Reference 45.

There are not many supersonic airfoils from which to choose. The basic shapes are the diamond and the lenticular. Diamond derivatives include the wedge and the hexagonal. Sharp LE's are used in supersonic flow since they produce weak oblique shock waves which are attached at the LE. A blunt LE would produce a strong bow shock out in front which would result in higher wave drag. In order to reduce the shock wave drag, strong shocks are avoided. According to Wood and Miller of NASA LaRC, Reference 46, historically, airfoil profile selection for supersonic flight has primarily been based upon structural considerations and linearized theory aerodynamic estimates. They have found that comparisons of the linear theory estimates with experimental data have shown an inability of the predictions to match the experimental results. They have also found that the lift results in a detailed study of surface pressures revealed no discernible differences between the diamond and the lenticular airfoil (Fig. 78). [46] From equation (8) above it can be seen that for the experimental LV considered



here, the optimum control augmentation will be provided to the vehicle when  $C_{N_{\alpha F}}$  is a maximum. Thus, since Figure 78 has shown that differences in lift were not discernible between supersonic diamond and lenticular airfoils for a delta wing, the author has chosen a variety of diamond and lenticular airfoils for the FCA designs to be wind tunnel tested.

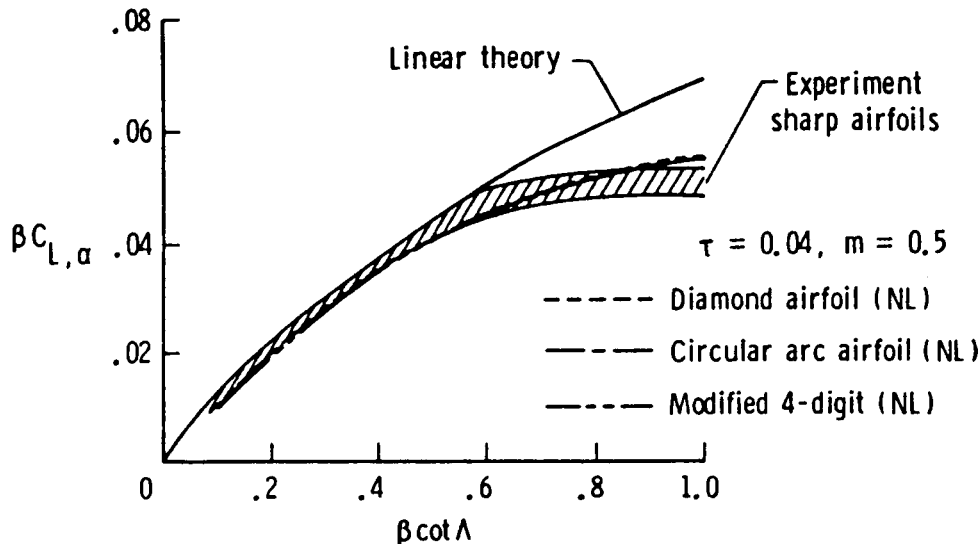


Figure 78. Comparison of lift for diamond and lenticular airfoils.

The primary reason for any variety at all in supersonic airfoils is drag reduction. Drag is a function of  $t/c$ , camber,  $\alpha$ , and  $M$ . Figure 79 shows the effect of camber on drag and pitching moment of two profiles with the same thickness distribution. [47] It can be seen that camber can reduce the critical Mach number by a small amount and increase the transonic and supersonic drag; camber also produces an undesirable pitching moment. Camber gives a minimum drag coefficient at a positive lift coefficient, but this is irrelevant for the LV considered here. While drag minimization is a driver in many airfoil designs, it is not the case for those of a LV. The aerodynamic surface drag of the vehicles studied in Chapters II and III was  $< 2\%$  of the total vehicle drag. The aerodynamic surfaces of the Saturn series and the LV's covered in Chapter II and III were

uncambered. Since drag minimization is not a major consideration for the FCA designs, symmetrical airfoils without camber are chosen.

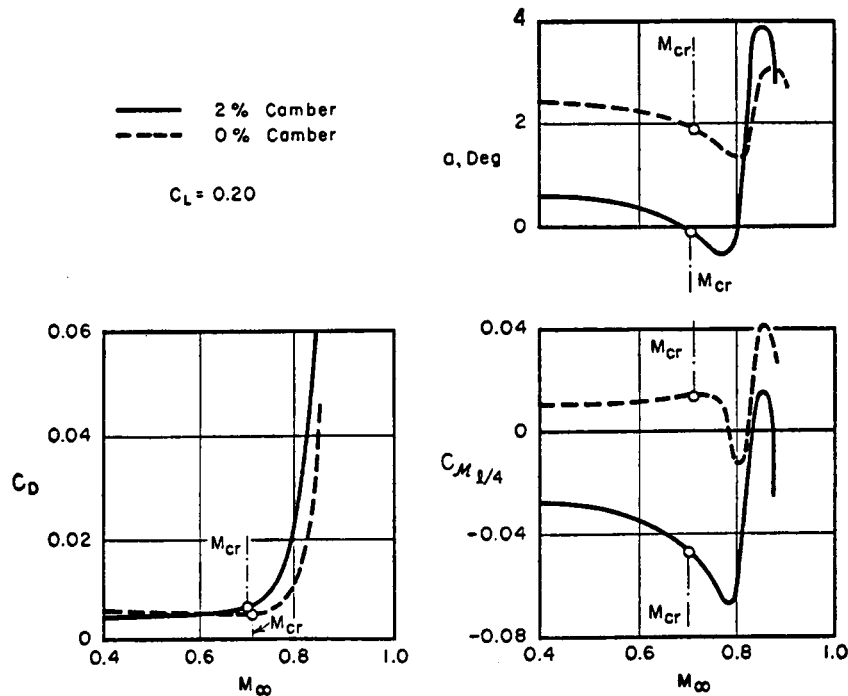


Figure 79. Effect of camber on aerodynamic properties of two profiles with same thickness distribution.

A detailed study (Wood and Miller, NASA LaRC) using a full-potential solution method has shown that the diamond airfoil has lower wave drag than the lenticular. They found the wave drag of the lenticular was from 15% to 40% higher than the diamond airfoil. Of the various diamond and lenticular airfoils they studied, 90% of the wave drag was produced at the airfoil apex and TE. A detailed proof of why the diamond airfoil, having a profile that is formed by straight lines, has less drag than the lenticular, due to the increased surface slopes of the lenticular, can be found in Reference 48. It is also shown there why the lowest drag profile is symmetrical with respect to the chord and has its maximum thickness located just beyond 50% of the chord.

Reference 46 has also shown that zero-lift wave drag does vary with thickness-to-chord ratio  $(t/c)$  squared when  $A\beta < 2$ , but the nonlinear results when  $A\beta > 2$  diverge from linear theory (where  $\beta = \sqrt{M^2 - 1}$ ). When  $A\beta < 2$ , the wave drag for supersonic airfoil sections can be expressed as,

$$C_{D_{\text{wave}}} = K \frac{(t/c)^2}{\sqrt{M^2 - 1}} \quad (11)$$

where  $K$  is a constant factor for a given sharp-nosed airfoil section and has values as given in Figure 80. [49, 45]

|          |                                                                                   |                                                                                   |                                                                                    |                                                                                     |
|----------|-----------------------------------------------------------------------------------|-----------------------------------------------------------------------------------|------------------------------------------------------------------------------------|-------------------------------------------------------------------------------------|
| Section: |  |  |  |  |
| $K$      | 1.0                                                                               | 4.0                                                                               | 5.33                                                                               | 6.0                                                                                 |

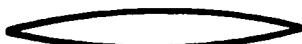
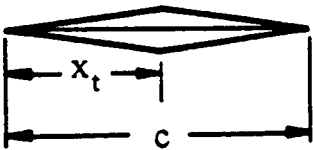
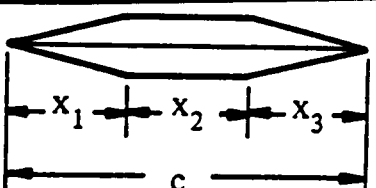
| $K$                          | Section                                                                              |
|------------------------------|--------------------------------------------------------------------------------------|
| $\frac{16}{3}$               |  |
| $\frac{c/x_t}{1 - x_t/c}$    |  |
| $\frac{c(c - x_2)}{x_1 x_3}$ |  |

Figure 80. Basic LV supersonic airfoils.

For structural and heat transfer reasons, it is desirable to slightly round the corners on all ridge lines such as are encountered on the diamond. [49] Figure 81 [45] shows some experimental results of  $t/c$  variations for supersonic airfoils on the cp location.

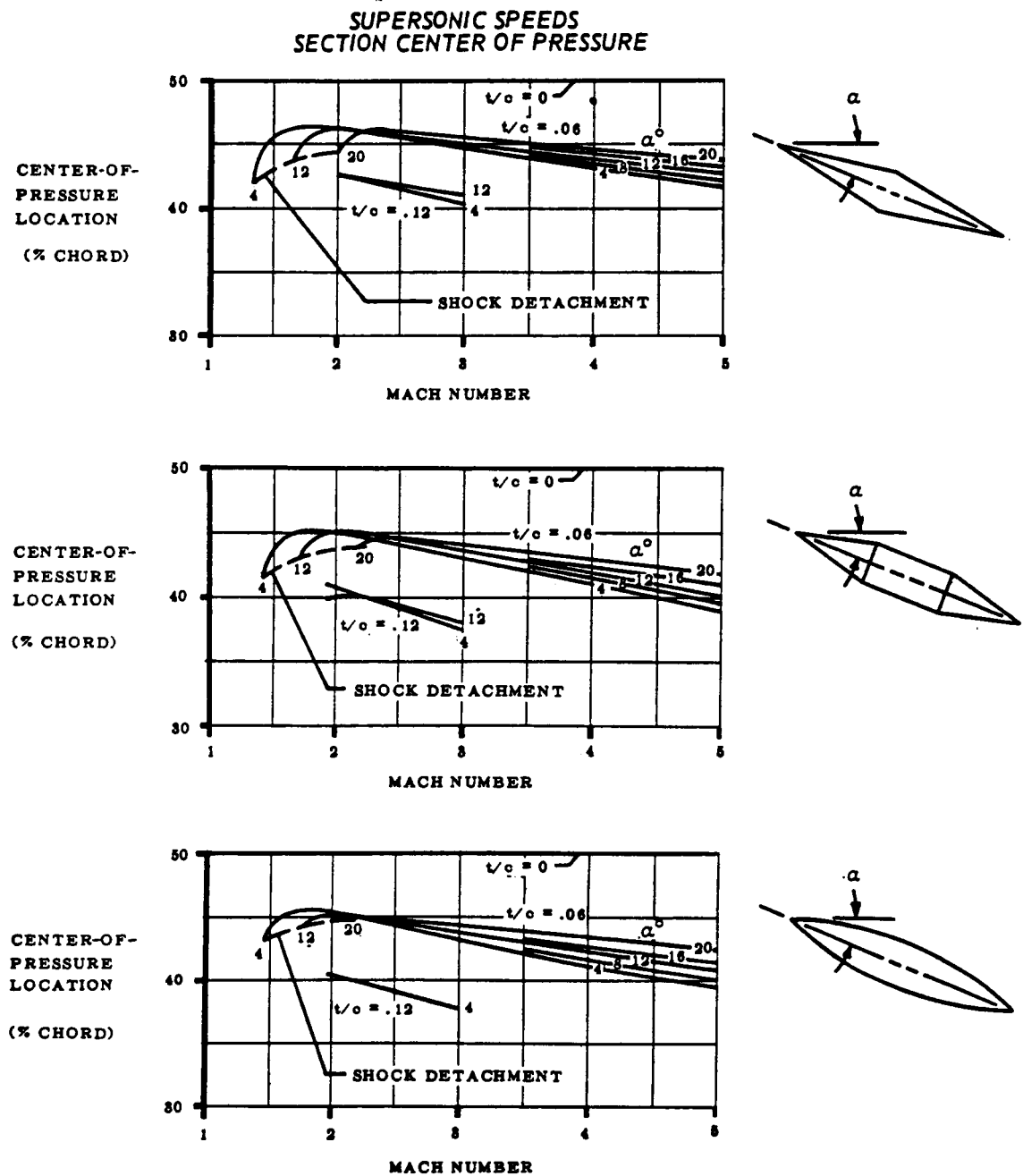


Figure 81. Effect of thickness on center of pressure location.

For both the diamond and the lenticular airfoils, the profile drag is lowest when the maximum  $t / c$  is beyond the 50% chord point. Equation (11) has shown that  $C_D$  increases with  $(t / c)$ , however, the  $t / c$  must also be chosen with concern for structural requirements and flutter.

Most of the analytic efforts in supersonic airfoil theory have been to quantify the wave drag attributed to the different airfoil parameters. [46] For this research, using a set of four FCA's at 80 ft<sup>2</sup> each, the total FCA  $S_F = 320$  ft<sup>2</sup>. The wetted area of the experimental LV has been calculated and it has been found that for the LV treated here,

$$\frac{S_F}{S_{\text{wetted}}} = 1.6\% \quad (12)$$

In addition, the six bipropellant liquid engines of the experimental LV produce 583,400 lb of vacuum thrust each, for a total, immense thrust of,

$$\text{Thrust}_{\text{total}} = 3,500,400 \text{ lb} \quad ; \quad \text{time}_{\text{FCA}} = 4 \text{ min} \quad (13)$$

The FCA time of flight is very short. From liftoff to the time of payload bay shroud ejection, at which time the FCA's and engines may also be ejected, is 4 minutes. The Shuttle aerodynamic flight control surfaces become effective approximately 10 seconds into the ascent trajectory. For the experimental LV, its trajectory data shows that 10 seconds corresponds to  $M = 0.2$ . From Figure 65 above, at  $M = 0.2$ ,  $Q = 100$  psf. After max  $Q$ ,  $Q$  falls off to 100 psf again at  $M = 3.8$  which means that the LV is through the usable dynamic pressure by  $M = 3.8$  which corresponds to 1.8 minutes of use of dynamic pressure by the FCA's. This means that the FCA's will be producing  $< 2\%$  of the total LV drag for 1.8 minutes. In view of equations (12) and (13), the author feels it is unnecessary to be overly concerned about the wave drag produced by the FCA's. Thus, since drag is not a major consideration, a variety of  $t / c$  are used for the FCA's designed here.

The controllability of a LV usually increases with slenderness ratio (length/diameter) due to the decreasing lift forces produced for a constant volume with increasing length hence lower aerodynamic moment, however the vehicle aeroelastic effects (to be discussed in Chapter XI) increase also. Table 2 shows a comparison of the slenderness ratios and of the area ratios of some of the key LV's studied thus far, the Redstone and Saturn family, and the experimental LV. It can be seen how closely the slenderness ratio of the experimental LV compares to that of these previous vehicles. Ratios of aerodynamic surface area to the total vehicle wetted areas are also shown along with the A's of the aerodynamic surfaces. Note the very close comparison of the test LV and the Saturn V in the second column.

Table 2. LV Slenderness Ratios and Area Ratios.

| LV              | Slenderness Ratio | $S / S_{\text{wetted}}$ | A        |
|-----------------|-------------------|-------------------------|----------|
| Redstone        | 10.8              | 0.0558                  | 0.843    |
| Saturn I        | 8.49              | 0.0793                  | 1.50     |
| Saturn IB       | 10.5              | 0.0337                  | 2.57     |
| Saturn V        | 11.0              | 0.0106                  | 3.4      |
| Experimental LV | 9.76              | 0.0161                  | Variable |

Blunt rather than sharp TE's can be used on a LV, since aerodynamic surface drag is normally < 2% of the total vehicle drag, as was the case on the Jupiter and many of the LV's covered in Chapters II and III. Blunt TE's have been used successfully in many previous LV's, as cited in Chapters II and III. The blunt TE was used for many reasons, including the ones mentioned in the previous chapters: avoidance of transonic problems, ease of fabrication, utility for holddown posts shown clearly in Figure 16 (Chapter II), structural rigidity to help prevent flutter, and, in the case of the X-15, for better high speed directional control (Chapter III). Blunt TE's have also been used to solve transonic problems on French sounding rockets such as the D6 (ONERA). ONERA in France had

the transonic problem of shock-induced boundary layer separation on the downstream part of airfoils used on the base of the first stage. This resulted in decreased stability and increased load factors. This was corrected by the use of thick TE's, and they found the wedge shape to experience no transonic problems and produce much improvement in static flight stability. [50]

Better control effectiveness is what is being sought here for the designed FCA's. As mentioned previously, none of the blunt TE surfaces of the Saturn series moved so the amount of increased effectiveness on a LV is not known. Can the increased control effectiveness of the blunt TE achieved on the X-15 be realized on a LV? The blunt TE also helps to avoid some transonic problems of fluctuating  $C_m$  and negative local lift due to transonic shocks. Figure 82 shows the pressure at the TE of the Saturn IB LV aerodynamic surfaces at transonic Mach numbers. [24]

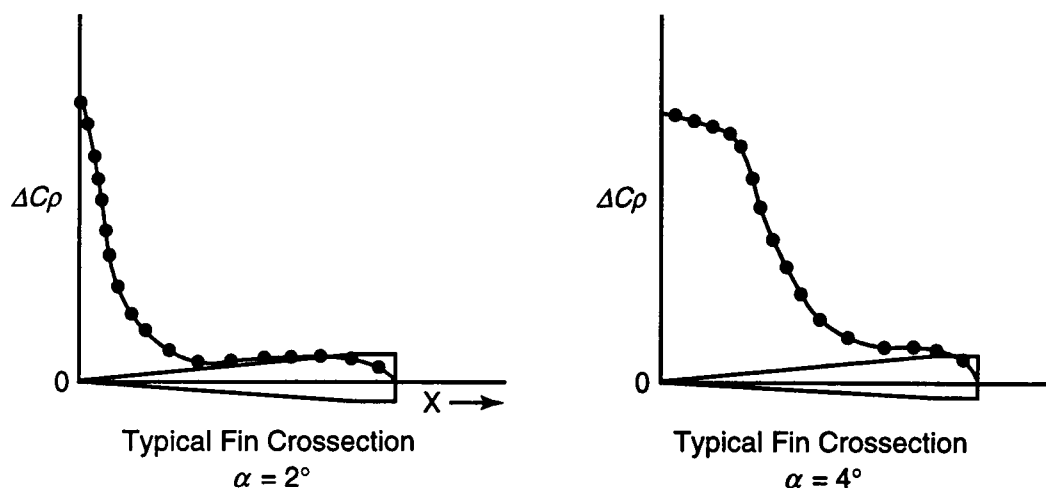


Figure 82. Pressure variation over Saturn IB aerodynamic surfaces from wind tunnel test.

The author desired to know whether the increased control effectiveness achieved with the blunt TE on the X-15 was achievable with all-movable FCA's for a LV. In addition, the author found the blunt TE structurally appealing for possible future insertion

of RCS and/or smart material actuator mechanisms due to the space available in a blunt TE airfoil. Therefore, the author's designs incorporate some sets of identical planforms, one with sharp TE's and one with blunt TE's to see if differences could be discerned in the wind tunnel testing.

Generally, planform LE sweep can be used at high subsonic and transonic speeds to delay the onset of drag-divergence. Use of a LE sweep angle ( $\Lambda$ ) increases the critical Mach number ( $M_{cr}$ ). The free-stream Mach number at which sonic velocity is first attained on some portion of the airfoil is the critical Mach number. The effect of  $\Lambda$  is to delay the effect of Mach number since only the velocity component perpendicular to the LE,  $M \cos \Lambda$ , provides influence on the pressures. Thus, the apparent Mach number is less than the actual Mach number.

For 2-D airfoils, the peak wave drag occurs near  $M = 1$ . Airfoil design philosophy which is based on linear theory assumes that the lift and drag of a flat airfoil is only a function of  $\Lambda$  and Mach number. From linearized supersonic flow theory,

$$\beta \equiv \sqrt{|M^2 - 1|} \quad (14)$$

and

$$C_p = \frac{2 \theta}{\beta} \quad (15)$$

where  $C_p$  = coefficient of pressure and  $\theta$  = local flow deflections in radians. The analysis of the pressure distribution over an airfoil usually assumes a subsonic lift distribution when the LE is subsonic, and a supersonic or uniform lift distribution when the LE is supersonic (Fig. 83) [49]



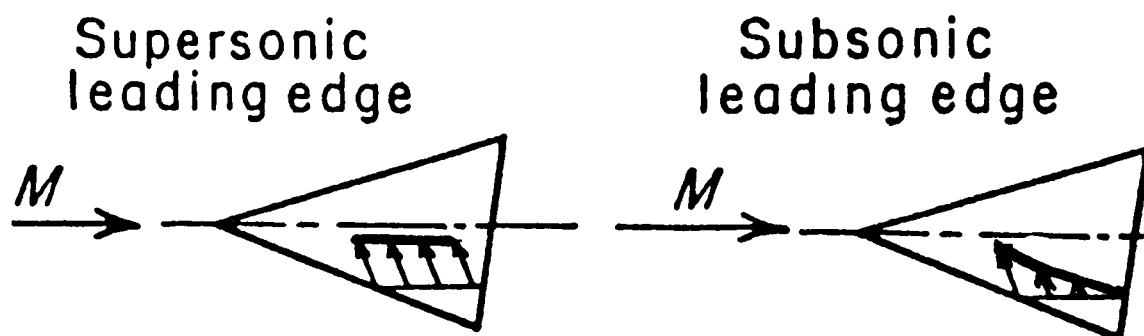


Figure 83. Pressure distributions over airfoils in supersonic flow.

When the LE becomes supersonic, i.e.,  $M \cos \Lambda > 1$ , then for triangular planforms, which are considered for the FCA's here,  $C_{L_\alpha}$  is the same as the 2-D value and is given by linear theory,

$$C_{L_\alpha} = \frac{4}{57.3} \frac{1}{\sqrt{M^2 - 1}} = \frac{4}{57.3 \beta} \quad (16)$$

For planforms of arbitrary  $\Lambda$ ,  $C_{L_\alpha}$  can be related to the 2-D linear theory  $C_{\ell_\alpha}$ , and for small  $\alpha$ ,  $C_{N_\alpha} \approx C_{L_\alpha}$ . Figure 84 [49] shows supersonic characteristics of  $\Lambda$ ,  $\beta$ , and taper ratio ( $\tau$ ) for a variety of planforms. In subsonic flow, [45]

$$C_{L_\alpha} = \frac{2 \pi A}{2 + \frac{A^2 \beta^2}{K^2} \left[ 1 + \frac{\tan^2 \Lambda}{\beta^2} \right] + 4} \quad (17)$$

where

$$K = \frac{C_{\ell_\alpha}}{2 \pi / \beta} \quad (\text{per radian})$$

For supersonic flow, Figure 85 [45] shows  $C_{N_\alpha}$  as a function of  $\Lambda$ .

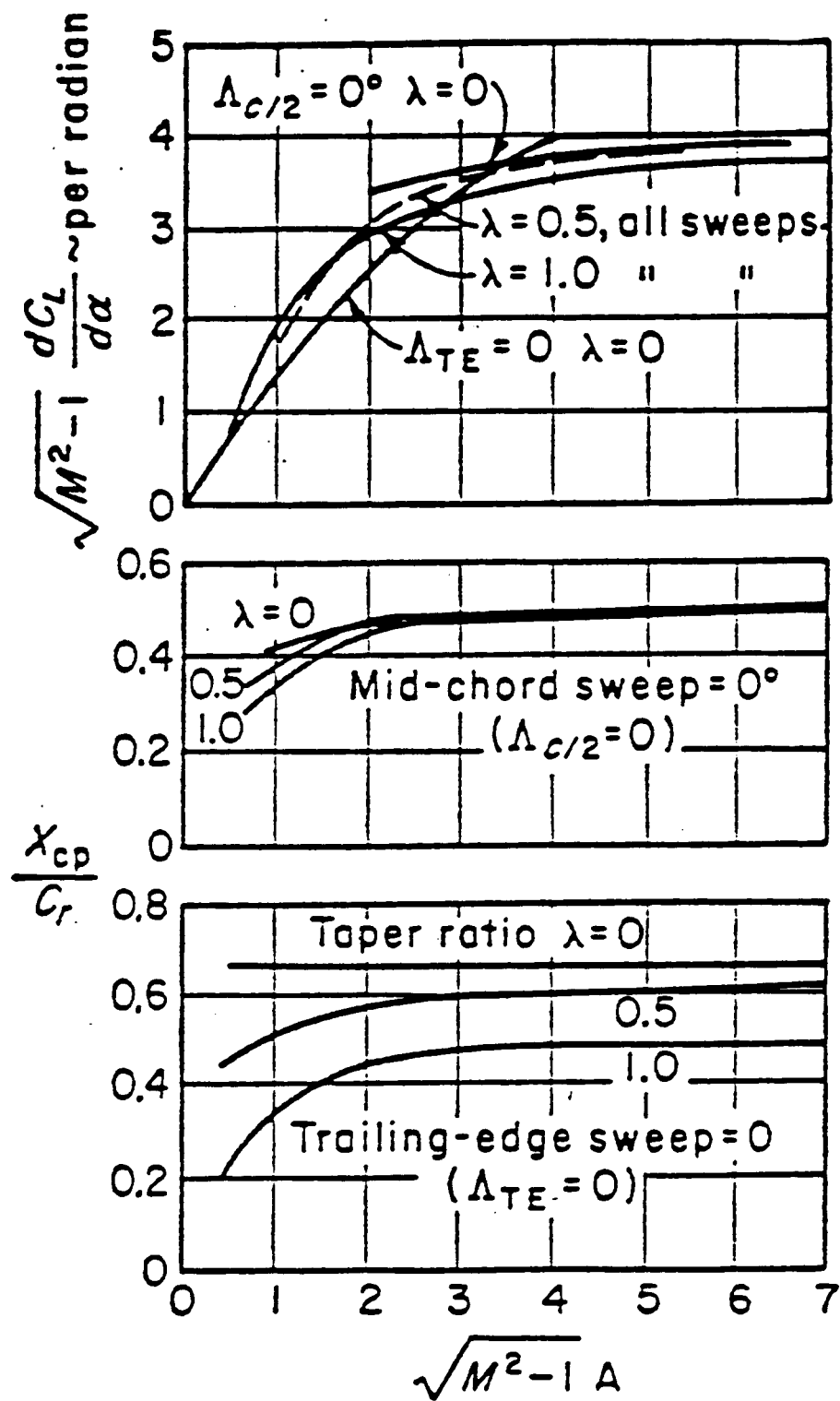
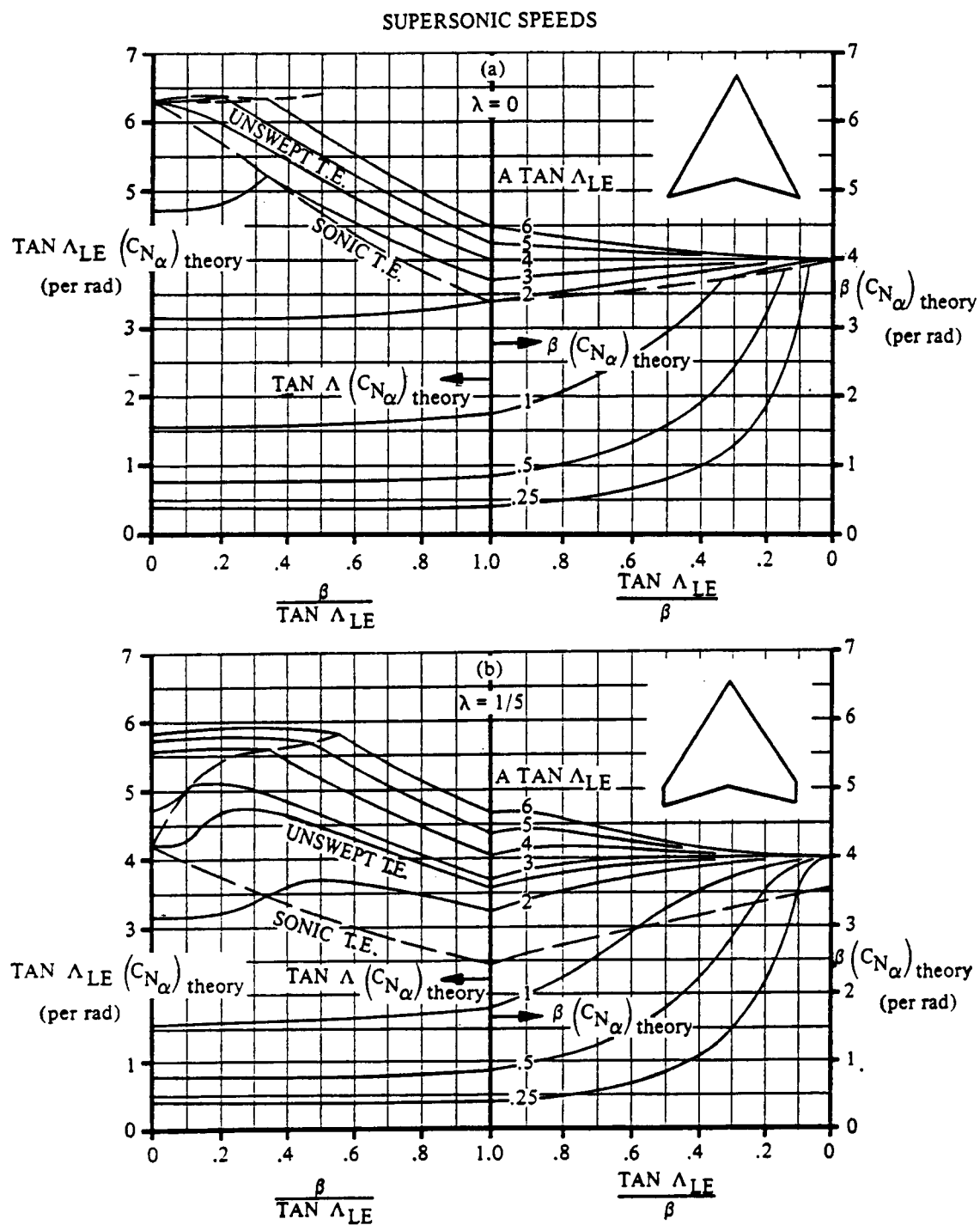


Figure 84. Supersonic characteristics of sweep and taper ratio.



For the experimental LV at the design point of  $M = 1.46$ ,  $\beta = 1.064$ . It is desired to maintain a supersonic LE to have the shock wave attached giving a steadier flow. To maintain a supersonic LE,

$$M \cos \Lambda > 1 \quad (18)$$

This governs the largest  $\Lambda$  that can be used. From equation (18) it is determined that the largest  $\Lambda$  to be used is  $45^\circ$ . Based on the work of Chapters II and III, the Saturn IB was a successful design using  $\Lambda$  of  $45^\circ$ . The Saturn V flew 13 successful missions and took us to the Moon using  $\Lambda = 30^\circ$  on its aerodynamic surfaces. Therefore, the two chosen LE sweep angles are  $\Lambda = 45^\circ$  and  $\Lambda = 30^\circ$ . Figure 86 [47] shows very little change in  $C_{L_\alpha}$  with forward or aft sweep.

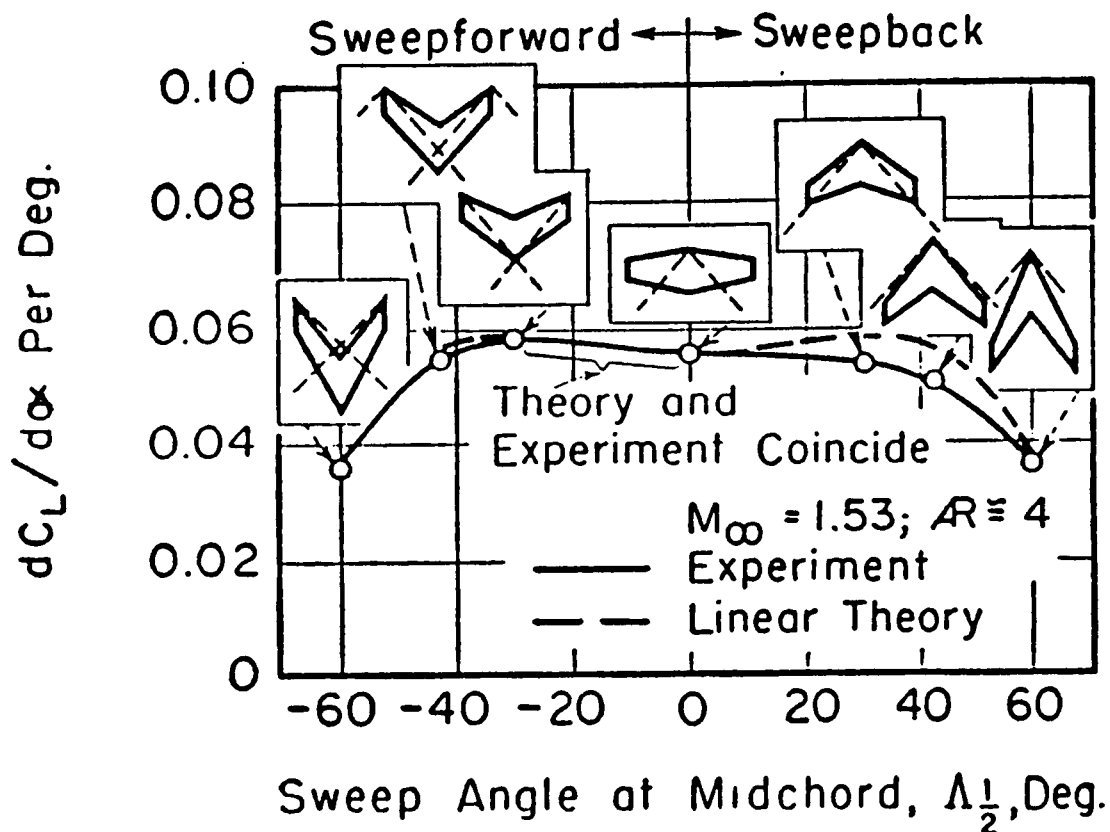


Figure 86. Comparison of experiment and theory for lift gradient of swept airfoils.

Figure 87 [49] show subsonic flow effects of taper ratio ( $\tau$ ) and aspect ratio (A). However, it has been found from wind tunnel tests at Arnold Engineering Development Center, that for transonic and supersonic flow over a missile body and a variety of fin planforms, that  $C_N$  was independent of  $\tau$ , and that for fins of constant area, changing  $b$ ,  $A$ , and  $\tau$  had relatively no effect on the body + fin  $C_N$ . [51]

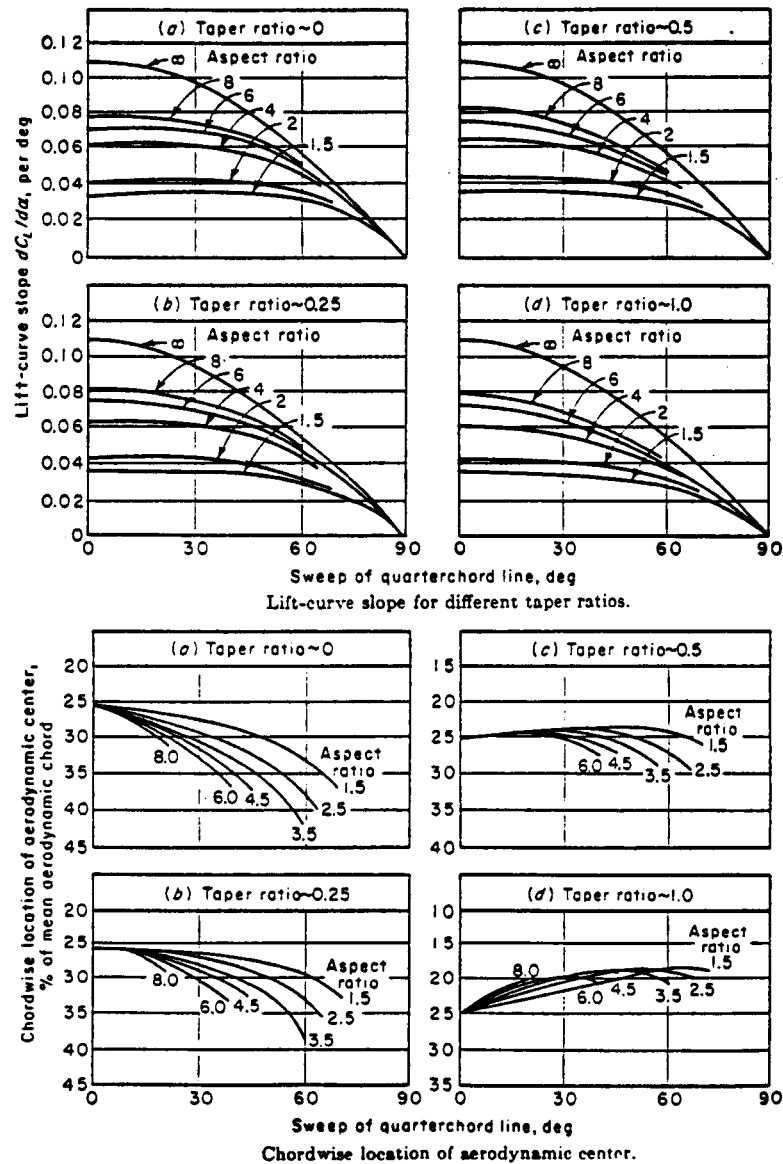


Figure 87. Taper ratio effects.

In considering  $A$ ,

$$A = \frac{b}{\bar{c}} = \frac{b^2}{S} \quad (19)$$

where

$b$  = span

$\bar{c}$  = average chord

For triangular planforms, which are considered for the FCA's,

$$A = 4 \cot \Lambda \quad (20)$$

hence, once  $\Lambda$  has been chosen so has  $A$ . Generally, increasing  $A$  will increase  $C_N$  for a fixed area, i.e., because of 3-D effects, the farther away from the fuselage the tip vortices are the more lift the aero surface can produce. Commonly used expressions are,

$$C_L = C_{\ell\alpha} \left[ \frac{A}{A+2} \right] \alpha \quad (21)$$

and

$$C_{L\alpha} = \frac{C_{\ell\alpha}}{1 + \frac{C_{\ell\alpha}}{\pi A}} \quad (22)$$

Since  $A$  is a function of  $b$ , a FCA with large  $b$  would have higher  $A$ , its centroid would protrude farther out into the free stream resulting in less body influence, and better  $C_N$ . In 3-D space there is a spreading of the effect from one part of the body to another. The generation of FCA lift on a body of revolution is due not only to its own lifting efficiency, but also to the influence of the LV body upwash field and the induced pressure field on the body from the presence of the FCA's. These effects can increase the

effectiveness of a given FCA by as much as 50 or 60%, depending on span. Local upwash varies (Beskin's approximation) as,

$$\left(\frac{\alpha_F}{\alpha}\right)_L = 1 + \left(\frac{R}{r}\right)^2 \quad (23)$$

and total upwash varies as

$$\left(\frac{\alpha_F}{\alpha}\right)_T = \frac{1}{\bar{c}} \int_R^{b/2} \frac{\left[1 + \left(\frac{R}{r}\right)^2\right] c}{b/2} dr \quad (24)$$

and

$$\frac{C_{L_{\alpha FU}}}{C_{L_{\alpha F}}} = \left(\frac{\alpha_F}{\alpha}\right)_T \quad (25)$$

where

$R$  = body radius

$r$  = distance from fuselage centerline to point on FCA

$C_{L_{\alpha FU}}$  = lift curve slope of FCA with upwash

In addition, an airfoil with a small chord has lower hinge moments which would reduce control actuator requirements. For a fixed planform area, a smaller chord implies a larger span. However, LV aerodynamic surface planforms are generally selected to optimize launch pad clearance problems which limits the span  $b$ . Trapezoidal planforms minimize  $b$  and launch pad clearance problems. There is a trade between the aerodynamic benefits of large  $b$ , hence  $A$  and  $C_N$ , and the practicality of launch pad logistics.

#### **5.4 Flight Control Augmentor Designs**

While there is some preference for triangular planforms for supersonic speeds because of smaller  $c_p$  shifts in going from subsonic to supersonic speeds, more vortex lift and more negative pitching moments, and there is some preference for trapezoidal planforms for subsonic speeds, the recent literature shows that wind tunnel test is still being used extensively to make final decisions on planform shape parameters and also to increase data bases so that current aerodynamic prediction codes can be further refined. Reference 52 outlines the wind tunnel tests performed at NASA LaRC to assess the effects of a variety of planforms, including both triangular and rectangular, on a cruciform missile configuration. Reference 53 is a wind-tunnel investigation performed at NASA LaRC of the aerodynamic characteristics of two missile configurations having a variety of trapezoidal and triangular wings and canard controls. Reference 51 describes the wind tunnel test program of a variety of missile planform shapes in order to provide a parametric variation and establish a data base from which to write a predictive computer code. The fine distinctions of the lift produced, among planform designs differing only in taper ratio, thickness-to-chord ratio, and supersonic airfoil choice, can sometimes better be determined by wind tunnel test (or flight test) than by theoretical prediction methods.

Most of the important missile development efforts from World War II through the 1960's were based on experience gained through wind tunnel test and flight test. Figure 6 in Chapter II shows the 16 planforms that were tested on the V-2. Chapter III shows the many X-15 empennage planforms that were tested, along with the constantly evolving planform shapes and areas of the aerodynamic surfaces of the Saturn series of LV's. Analytic techniques were found inadequate for the design process until the late 1960's when larger wind tunnel test data bases were available. [29] Most of the aerodynamic, and stability and control computer prediction codes today are heavily based on wind tunnel test data and are classified as semi-empirical. Because these codes are based on wind tunnel test data, each code has a range of applicability and set restrictions which limits the designer. [54] Chapter IV has shown the variety of LV aerodynamic



surfaces in current use by a number of other nations. Reference 55, "Fin-Planform Effects on Lift and Center of Pressure for Supersonic Missiles," outlines work done in 1993 using component buildup and paneling methods, and states "Considerable research is still needed to improve and extend prediction tools."

Based on all of the studies and analyses thus far, several aft FCA planforms, all of the same required planform area, have been designed for the experimental LV. The planform area, that was selected above, is 80 ft<sup>2</sup>. A total of 12 candidate FCA's have been designed for wind tunnel test: 11 aft FCA's and one forward canard FCA to be described below. Since the critical design point is supersonic, 8 triangular family aft FCA's and a triangular canard were chosen. Based on Saturn IB LV success (Chapters II and III), 4 have  $\Lambda = 45^\circ$ ; based on Saturn V LV success (Chapters II and III), 4 have  $\Lambda = 30^\circ$ ; these LV's also had supersonic LE's at the critical design Mach number. The 8 in the triangular family were all originally true triangles, however for launch pad logistics (and structural/fabrication considerations) some of the tips were clipped. These clipped tip designs are of course strictly trapezoids. The total consist of a variety of diamond, half diamond, and lenticular airfoils; both blunt and sharp TE's, both long spans, and clipped tip for launch pad clearance. For comparison to the triangular planforms, an aft swept, forward swept, and trapezoidal tapered planform all with diamond airfoils have been used. Aspect ratios range from 1.74 to 5.625.

Figure 88 lists the FCA planform design candidates for test. Figure 89 is a summary planform shape sheet of the FCA surface designs. More detailed, dimensioned, scale drawing can be found in Appendix A.4. While aerodynamic surfaces have been used for flight stability on several previous LV's, and small tip controls have sometimes been used at the tips of these as outlined above, this is the first proposed use of all-movable flight control augmentors on a large Saturn-class LV to the author's knowledge.

## ***Candidate FCA Designs***

### **12 FCA Designs wind tunnel tested:**

- 4 triangular family planforms;  $L E \Lambda = 45^\circ$ ;  $T E = 0^\circ$ 
  - 1 full diamond airfoil (#1)
  - 1 half diamond; blunt  $T E$  (#2)
  - 1 full diamond; clipped tip (#3)
  - 1 Lenticular airfoil; (#4)
- 4 triangular family planforms;  $L E \Lambda = 30^\circ$ ;  $T E = 0^\circ$ 
  - 1 full diamond airfoil (#5)
  - 1 half diamond; blunt  $T E$  (#6)
  - 1 full diamond; clipped tip (#7)
  - 1 Lenticular airfoil; (#8)
- 1 aft swept trapezoidal planform;  $L E \Lambda = 45^\circ$ ;  $T E = 30^\circ$  (#9)
- 1 forward swept trapezoidal planform;  $L E \Lambda = 45^\circ$ ;  $T E = 55^\circ$  (#10)
- 1 tapered trapezoidal planform;  $L E \Lambda = 45^\circ$ ;  $T E = 30^\circ$  (#11)
- 1 triangular planform canard; half diamond; blunt  $T E$  (#12)

Figure 88. Candidate flight control augmentor design descriptions.

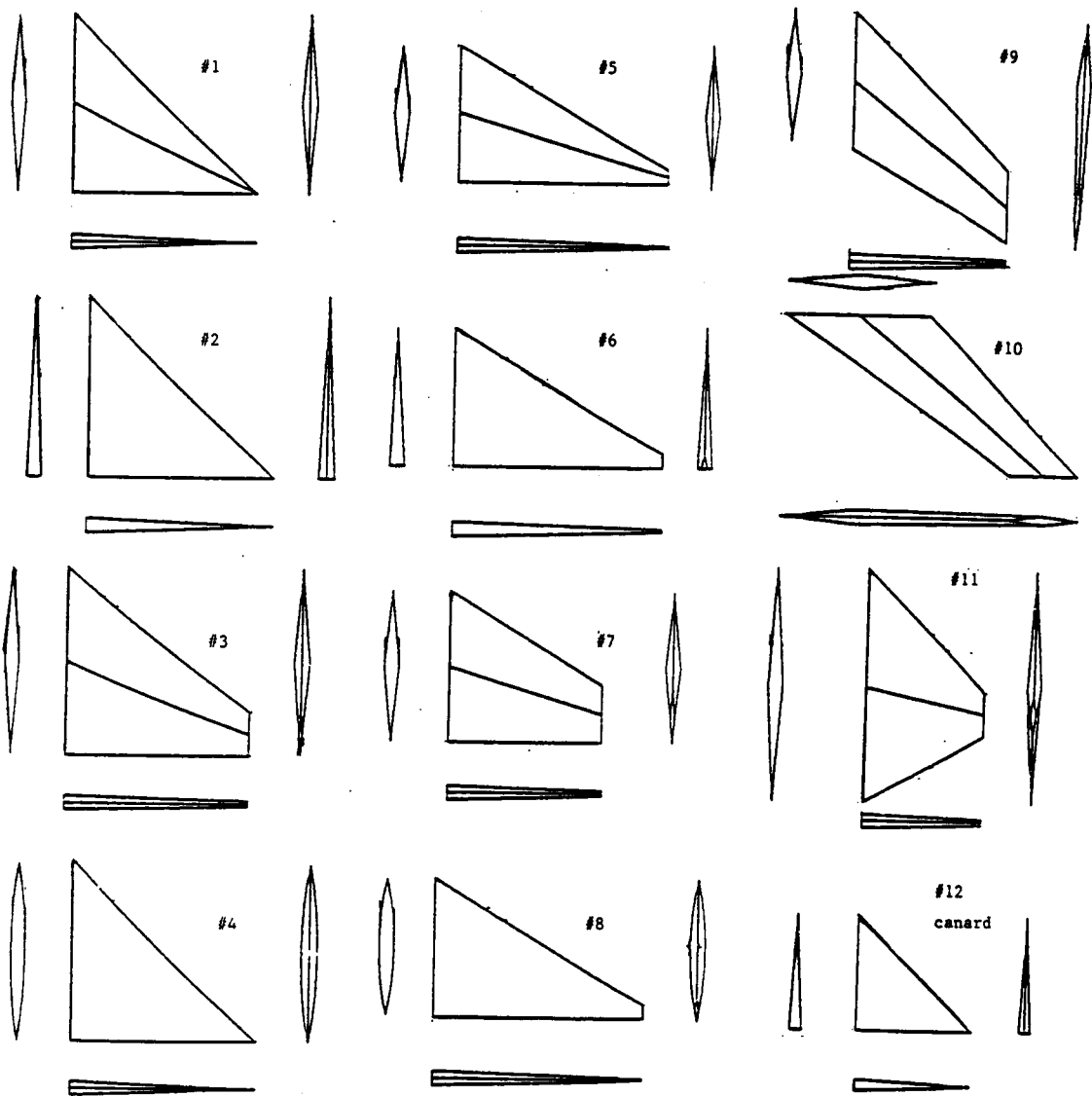


Figure 89. Candidate FCA designs.

## 5.5 Canard Considerations

It is well known that an aerodynamic flight control surface normally located forward of the cg on a flight vehicle produces an effect which is destabilizing to its static stability. Although canards cause static instability, they have been used extensively on subsonic and supersonic aircraft and missiles for a variety of reasons. Some advantages provided by canard usage include: production of lift at the nose, high maneuverability due to fast response time, and reduced trim drag in cruise flight. For example, a canard has been investigated for the Shuttle Orbiter to produce an additional nose-up pitching moment, which would allow a more forward cg location, and trim out the nose-down pitching moment created by the down-elevon deflection. The Orbiter forward cg is limited to 0.65 reference body length (Operational Design Data Book, Rockwell, 1991), which restricts its payload carrying capability. Some of these studies have been conducted by NASA ARC and published in 1993. [56] Deployable canards have also been studied to provide positive (nose-up) pitch authority in the landing phase and relax the pilot workload. [57]

The objective of this research is to augment the deficient control torque due to an aft cg location. The traditional use of canards is to treat forward cg problems, and traditional usage would exacerbate an aft cg problem. However, a nontraditional canard control can be used here to provide negative lift on the nose and a pitch-down torque which would achieve the control objectives. For the application here, the vortices from the canards onto the LV empennage must be considered, as the canard downwash acting on the empennage can reduce the FCA control effectiveness. Since the location of the FCA's on the experimental LV has been selected, based on the location of the six engines, and four engine shrouds, the canards are aligned with the FCA's in the same manner. The biconic nose of the experiment LV has been redesigned to provide cleaner and more uniform flow in proximity to the canard controls. The 1.2 caliber ogive nose accommodates the canards as shown below. The longitudinal body station location of the

canards is sometimes a trade between distancing from the bow and oblique shocks and their effects on the control authority of the canards, and the structural considerations of having the canards located at a major bulkhead. The canards have been located on the payload bay bulkhead for structural considerations.

## 5.6 Canard Designs

For completeness in this research, a set of cruciform canards has been designed by using linearized supersonic theory for the canard lift coefficient, along with the same moment balance method used above and shown in Figure 90. Using equation (1), in the same manner as in section 5.2 for the aft FCA's, the all-movable canard planforms which will provide the required controllability are sized,

$$CR = \frac{CT + N_C \ell_C}{AT} = \frac{T \sin \delta_E (E) \ell_E + C_{N_C} Q^2 S_C \ell_C}{C_{N_\alpha} \alpha Q S \ell_{cp}} \geq 1.5 \quad (26)$$

and

$$C_{N_C} = \frac{4 (\alpha + \delta_C) \text{ (in radians)}}{\sqrt{M_\infty^2 - 1}} \quad (27)$$

where  $N_C$  = normal force provided by canard  
 $C_{N_C}$  = canard normal force coefficient  
 $S_C$  = planform area of one canard in a horizontal position  
 $\ell_C$  = moment arm from LV cg to ac of canard  
 $\delta_C$  = deflection angle of the all-movable canard in degrees

It is assumed here that the ratio of local dynamic pressures at the canard location is very close to 1, that the ac of the canard is near the half-chord location (which is the conservative assumption for the canard), and that the moment about the canard ( $C_{m_{ac}}$ ) is near zero, i.e., that the canard is symmetrical and without camber.

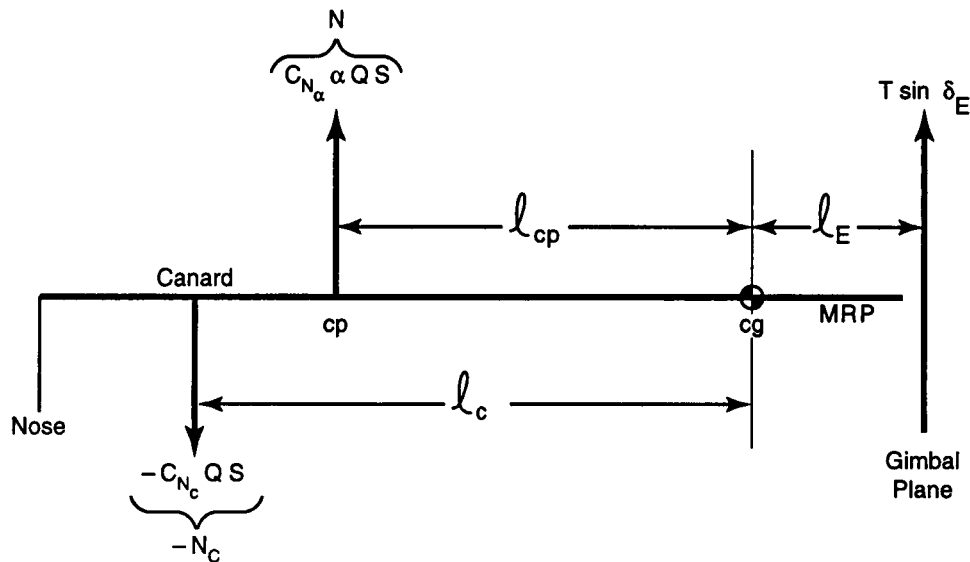


Figure 90. Canard control augmentation.

For the experimental LV, equation (27) becomes,

$$C_{N_c} = 4.024 (0.06981 + \delta_c) ; \quad \text{and} \quad l_c = 147.64 \text{ ft}$$

When these values are substituted into equation (28) along with the values used in section 5.2,

| $\delta_c^\circ$ | $S_c \text{ (ft}^2\text{)}$ |
|------------------|-----------------------------|
| 7                | 43.74                       |
| 8                | 32.80                       |
| 9                | 26.24                       |

Choose  $S_c = 32 \text{ ft}^2$

Figure 91 shows the designed triangular 32 ft<sup>2</sup> canard planform and the fuselage location selection factors. One set of four in line with the aft FCA locations are used. They have blunt TE's, are hinged at their TE's and located on the LV fuselage 1.2 calibers from the tip of the ogive nose. This location on the payload bay forward bulkhead is not only advantageous for structural reasons, but it is believed the canards will be free of the bow

or oblique shock wave emanating from the LV nose. It is also felt that this location is far enough forward on the fuselage to prevent any serious blanketing of aft FCA's, if they are used simultaneously, from the canard vortex downwash. Table 3 summarizes the canard design parameters used for simplicity.

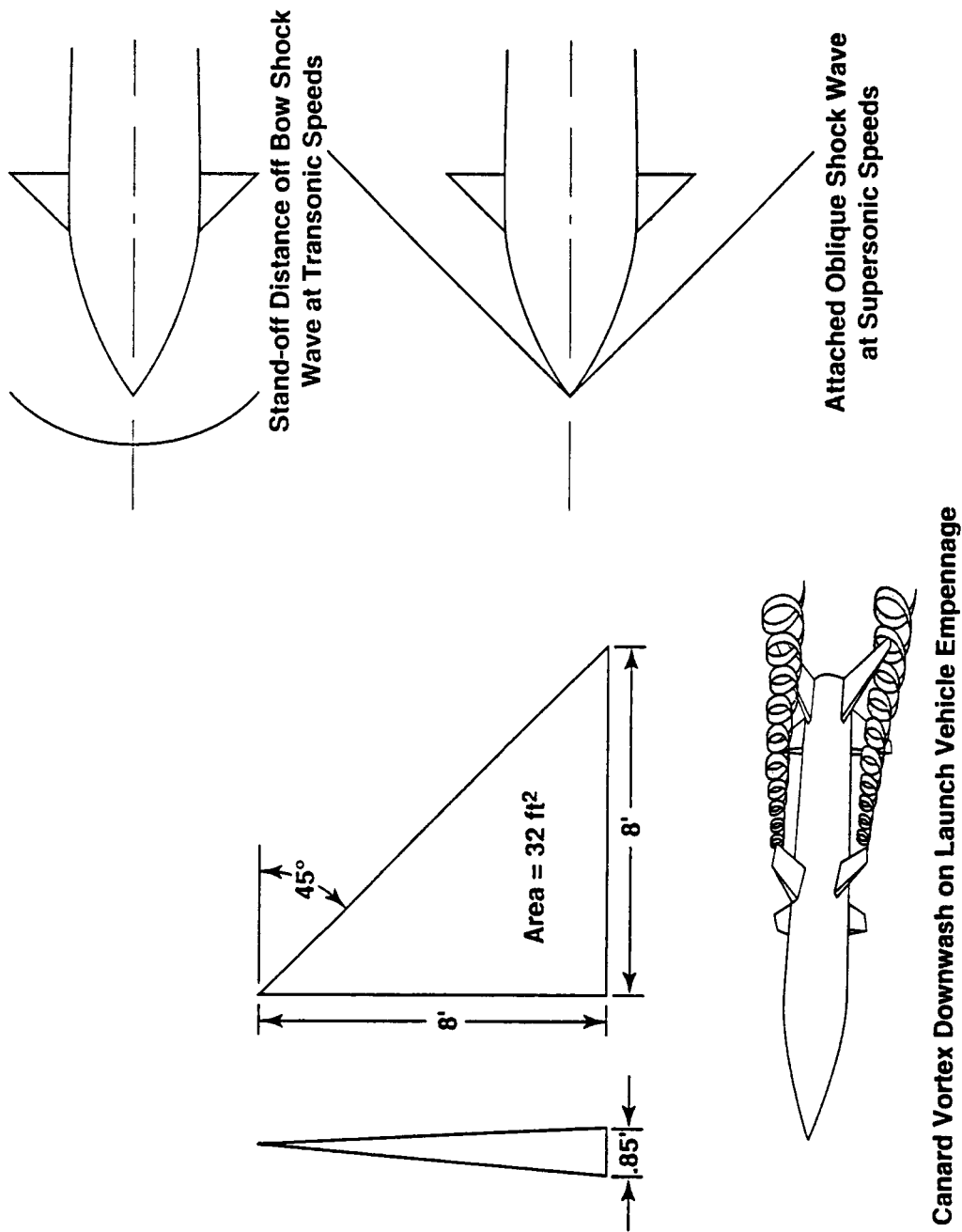


Figure 91. Canard surface planform design and factors in selecting location on launch vehicle fuselage.

Table 3. Canard design parameters.

|                                                     |
|-----------------------------------------------------|
| Redesigned nose to accommodate canard controls:     |
| Bi-conic redesigned to 1.2 caliber ogive nose       |
| Canard Controls:                                    |
| Four Cruciform Triangular Planforms                 |
| Span = 16 ft                                        |
| Root Chord = 8 ft                                   |
| Leading Edge Sweep = 45°                            |
| Airfoil = 0.85 ft ramp                              |
| Longitudinal Placement = 1.2 calibers from nose tip |

### 5.7 Options for Increased Control Effectiveness

The Shuttle aerodynamic flight control surfaces become effective approximately 10 seconds into the launch ascent trajectory, at which time  $Q$  is of sufficient magnitude. As an added way to increase the torque provided by the designed FCA's, RCS could be employed in the blunt TE's or the concept of jet flaps could be employed, since the blunt TE's provide additional space. In the jet flap concept, a propulsive jet can be emitted from the TE of the FCA, as shown in Figure 92, with the primary objective of increasing the lift of the surface, hence control moment. Also, a recent investigation (Arena, Nelson, and Schiff; 1993) has shown that an effective technique for generating control torque when conventional control surfaces lose their effectiveness is to use pneumatic blowing through a chined forebody. Again, these additions could be explored if the surfaces were needed to be more effective sooner than 10 seconds into the ascent, e.g., for very reduced wind restrictions at liftoff. In the upper portion of the ascent, once the FCA's have ceased to be effective they can be dropped off perhaps with the engines, since this is an expendable LV.



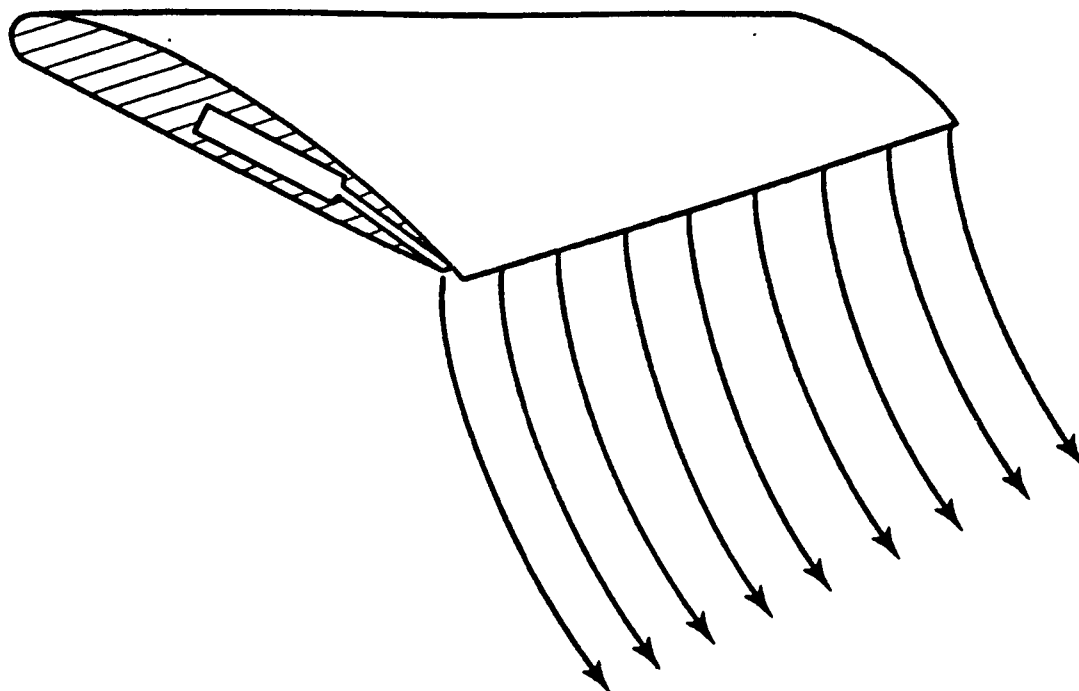


Figure 92. Flight control augmentor with internally ducted jet.

Now that the candidate FCA's have been designed, they will be challenged to provide sufficient control torque augmentation on an even farther aft cg. Updated mass properties and an updated ascent trajectory have been generated which reflect a very far aft cg, and represents a worst possible case of aft cg, which will be treated in the remainder of this research. These can be found in Appendix A, and will surely put the FCA's to the test of providing the required augmentation to control the experimental LV.

## CHAPTER VI. STATIC WIND TUNNEL TEST

### 6.1 Design of the Static Stability Wind Tunnel Test Program

The purpose of the Static Stability Wind Tunnel Test Program conducted at the NASA MSFC wind tunnel test facility was to obtain the necessary aerodynamic data and stability derivatives for the experimental LV and the designed FCA's so that a static flight stability analysis could be performed, and to determine the best candidate designs for the follow-on dynamic stability test program. The author, as Principal Investigator on this research, designed the wind tunnel test program requirements and was actively involved throughout the wind tunnel test program. Figure 93 shows an outline of the test requirements: center-of-pressure (cp) location on the vehicle; coefficients of normal, axial, and side forces; coefficients of pitching, yawing, and rolling moments; and partial derivatives of these with respect to  $\alpha$ ,  $\beta$ , and the FCA deflections,  $\delta_F$ . Of the items listed in the static requirements, only three forces and three moments were actually measured in the wind tunnel test program. All the other parameters were obtained by the author's data reduction and data analysis efforts which constituted a sizable time segment of this research and are detailed in the following chapter.

The static stability test program was conducted at Mach numbers of 0.2, 0.4, 0.8, 0.95, 1.2, 1.46, 1.96, 2.74, 3.48, and 4.96. All aft FCA's were tested to deflections of up to  $15^\circ$ , forward FCA's were tested to deflections of up to  $30^\circ$ ;  $\alpha$  and  $\beta$  were  $\pm 15^\circ$ . For comparison, a base run was made at each Mach number using the experimental LV test article without FCA's. The three forces and three moments were measured on a six-component (normal, axial, and side forces; pitch, yaw, and roll moments) strain-gauge balance mounted internally in the experimental LV test article (model) and detailed below. A sample run sheet, which is the devised matrix of test parameters, can be seen in Appendix B. To insure a turbulent boundary layer over the model, silicone carbon grit,

## Design of Wind Tunnel Test Program

### Requirements:

$\alpha, \beta$ , range  $\pm 15^\circ$ ; Aft FCA  $\delta_F + 15^\circ$  in  $3^\circ$  increments;  
Forward FCA  $\delta_c + 30^\circ$  in  $5^\circ$  increments  
Mach range .2–5.0; Base run no FCA's

### Static Tests:

$X_{cp}, C_N, C_A, C_Y, C_m, C_n, C_l$ ;

$\frac{\partial C_N}{\partial \alpha}, \frac{\partial C_m}{\partial \alpha}, \frac{\partial C_Y}{\partial \beta}, \frac{\partial C_n}{\partial \beta}, \frac{\partial C_l}{\partial \beta}$

$\frac{\partial C_N}{\partial \delta_F}, \frac{\partial C_A}{\partial \delta_F}, \frac{\partial C_Y}{\partial \delta_F}, \frac{\partial C_m}{\partial \delta_F}, \frac{\partial C_n}{\partial \delta_F}, \frac{\partial C_l}{\partial \delta_F}$

### Dynamic Tests:

$\frac{\partial C_N}{\partial \dot{\alpha}}, \frac{\partial C_m}{\partial \dot{\alpha}}, \frac{\partial C_N}{\partial q}, \frac{\partial C_m}{\partial q}, \frac{\partial C_N}{\partial \delta_F}, \frac{\partial C_m}{\partial \delta_F}, etc.$

Figure 93. Design of wind tunnel test program.

size #100, was applied as a 3-D roughness on the nose beginning at  $M = 2.74$ . It has been determined from years of wind tunnel testing experience in the MSFC wind tunnel and observation of tunnel turbulence levels, that from the discrete Mach numbers listed above this is the Mach number where the boundary layer trip is required. The silicone carbon grit applied to the model is shown in photographs in Appendix B. Flow field and shocks were recorded on videotape by using the wind tunnel Schlieren system to provide qualitative information about the shock patterns. The Schlieren images are proportional to the density gradient, and therefore provide good viewing of both the location of shock waves on the model and of complicated boundary layer phenomena in supersonic flow.

## **6.2 NASA MSFC Wind Tunnel Test Facility**

The MSFC wind tunnel used in this research was the 14×14-Inch Trisonic Wind Tunnel. The same 14-inch tunnel has also be used to test the Pershing, Redstone, Jupiter, and Saturn series outlined in Chapters II and III. This wind tunnel is an intermittent, trisonic, blowdown tunnel with interchangeable test sections. During this research, the author was involved throughout the testing and tunnel changeout procedures and had the opportunity to become familiar with this tunnel. The test section is 14 inches x 14 inches x 20 inches. In the transonic range, interchangeable fixed contour blocks were used to form nozzles. In the supersonic range, fixed contour plates positioned by hydraulic screw jacks were used to form the nozzles. The Mach range is from 0.2 to 5.0; dynamic pressure range is from 0.5 to 20 psia; stagnation pressure range is from 22 to 105 psia. Reynolds number (Re) range is from 1 to 18 million/ft; stagnation temperature range is from ambient to 200 °F; the run times are 1.5 to 2 minutes for transonic, and 0.75 to 0.83 minutes for supersonic; the recharge time is 10 minutes for transonic and 20 minutes for supersonic; the  $\alpha$  and  $\beta$  ranges are  $\pm 10^\circ$  with added range available with the use of offset stings.

Speeds are varied in the subsonic range by a controllable diffuser; in the transonic range by the perforated tunnel walls; in the low (1.5 to 2.5) supersonic range by interchangeable contour nozzle blocks; and in the higher (2.75 to 5.00) supersonic range by tilting fixed contour blocks (Fig. 94). [58] The transonic section has variable porosity walls that allow for optimum shock wave cancellation in the transonic flow region. The 5/32 inch holes are inclined 30° with respect to the flow direction. The porosity of the walls is varied remotely by the use of a double wall arrangement from 0% to 5.4%. It is this feature that makes possible the cancellation of shock wave reflections, hence better data accuracy in the transonic Mach range (to 1.30) than would be possible with fixed porosity walls where wall corrections would be needed. An age old fundamental

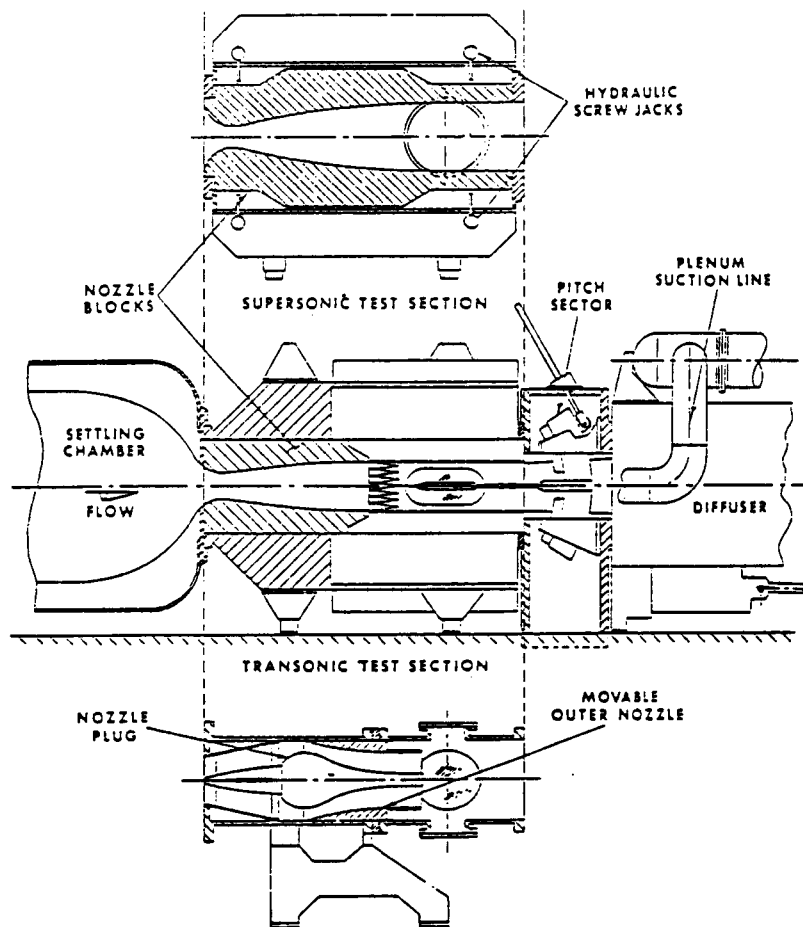


Figure 94. Interchangeable test sections of wind tunnel used for the testing.

difficulty that arises in wind tunnel test at transonic speeds is that the shock waves reflect from the tunnel walls back onto the model, since the shock waves are inclined at a large angle to the velocity direction in transonic flow. The MSFC variable porosity walls allow for optimum wave cancellation in the transonic flow region, thus eradicate the need for wall interference corrections. The variable diffuser, with its movable floor and ceiling panels, is the primary means for controlling the subsonic speeds.

The sector assembly and diffuser telescope to allow easy access to the model and test section. An automatic Mach controller maintains a constant Mach number in the transonic range by controlling the plenum suction. The wind tunnel test section has been calibrated with  $10^\circ$  cone/cylinders. There are a set of four brass cone/cylinders which represent 1%, 2%, 3%, and 4% tunnel blockage, respectively. The pressure distribution over the cone/cylinder in the MSFC wind tunnel was compared to predetermined pressure distributions from other calibrated wind tunnels. The variable wall porosity has been adjusted accordingly to match the pressure distributions. It has been determined that the tunnel blockage is  $\leq 2\%$ .

### **6.3 Instrumentation**

For the static stability wind tunnel testing, the strain gauge balance used was a six-component, internally mounted, MSFC #250, which was 1/2 inches in diameter by 3-1/2 inches in length, and was used to measure three forces and three moments. Table 4 lists the accuracy and maximum loads for this balance, along with the resulting tolerances on the aerodynamic coefficients at the tested Mach numbers. The abbreviations used in Table 4 are the standard MSFC wind tunnel test data computer printout format: NF = normal force; SF = side force; AF = axial force; PM = pitching moment; RM = rolling moment; YM = yawing moment; CN = coefficient of normal force; Cm = coefficient of pitching moment; CA = coefficient of axial force; CY = coefficient of side force;

Cyn = coefficient of yawing moment (this use of extra letters has been the wind tunnel printout format custom to prevent confusion with multiple-segmented bodies); Cbl = coefficient of rolling moment.

Table 4. Strain gauge balance 250.

| Capacity |           | Accuracy    | Max Loads* |
|----------|-----------|-------------|------------|
| NF       | 200 lb    | +1.00 lb    | 21 lb      |
| SF       | 107 lb    | +0.50 lb    | 4.5 lb     |
| AF       | 75 lb     | +0.25 lb    | 12 lb      |
| PM       | 200 in-lb | +0.99 in-lb | 14 in-lb   |
| RM       | 50 in-lb  | +0.25 in-lb | 1.5 in-lb  |
| YM       | 107 in-lb | +0.49 in-lb | 16 in-lb   |

\*Maximum estimated running loads and not starting loads.

#### Coefficient Tolerances

| Mach No. | Dyn. Press. | CN     | Cm     | CA     | CY     | Cyn    | Cbl    |
|----------|-------------|--------|--------|--------|--------|--------|--------|
| 0.2      | 0.58        | 0.8057 | 0.4819 | 0.2014 | 0.4028 | 0.2385 | 0.1217 |
| 0.4      | 2.21        | 0.2114 | 0.1265 | 0.0529 | 0.1057 | 0.0626 | 0.0319 |
| 0.8      | 6.47        | 0.0722 | 0.0432 | 0.0181 | 0.0361 | 0.0214 | 0.0109 |
| 0.95     | 7.76        | 0.0602 | 0.0360 | 0.0151 | 0.0301 | 0.0178 | 0.0091 |
| 1.2      | 9.15        | 0.0511 | 0.0305 | 0.0128 | 0.0255 | 0.0151 | 0.0077 |
| 1.46     | 9.49        | 0.0492 | 0.0295 | 0.0123 | 0.0246 | 0.0146 | 0.0074 |
| 1.96     | 11          | 0.0425 | 0.0254 | 0.0106 | 0.0212 | 0.0126 | 0.0064 |
| 2.74     | 6.38        | 0.0732 | 0.0438 | 0.0183 | 0.0366 | 0.0217 | 0.0111 |
| 3.48     | 5.15        | 0.0907 | 0.0548 | 0.0227 | 0.0454 | 0.0269 | 0.0137 |
| 4.96     | 2.73        | 0.1712 | 0.1024 | 0.0428 | 0.0856 | 0.0507 | 0.0259 |

The sting used was a MSFC #82 with an overall length of 21 inches, 5/8 inch diameter at the model end, and 3/4 inch diameter at the balance end. The  $\alpha$  and  $\beta$  sweeps are conducted in 2° increments to  $\pm 10^\circ$  with additional range to  $\pm 15^\circ$  with sting offsets.

The wind tunnel blockage for the model was determined to be  $\approx 1.5\%$  based on comparison with the tunnel calibration cone/cylinder.

The data acquisition system is a Hewlett-Packard 349A controller with a digitizer. The unit is equipped with various modules for facility system control,  $\alpha$  readout, Scanivalve® (scanning voltmeter) control, etc. Currently, the system is configured to 40 low-level strain gauge, thermocouple, or pressure channels per second, with a 2 to 3 second recycle time to change  $\alpha$  and allow for settling. Low pass filters are available for all channels and are routinely used on strain gauge balance channels. More detailed information on this MSFC facility, the instrumentation, and data handling processes can be found in Reference 54.

#### **6.4 Fabrication of Test Articles**

The test articles were fabricated of stainless steel to a scale of 0.5% by Dynamic Engineering Incorporated. Figure 95 shows a diagram of the test LV with its 1.2 caliber ogive nose redesign, engine shrouds, aft FCA's, balance center (B.C.), and the Moment Reference Point (MRP) the author chose to use in this test. This MRP represents the vehicle aft cg location at max Q. Figure 96 shows the drawing of the test LV model with the attachment of the forward canards (set of four), and aft FCA's (set of four) on their corresponding engine shrouds. Figure 97 shows the test article details of the aft FCA's. Figure 98 is a photograph of the fabricated test LV with a set of aft FCA designs. Figure 99 is a photograph of the fabricated 11 aft FCA designs tested. Figure 100 is a photograph of the fabricated test LV with canards and aft FCA's. Photographs of details of all aft FCA's, along with several views of the fabricated canards (all of the same design; each insert is set at a prescribed deflection angle), and additional photographs of the test LV in various configurations can be seen in Appendix B.



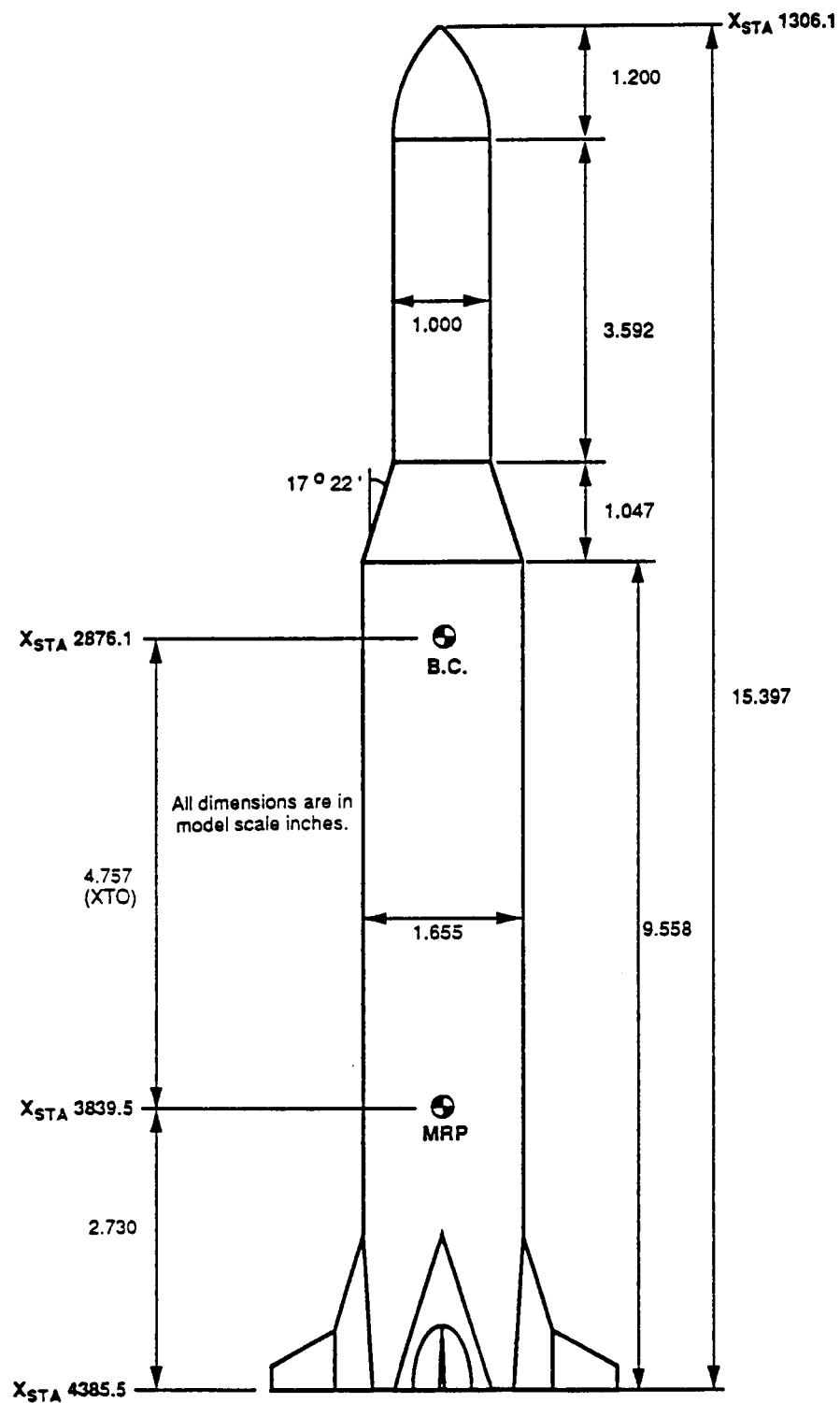
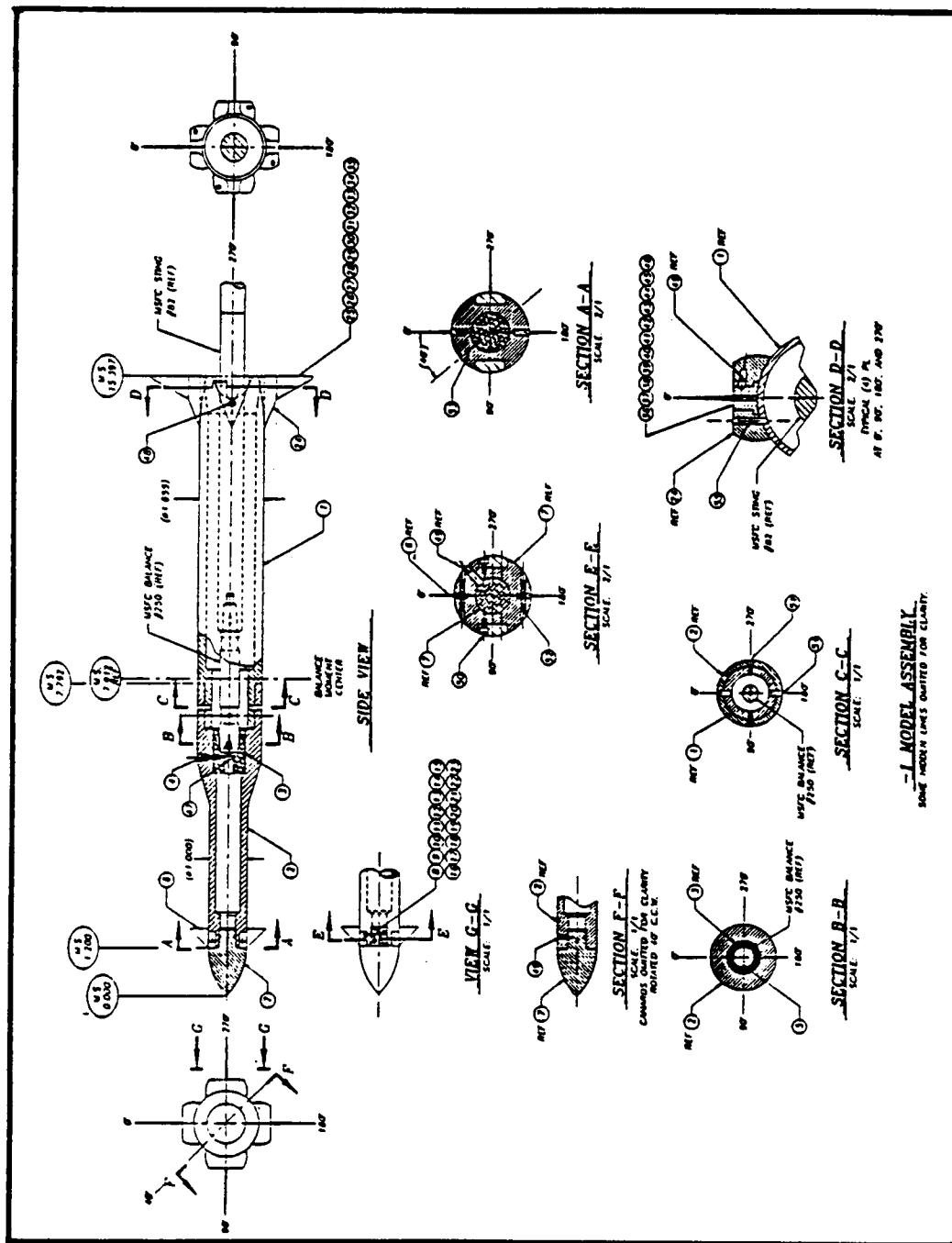


Figure 95. Test LV model (0.005-scale).



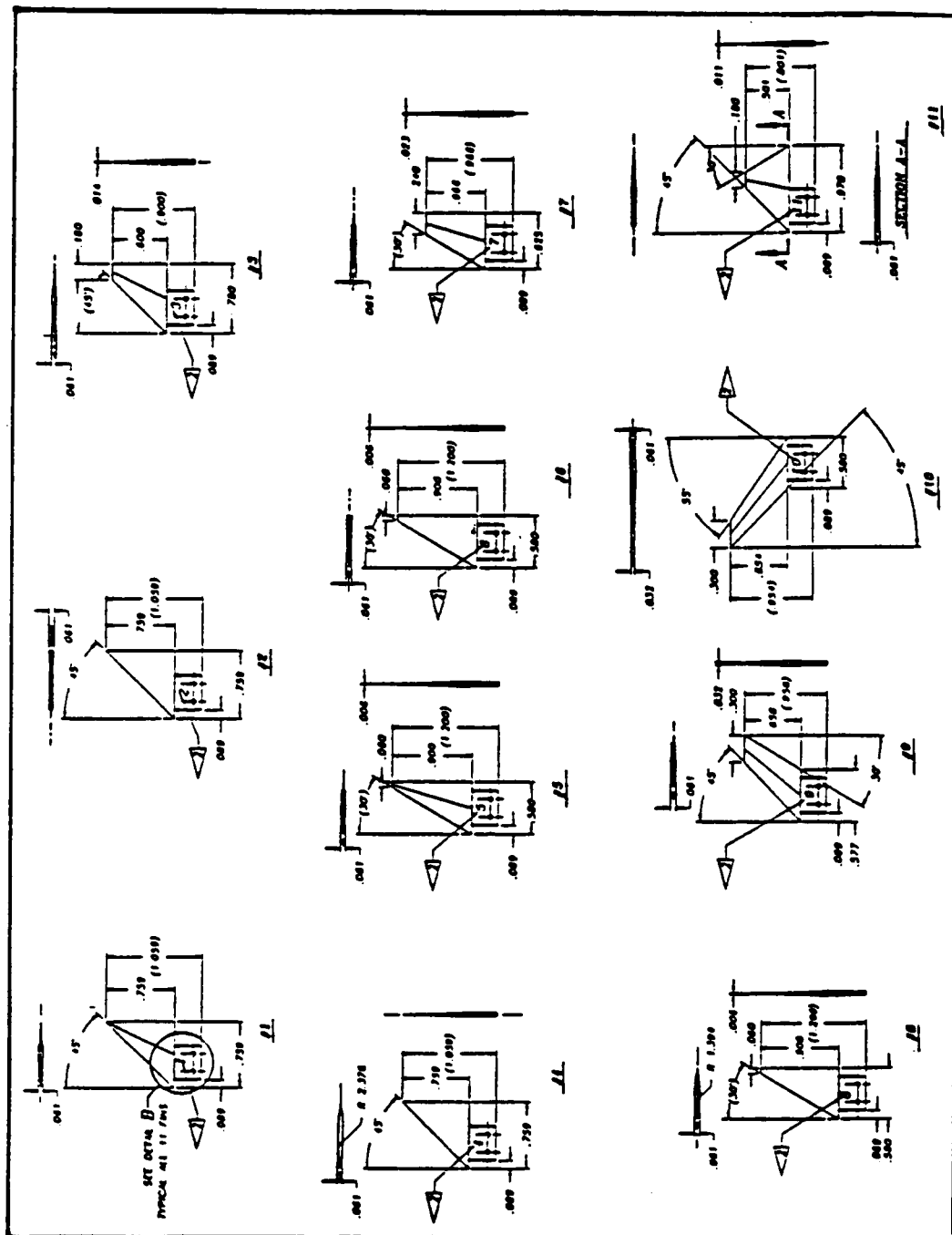




Figure 98. Photograph of fabricated test LV with aft FCA's.

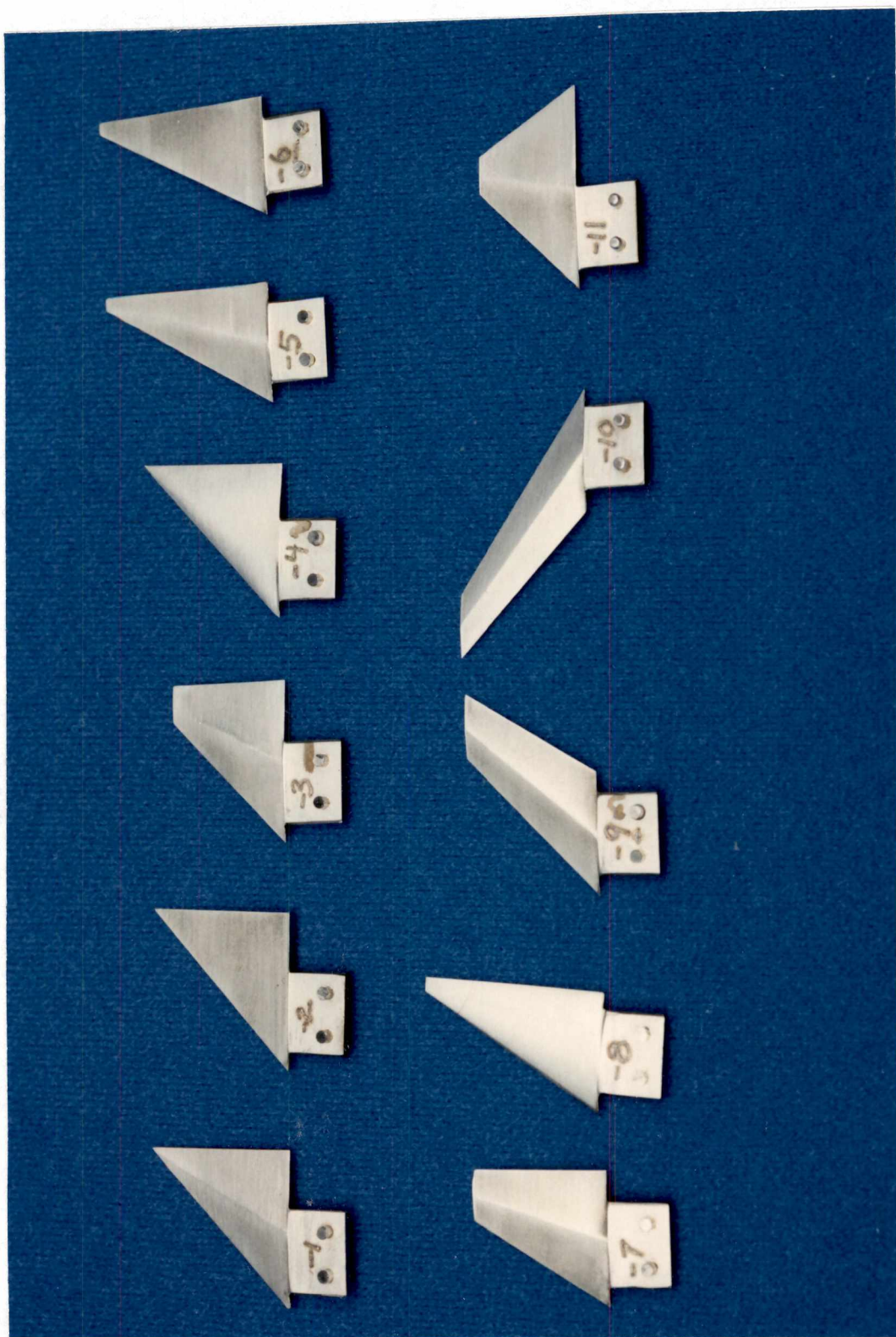


Figure 99. Photograph of fabricated 11 aft FCA designs tested.





Figure 100. Photograph of fabricated test LV with canards and aft FCA's.

## 6.5 Test Conditions

Table 5 shows the wind tunnel operating conditions during the static stability wind tunnel test program. The Reynolds numbers in the wind tunnel testing were much lower than those for the proposed LV in actual free flight. In order to expedite the testing, the  $\alpha$  and  $\beta$  ranges were subsequently reduced to avoid use of the sting offsets, the initial FCA deflection increments were increased, and low priority runs were deleted. The 12 configurations were tested over an  $\alpha$  and  $\beta$  range of  $\pm 10^\circ$ , the aft FCA's were tested to  $15^\circ$  deflection in increments of  $3^\circ$ , and the canards were tested to  $30^\circ$  deflection in increments of  $5^\circ$ .

Table 5. Wind tunnel operating conditions.

| Mach Number | Reynolds Number<br>( $\times 10^6/\text{ft}$ ) | Dynamic Pressure<br>( $\text{lb}/\text{in}^2$ ) | Stagnation Temperature<br>( $^\circ\text{F}$ ) | Stagnation Pressure<br>( $\text{lb}/\text{in}^2$ ) |
|-------------|------------------------------------------------|-------------------------------------------------|------------------------------------------------|----------------------------------------------------|
| 0.20        | 1.6                                            | 0.58                                            | 100                                            | 22                                                 |
| 0.40        | 3.4                                            | 2.21                                            | 100                                            | 22                                                 |
| 0.80        | 5.5                                            | 6.47                                            | 100                                            | 22                                                 |
| 0.95        | 6.2                                            | 7.76                                            | 100                                            | 22                                                 |
| 1.20        | 6.5                                            | 9.15                                            | 100                                            | 22                                                 |
| 1.46        | 6.0                                            | 9.49                                            | 100                                            | 22                                                 |
| 1.96        | 7.2                                            | 11.00                                           | 100                                            | 30                                                 |
| 2.74        | 4.7                                            | 6.38                                            | 140                                            | 30                                                 |
| 3.48        | 4.8                                            | 5.15                                            | 140                                            | 45                                                 |
| 4.96        | 4.4                                            | 2.73                                            | 140                                            | 80                                                 |

As can be seen in Table 4, for the lower Mach number of 0.2 and 0.4, the tolerances were large. MSFC wind tunnel tests are not normally conducted at Mach number below 0.6. The author requested the low speed tests in an attempt to determine

when the FCA's became effective. That is, at what speed was there enough dynamic pressure on the surfaces to provide useful control augmentation for the LV. As mentioned in the last chapter, the Space Shuttle flight controls are effective at  $M = 0.2$ . It was determined that the loads measured on the model at those Mach numbers below  $M = 0.6$  were very small. The #250 balance (normally used for  $M = 0.6$  and above) was not capable of measuring such small loads with the accuracy desired. Consequently, the data at the Mach numbers of 0.2 and 0.4 has only be used to evaluate trends and not to establish actual values.



## CHAPTER VII. STATIC STABILITY AND CONTROL ANALYSES

### 7.1 Data Reduction and Linearization

In order to perform the LV ascent static stability and control analysis, considerable time was spent on wind tunnel test data reduction and data linearization in order to obtain the stability derivatives, the rates of change of the three forces and the three moments with respect to  $\alpha$  and  $\beta$ ,

$$\frac{\partial C_N}{\partial \alpha}, \quad \frac{\partial C_m}{\partial \alpha}, \quad \frac{\partial C_A}{\partial \alpha}, \quad \frac{\partial C_Y}{\partial \beta}, \quad \frac{\partial C_n}{\partial \beta}, \quad \frac{\partial C_\ell}{\partial \beta}$$

and the control derivatives, the rates of change of the forces and the moments with respect to the FCA deflections,

$$\frac{\partial C_N}{\partial \delta_F}, \quad \frac{\partial C_m}{\partial \delta_F}, \quad \frac{\partial C_A}{\partial \delta_F}, \quad \frac{\partial C_Y}{\partial \delta_F}, \quad \frac{\partial C_n}{\partial \delta_F}, \quad \frac{\partial C_\ell}{\partial \delta_F}, \quad \text{and} \quad X_{cp}$$

the cp locations. The pertinent raw wind tunnel test data from which to begin consist of the three force coefficients and the three moment coefficients.

It should be mentioned here that, as a routine practice at MSFC, wind tunnel data is studied, cleaned up, linearized, and tabulated for the user. The data is sent out, as mentioned previously, to other MSFC Divisions, other NASA Centers, industry, and academia in a linearized slope-of-coefficient format so that any aerodynamic angles can be readily used. Since this research was a special project with the author as Principal Investigator, the author elected to work with, study, and linearize the raw data as part of the research work, including the low speed data that was not used in this analysis.

A selected sample of raw wind tunnel test data is included in Appendix C.1 to show the format of the information presented. Table C1 shows the raw longitudinal data

for FCA #3 at  $\delta_F = 0^\circ$  and  $M = 0.2$ . For the longitudinal data, sweeps were made in  $\alpha$  and only the longitudinal coefficients are valid. For the lateral data, sweeps were made in  $\beta$  and only the lateral coefficients are valid. Tables C2 (positive sweep) and C3 (negative sweep) show the raw wind tunnel data for FCA #4 at  $\delta_F = 15^\circ$  and  $M = 4.96$ . Due to the shorter run times at supersonic speeds, the  $\alpha$  and  $\beta$  sweeps had to be broken into two parts. One run covered the sweep ideally from  $-10^\circ$  to  $0^\circ$ , and the other from  $0^\circ$  to  $10^\circ$  in  $2^\circ$  increments.

In the author's data study, three minor but time consuming problems were examined and dealt with in the data reduction. (1) The ideal  $2^\circ$   $\alpha$  and  $\beta$  increments from  $-10^\circ$  to  $10^\circ$ , (i.e.,  $-10^\circ, -8^\circ, -6^\circ, \dots, 6^\circ, 8^\circ, 10^\circ$ ) were not realized. The actual increments were not uniform or regular as can be seen in Tables C1 to C3. This is common in real world data acquisition due to deflection of the sting and balance with loads on the model. For example, values of coefficients were recorded at aerodynamic angles such as  $-10.18^\circ, -9.92^\circ, -7.84^\circ, -5.88^\circ, 9.22^\circ, 9.35^\circ$ , etc. (2) In the supersonic range, when two runs were required to complete an  $\alpha$  or  $\beta$  sweep, the stopping and starting points of the sweeps near  $0^\circ$  did not match up exactly as can be seen in Tables C2 and C3. The data then suggested a discontinuity around the origin. (3) In calculating  $X_{cp}$  it is desirable to have  $C_m/C_N$  rather than the quotient of their gradients. However, the quotient of  $C_m/C_N$  was not uniformly realistic because either or both of these coefficients sometimes had very small negative values near  $\alpha = 0^\circ$  when they should have actually been positive, or vice versa. These small discrepancies could have been due to tunnel turbulence and/or a hydraulic pitch sector transient vibration. At other times, very small magnitudes of these coefficients resulted in a quotient that was zero, or tended to infinity. For example:  $C_N = 0$  has been recorded; at  $M = 2.74$ , with no FCA's, at  $\alpha = +0.03^\circ$ ,  $C_N = -0.0129$  and  $C_m = -0.0353$ ; at  $M = 1.2$ , with FCA #1, at  $\alpha = +0.05^\circ$ ,  $C_N = +0.0087$ , and  $C_m = -0.0047$ .

To overcome these problems, the raw data was linearized. To begin, plots were made of coefficients vs. their respective aerodynamic angles, e.g.,  $C_N$  and  $C_m$  vs.  $\alpha$

for the LV with no FCA's at each of the 10 Mach numbers, and then for the LV with each of the 12 FCA designs at  $\delta_F = 0^\circ$  at fixed Mach numbers. The slopes of these plots were calculated and tabulated. The reduction and linearization of only selected longitudinal stability and control derivatives is presented here; the others are done likewise. The method used was to first plot  $C_N$  vs.  $\alpha$  and  $C_m$  vs.  $\alpha$  for each Mach number for the case of no FCA's, where it is known that the data must pass through the origin. Then the slopes through the origin of each plot were calculated, to be used in the  $C_{m_\alpha}$  and  $C_{N_\alpha}$  tabulations. Next, the same was done with each FCA at  $0^\circ$  deflection, i.e., the  $C_N$  vs.  $\alpha$  and  $C_m$  vs.  $\alpha$  were plotted to determine the slopes in the case of each FCA. These plots served as worksheets from which to extract slopes and extent of linear range, keeping in mind that the maximum  $\alpha$  for the vehicle was  $4^\circ$ . (See Figures 101 to 109.) The next round of plots considered the FCA's at each deflection up to  $15^\circ$  for aft FCA's and up to  $30^\circ$  for canards, and each Mach number run. These plots did not pass through the origin as was to be expected. Results of the data reduction and linearization for longitudinal stability parameters are tabulated in Table 6.

The data reduction for the control derivatives consisted of plotting all coefficients vs.  $\alpha$  and  $\beta$  in the cases of FCA deflections, then tabulating values of the coefficients at integral values of  $\alpha$  and  $\beta$ . Next, at fixed integer values of  $\alpha$  and  $\beta$  and fixed Mach numbers, plots were made of the coefficients vs. FCA deflection angles, and the data was linearized by taking the slopes. The method for determining the control derivatives is described using  $C_{m\delta_F}$  as example. First  $C_m$  vs.  $\alpha$  was plotted for a specific FCA and Mach number at all deflection angles. Figures 110 and 111 show  $C_m$  and  $C_N$  for FCA 1 at  $M = 1.2$  and  $\delta_F = 0^\circ$  to  $15^\circ$ . The tolerances on the aerodynamic coefficients are listed and it can be seen that for small  $\alpha$  the linearization is valid. Then the  $C_m$  values were obtained at integral values of  $\alpha$ . Next,

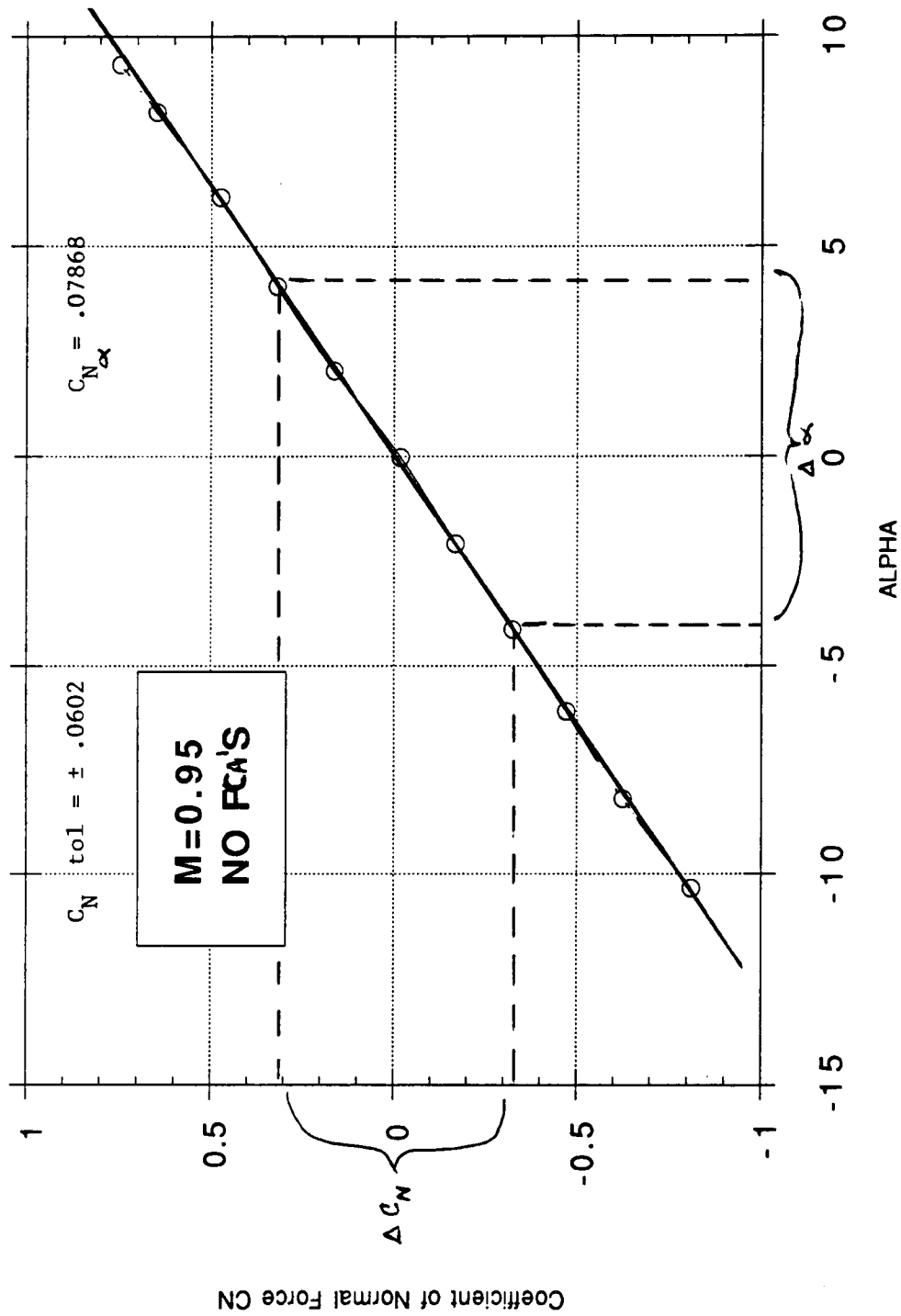


Figure 101.  $C_N$  vs.  $\alpha$  worksheet; no FCA's;  $M=0.95$ .

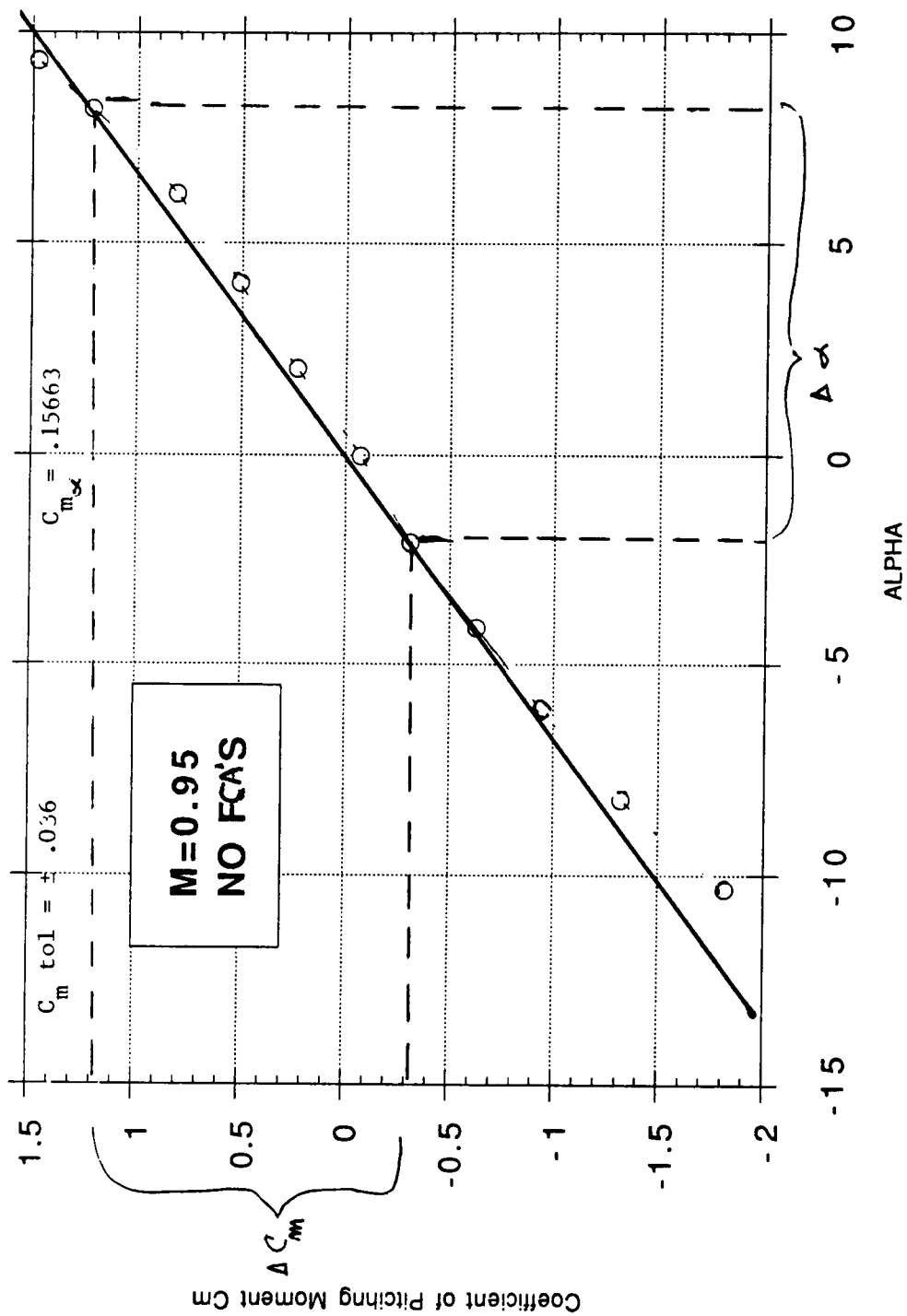


Figure 102.  $C_m$  vs.  $\alpha$  worksheet; no FCA's;  $M=0.95$ .

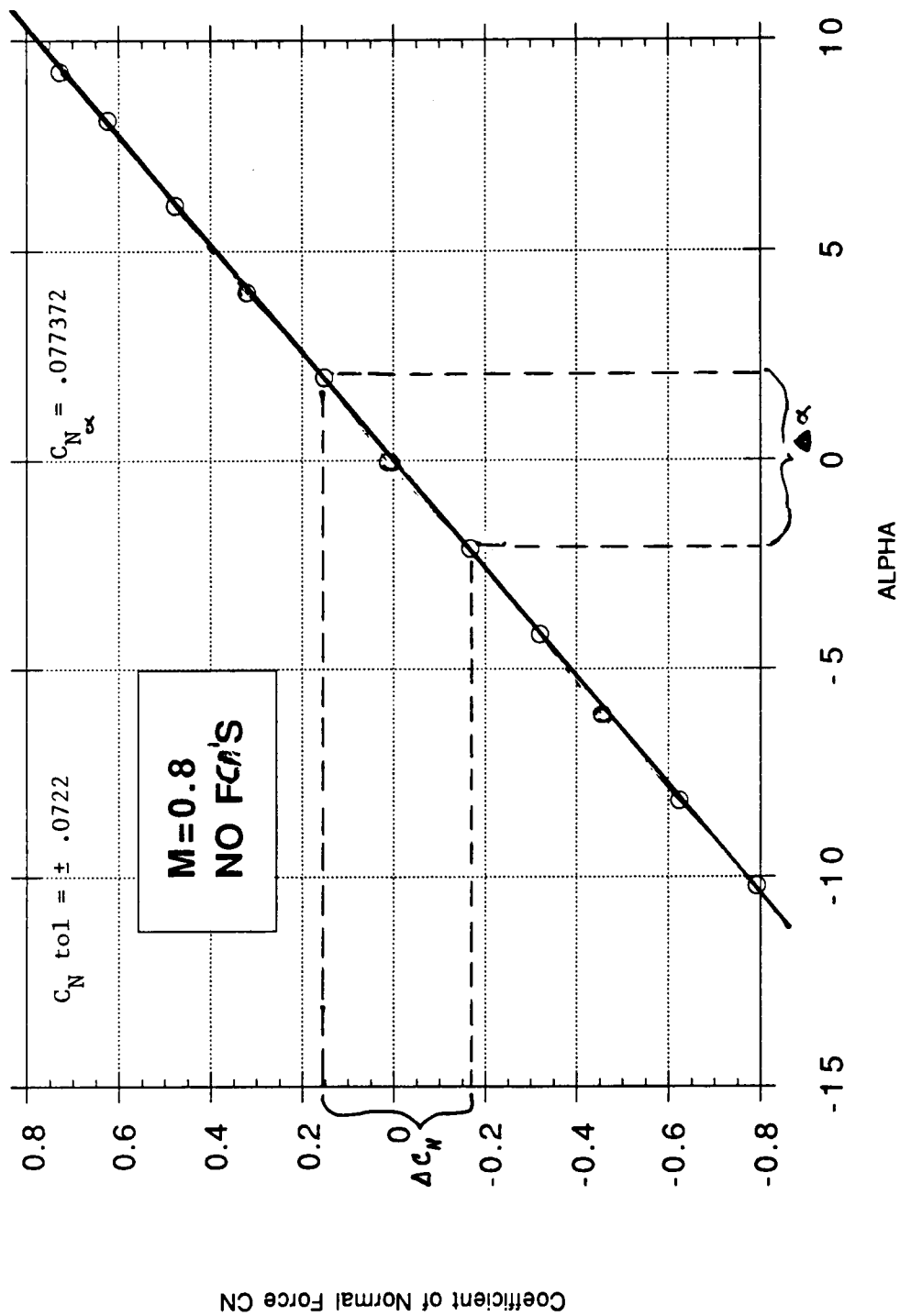


Figure 103.  $C_N$  vs.  $\alpha$  worksheet; no FCA's;  $M=0.8$ .

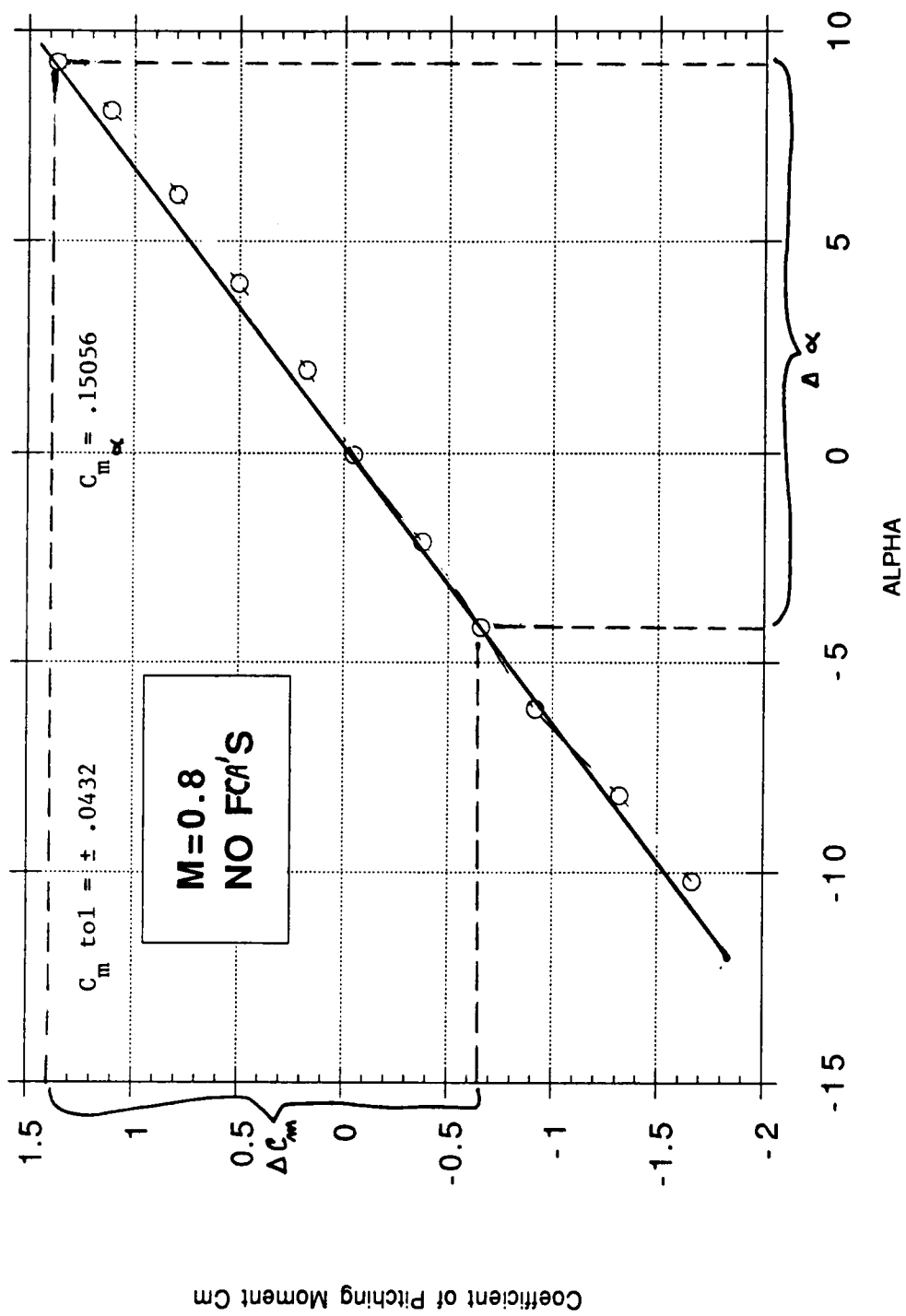


Figure 104.  $C_m$  vs.  $\alpha$  worksheet; no FCA's;  $M = 0.8$ .

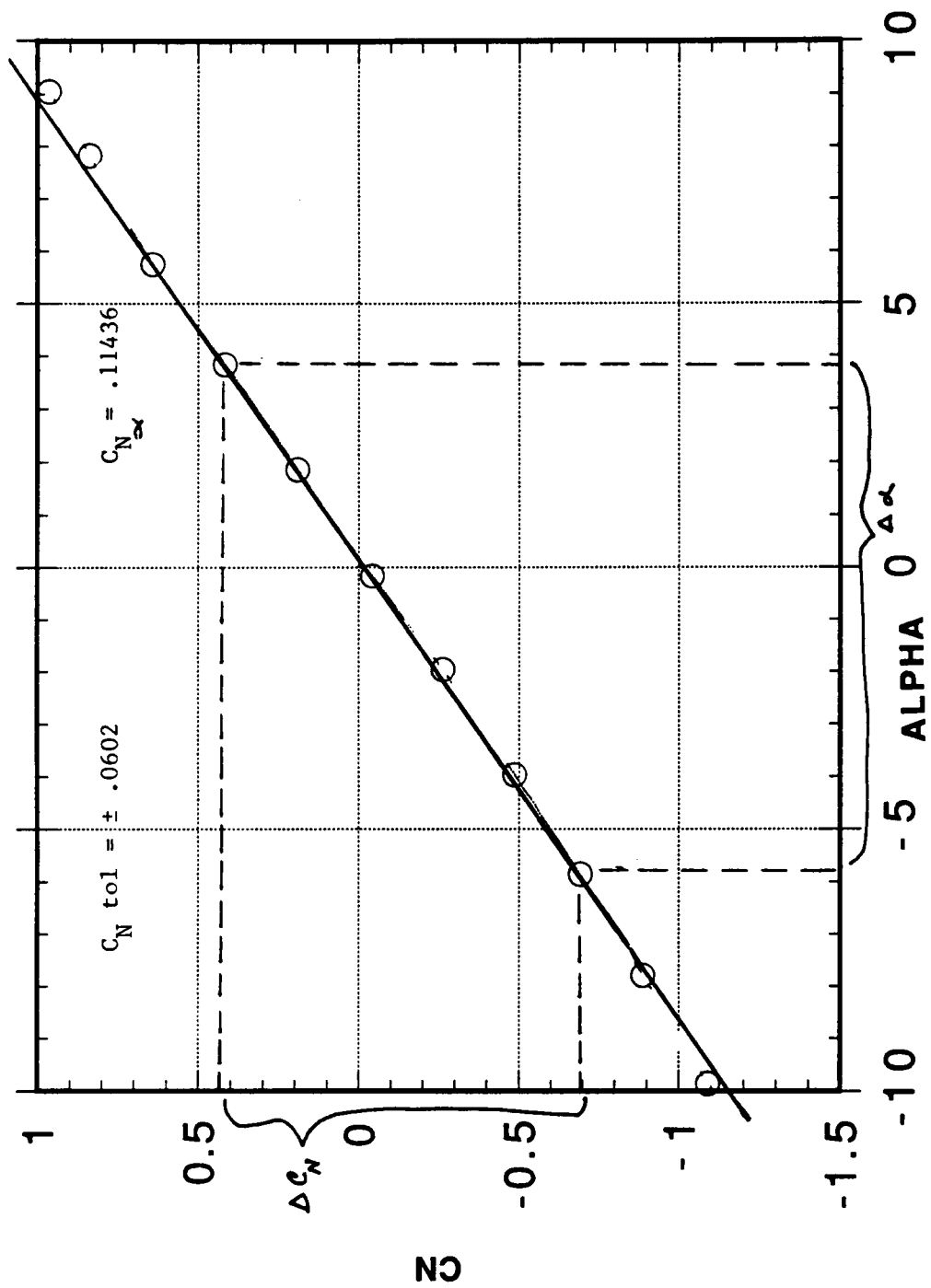


Figure 105.  $C_N$  vs.  $\alpha$  worksheet; FCA 5;  $M = 0.95$ .



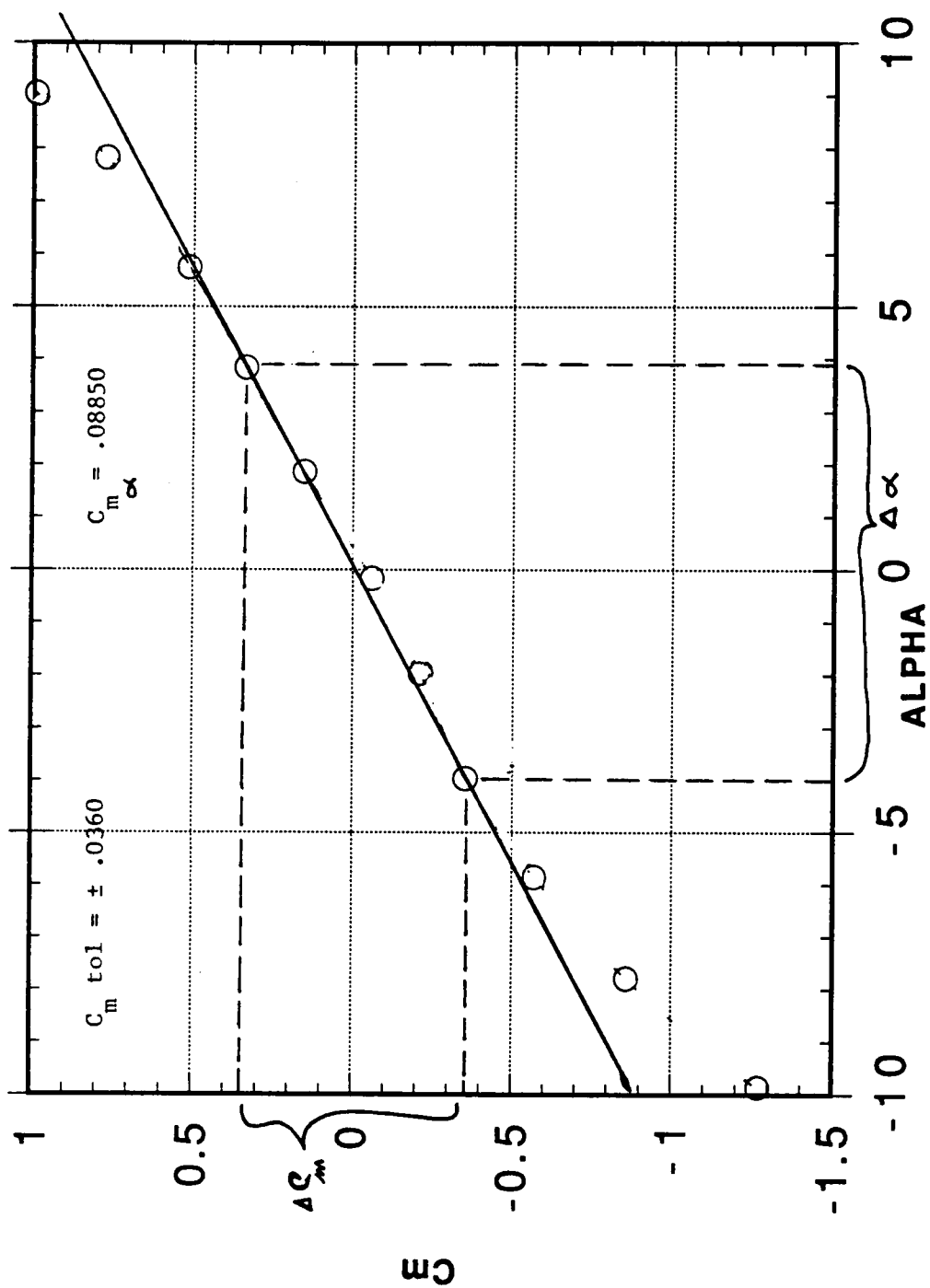


Figure 106.  $C_m$  vs.  $\alpha$  worksheet; FCA 5;  $M = 0.95$ .

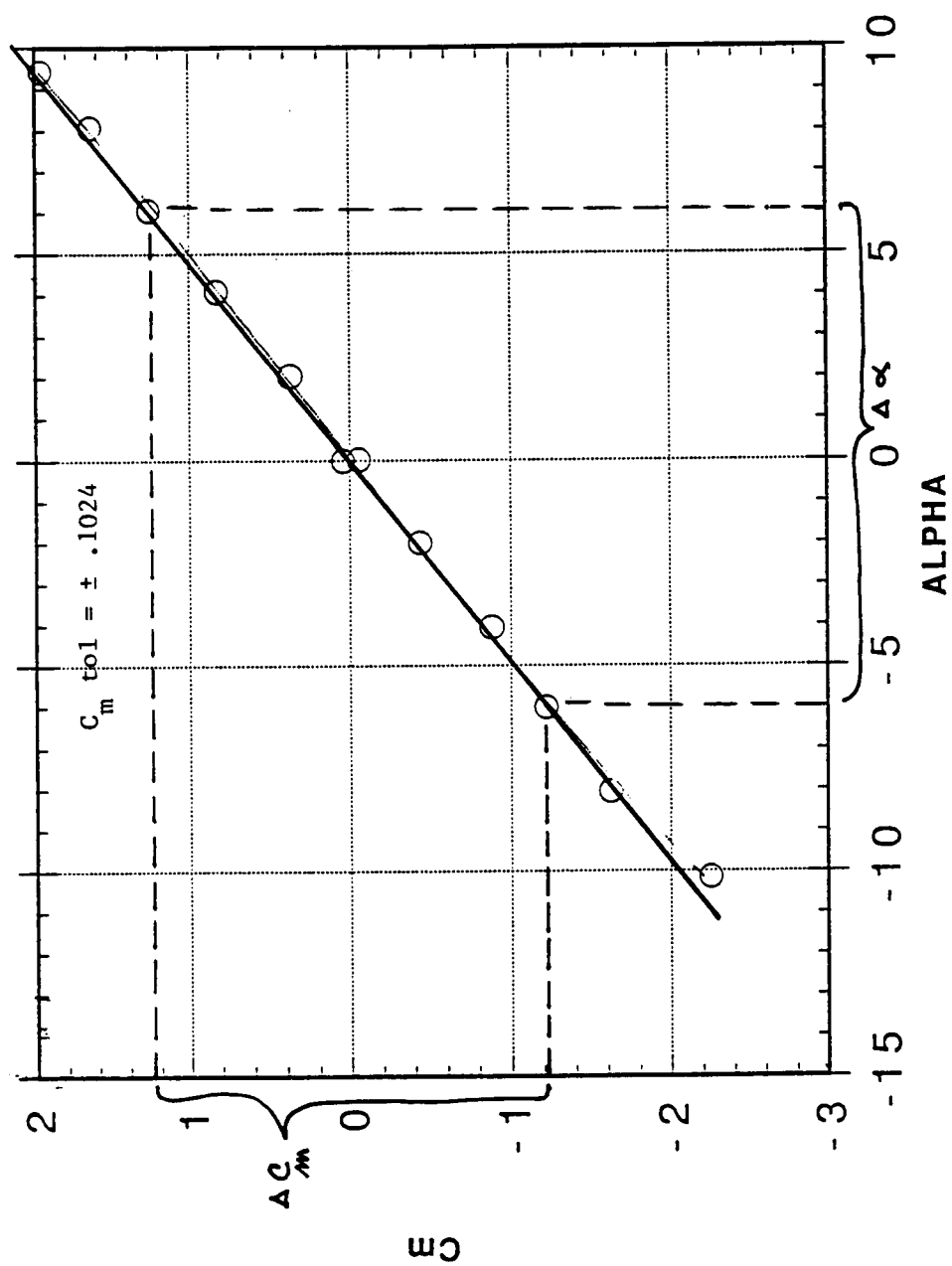


Figure 107.  $C_m$  vs.  $\alpha$  worksheet; FCA 4;  $M = 4.96$ .

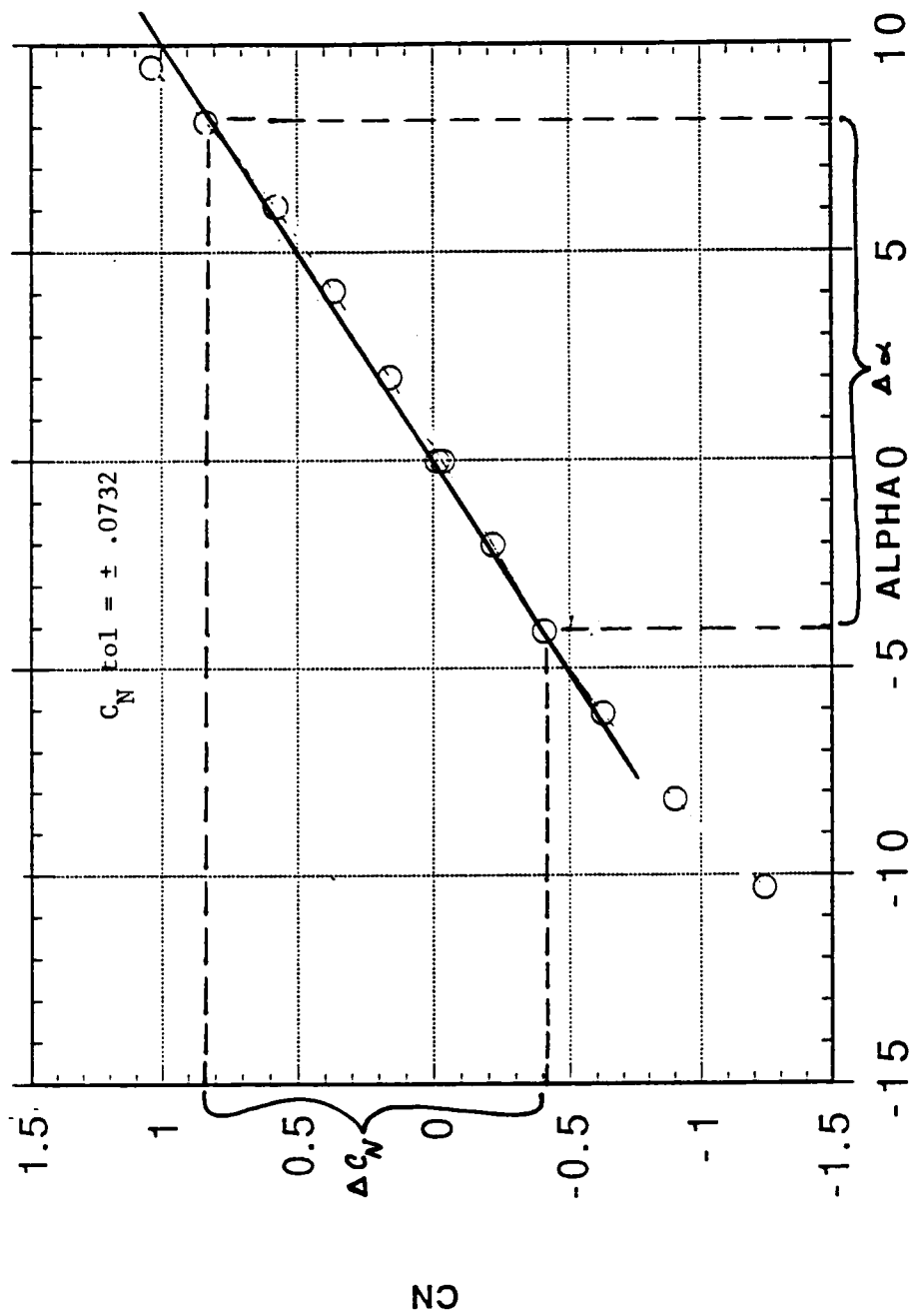


Figure 108.  $C_N$  vs.  $\alpha$  worksheet; FCA 7;  $M = 2.74$ .

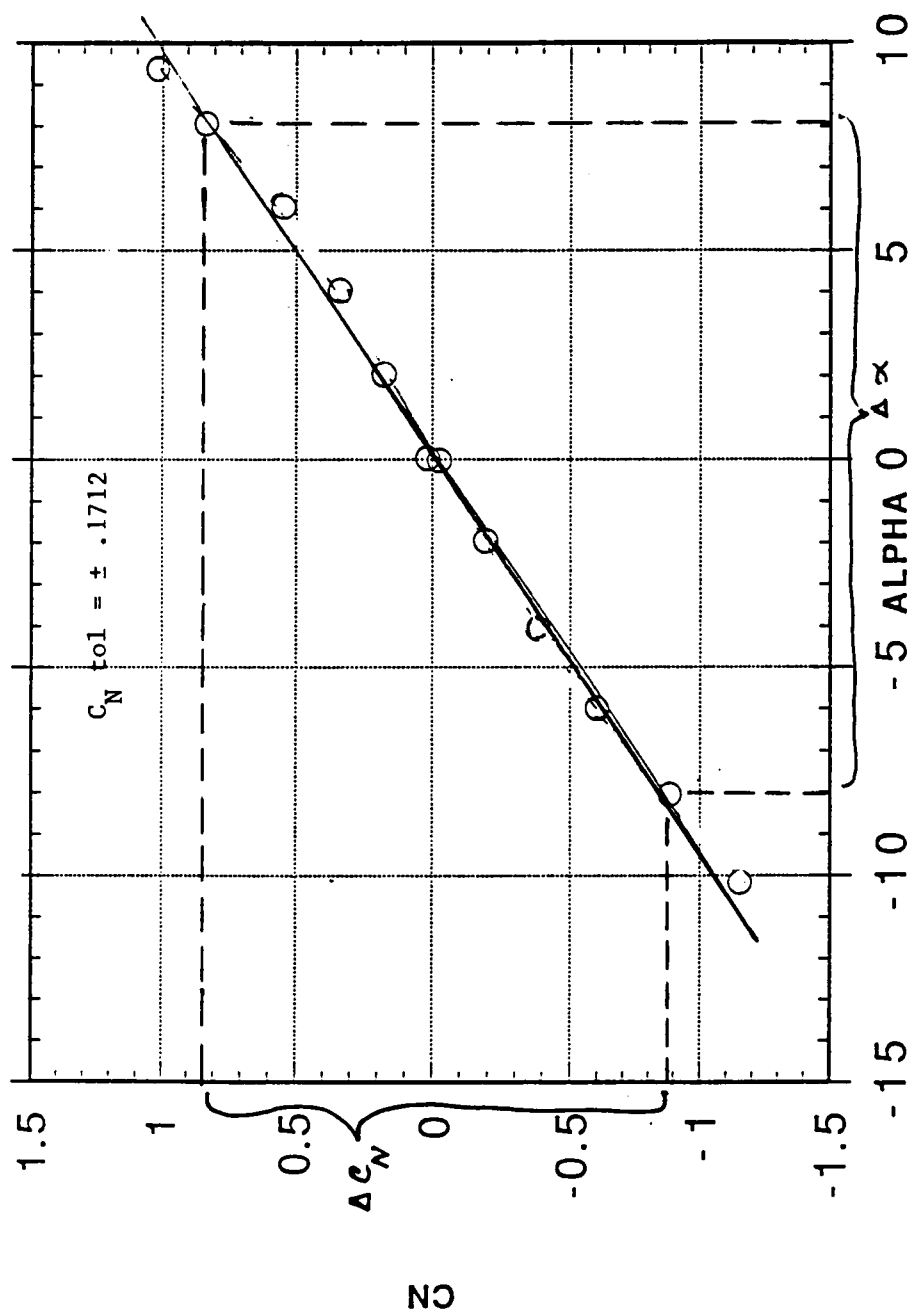


Figure 109.  $C_N$  vs.  $\alpha$  worksheet; FCA 7;  $M = 4.96$ .

Table 6. Calculated longitudinal static stability parameters.

| $C_{N_\alpha}$            | $C_{m_\alpha}$ | M    | SM       | Xcp From Aft<br>End |
|---------------------------|----------------|------|----------|---------------------|
| NO FCA's                  |                |      |          |                     |
| 0.077372                  | 0.15056        | 0.80 | -1.9459  | 3.594               |
| 0.07868                   | 0.15663        | 0.95 | -1.9907  | 3.639               |
| 0.082689                  | 0.17106        | 1.20 | -2.0687  | 3.717               |
| 0.079187                  | 0.18489        | 1.46 | -2.3348  | 3.983               |
| 0.075104                  | 0.19899        | 1.96 | -2.6495  | 4.298               |
| 0.083023                  | 0.21004        | 2.74 | -2.5299  | 4.178               |
| 0.082002                  | 0.21560        | 3.48 | -2.6292  | 4.277               |
| 0.091236                  | 0.20955        | 4.96 | -2.2968  | 3.945               |
| FCA #1 $\delta = 0^\circ$ |                |      |          |                     |
| 0.099690                  | 0.099983       | 0.80 | -1.0029  | 2.6509              |
| 0.10402                   | 0.095502       | 0.95 | -0.91811 | 2.5661              |
| 0.109740                  | 0.13015        | 1.20 | -1.186   | 2.834               |
| 0.101380                  | 0.15351        | 1.46 | -1.5142  | 3.162               |
| 0.088601                  | 0.17795        | 1.96 | -2.0084  | 3.656               |
| 0.09280                   | 0.18635        | 2.74 | -2.0081  | 3.656               |
| 0.087482                  | 0.19801        | 3.48 | -2.2634  | 3.911               |
| 0.094358                  | 0.19948        | 4.96 | -2.1141  | 3.762               |
| FCA #2 $\delta = 0^\circ$ |                |      |          |                     |
| 0.10528                   | 0.093632       | 0.80 | -0.88936 | 2.5374              |
| 0.109966                  | 0.10344        | 0.95 | -0.94063 | 2.5886              |
| 0.11264                   | 0.12034        | 1.20 | -1.06839 | 2.7164              |
| 0.10326                   | 0.16128        | 1.46 | -1.5619  | 3.2099              |
| 0.10222                   | 0.18884        | 1.96 | -1.8474  | 3.4954              |
| 0.101698                  | 0.20211        | 2.74 | -1.98735 | 3.6354              |
| 0.09794                   | 0.211667       | 3.48 | -2.1612  | 3.8092              |
| 0.10490                   | 0.19731        | 4.96 | -1.88096 | 3.529               |
| FCA #3 $\delta = 0^\circ$ |                |      |          |                     |
| 0.09918                   | 0.11368        | 0.80 | -1.1462  | 2.7942              |
| 0.103336                  | 0.117918       | 0.95 | -1.1411  | 2.7891              |
| 0.10647                   | 0.13574        | 1.20 | -1.2749  | 2.9229              |
| 0.10118                   | 0.15105        | 1.46 | -1.49288 | 3.1409              |
| 0.09359                   | 0.18060        | 1.96 | -1.9297  | 3.5777              |
| 0.09357                   | 0.19095        | 2.74 | -2.0407  | 3.6887              |
| 0.08836                   | 0.191127       | 3.48 | -2.16305 | 3.8111              |
| 0.099694                  | 0.20558        | 4.96 | -2.0630  | 3.7110              |
| FCA #4 $\delta = 0^\circ$ |                |      |          |                     |
| 0.099630                  | 0.11869        | 0.80 | -1.19130 | 2.839               |
| 0.10218                   | 0.12028        | 0.95 | -1.1771  | 2.825               |
| 0.10566                   | 0.15581        | 1.20 | -1.4747  | 3.123               |
| 0.098674                  | 0.16090        | 1.46 | -1.63066 | 3.279               |
| 0.096698                  | 0.17906        | 1.96 | -1.85175 | 3.5                 |
| 0.099144                  | 0.19011        | 2.74 | -1.91752 | 3.566               |
| 0.08991                   | 0.18692        | 3.48 | -2.0789  | 3.737               |
| 0.10592                   | 0.20566        | 4.96 | -1.94164 | 3.59                |

Table 6. Continued.

| $C_{N_\alpha}$            | $C_{m_\alpha}$ | M    | SM        | Xcp From Aft<br>End |
|---------------------------|----------------|------|-----------|---------------------|
| FCA #5 $\delta = 0^\circ$ |                |      |           |                     |
| 0.10869                   | 0.09955        | 0.80 | -0.91591  | 2.564               |
| 0.11436                   | 0.08850        | 0.95 | -0.7739   | 2.422               |
| 0.11396                   | 0.124375       | 1.20 | -1.09139  | 2.739               |
| 0.10438                   | 0.14710        | 1.46 | -1.40931  | 3.057               |
| 0.09609                   | 0.174545       | 1.96 | -1.81647  | 3.465               |
| 0.097064                  | 0.18572        | 2.74 | -1.91335  | 3.561               |
| 0.098096                  | 0.19644        | 3.48 | -2.0025   | 3.650               |
| 0.10609                   | 0.208325       | 4.96 | -1.9628   | 3.611               |
| FCA #6 $\delta = 0^\circ$ |                |      |           |                     |
| 0.11206                   | 0.081324       | 0.80 | -0.72572  | 2.374               |
| 0.12435                   | 0.07026        | 0.95 | -0.56506  | 2.213               |
| 0.12118                   | 0.11205        | 1.20 | -0.92465  | 2.573               |
| 0.10962                   | 0.13284        | 1.46 | -1.21178  | 2.86                |
| 0.10592                   | 0.16970        | 1.96 | -1.6022   | 3.250               |
| 0.099992                  | 0.18741        | 2.74 | -1.87423  | 3.522               |
| 0.098173                  | 0.19814        | 3.48 | -2.0182   | 3.666               |
| 0.15946                   | 0.20144        | 4.96 | -1.2632   | 2.911               |
| FCA #7 $\delta = 0^\circ$ |                |      |           |                     |
| 0.10027                   | 0.10192        | 0.80 | -1.01644  | 2.664               |
| 0.11251                   | 0.09041        | 0.95 | -0.803565 | 2.452               |
| 0.1108                    | 0.1276         | 1.20 | -1.1516   | 2.8                 |
| 0.10269                   | 0.15396        | 1.46 | -1.4993   | 3.147               |
| 0.09727                   | 0.17779        | 1.96 | -1.8278   | 3.476               |
| 0.10129                   | 0.20624        | 2.74 | -2.03615  | 3.684               |
| 0.09797                   | 0.20079        | 3.48 | -2.0495   | 3.698               |
| 0.10639                   | 0.18804        | 4.96 | -1.7675   | 3.415               |
| FCA #8 $\delta = 0^\circ$ |                |      |           |                     |
| 0.11928                   | 0.09747        | 0.80 | -0.81716  | 2.465               |
| 0.011862                  | 0.088773       | 0.95 | -0.74838  | 2.396               |
| 0.11282                   | 0.1258         | 1.20 | -1.1150   | 2.763               |
| 0.10202                   | 0.14912        | 1.46 | -1.46166  | 3.11                |
| 0.096823                  | 0.17788        | 1.96 | -1.8372   | 3.485               |
| 0.10069                   | 0.18865        | 2.74 | -1.8736   | 3.522               |
| 0.10248                   | 0.18655        | 3.48 | -1.82037  | 3.468               |
| 0.095482                  | 0.19311        | 4.96 | -2.0225   | 3.670               |
| FCA #9 $\delta = 0^\circ$ |                |      |           |                     |
| 0.10258                   | 0.10786        | 0.80 | -1.05144  | 2.699               |
| 0.10519                   | 0.10775        | 0.95 | -1.0243   | 2.672               |
| 0.10683                   | 0.13062        | 1.20 | -1.22267  | 2.871               |
| 0.10173                   | 0.15293        | 1.46 | -1.50325  | 3.151               |
| 0.091627                  | 0.17191        | 1.96 | -1.8762   | 3.524               |
| 0.095177                  | 0.195          | 2.74 | -2.0488   | 3.697               |

Table 22. Continued.

| $C_{N\alpha}$                                            | $C_{m\alpha}$ | M    | SM       | Xcp From Aft<br>End |
|----------------------------------------------------------|---------------|------|----------|---------------------|
| FCA #10 $\delta = 0^\circ$                               |               |      |          |                     |
| 0.10697                                                  | 0.11204       | 0.80 | -1.0474  | 2.695               |
| 0.11179                                                  | 0.106996      | 0.95 | -0.95712 | 2.605               |
| 0.11542                                                  | 0.13621       | 1.20 | -1.18016 | 2.828               |
| 0.10578                                                  | 0.15649       | 1.46 | -1.4794  | 3.127               |
| 0.099324                                                 | 0.17661       | 1.96 | -1.7781  | 3.426               |
| FCA #11 $\delta = 0^\circ$                               |               |      |          |                     |
| 0.09642                                                  | 0.11011       | 0.80 | -1.14199 | 2.79                |
| 0.10037                                                  | 0.10796       | 0.95 | -1.07567 | 2.724               |
| 0.10532                                                  | 0.13284       | 1.20 | -1.2613  | 2.909               |
| 0.10032                                                  | 0.15679       | 1.46 | -1.56288 | 3.211               |
| 0.094556                                                 | 0.1939        | 1.96 | -2.0506  | 3.699               |
| Canards $\delta = 0^\circ$ and FCA #3 $\delta = 0^\circ$ |               |      |          |                     |
| 0.10882                                                  | 0.21824       | 0.80 | -2.0055  | 3.654               |
| 0.11812                                                  | 0.25393       | 0.95 | -2.1497  | 3.798               |
| 0.11847                                                  | 0.23176       | 1.20 | -1.9563  | 3.604               |
| 0.10774                                                  | 0.22121       | 1.46 | -2.0615  | 3.71                |
| 0.09913                                                  | 0.22947       | 1.96 | -2.3138  | 3.963               |
| Canards $\delta = 5^\circ$ and FCA #3 $\delta = 0^\circ$ |               |      |          |                     |
| 0.10867                                                  | 0.2084        | 0.80 | -1.91778 | 3.566               |
| 0.11483                                                  | 0.23408       | 0.95 | -2.0385  | 3.686               |
| 0.11704                                                  | 0.23310       | 1.20 | -1.9916  | 3.64                |
| 0.10771                                                  | 0.23343       | 1.46 | -2.1672  | 3.815               |
| 0.09986                                                  | 0.2368        | 1.96 | -2.3714  | 4.019               |
| FCA #1 $\delta = 6^\circ$                                |               |      |          |                     |
| 0.10099                                                  | 0.11307       | 0.80 | -1.1196  | 2.7676              |
| 0.10236                                                  | 0.08738       | 0.95 | -0.85366 | 2.5016              |
| 0.10794                                                  | 0.13345       | 1.20 | -1.2363  | 2.8843              |
| 0.10003                                                  | 0.15802       | 1.46 | -1.5797  | 3.2277              |
| 0.090973                                                 | 0.17131       | 1.96 | -1.8831  | 3.5311              |
| 0.097038                                                 | 0.18363       | 2.74 | -1.8924  | 3.5404              |
| 0.94048                                                  | 0.19661       | 3.48 | -2.0905  | 3.7385              |
| 0.10412                                                  | 0.20299       | 4.96 | -1.9496  | 3.5976              |
| FCA #1 $\delta = 15^\circ$                               |               |      |          |                     |
| 0.096275                                                 | 0.108412      | 0.80 | -1.1261  | 2.7741              |
| 0.099769                                                 | 0.10369       | 0.95 | -1.0393  | 2.6873              |
| 0.10339                                                  | 0.13809       | 1.20 | -1.3356  | 2.9836              |
| 0.098243                                                 | 0.16253       | 1.46 | -1.6544  | 3.3024              |
| 0.089579                                                 | 0.1737        | 1.96 | -1.9391  | 3.5871              |
| 0.096615                                                 | 0.18538       | 2.74 | -1.9187  | 3.5667              |
| 0.091714                                                 | 0.19231       | 3.48 | -2.0968  | 3.7449              |
| 0.10186                                                  | 0.19365       | 4.96 | -1.9012  | 3.5492              |

Table 6. Continued.

| $C_{N_\alpha}$             | $C_{m_\alpha}$ | M    | SM       | Xcp From Aft<br>End |
|----------------------------|----------------|------|----------|---------------------|
| FCA #2 $\delta = 6^\circ$  |                |      |          |                     |
| 0.10216                    | 0.096275       | 0.80 | -0.9424  | 2.5904              |
| 0.10741                    | 0.094269       | 0.95 | -0.87766 | 2.5257              |
| 0.11117                    | 0.1255         | 1.20 | -1.1289  | 2.7769              |
| 0.10386                    | 0.14258        | 1.46 | -1.3728  | 3.0208              |
| 0.093458                   | 0.174088       | 1.96 | -1.8627  | 3.5107              |
| 0.099127                   | 0.18509        | 2.74 | -1.8672  | 3.5152              |
| 0.09352                    | 0.19297        | 3.48 | -2.0634  | 3.7114              |
| 0.097621                   | 0.18666        | 4.96 | -1.9120  | 3.5600              |
| FCA #2 $\delta = 15^\circ$ |                |      |          |                     |
| 0.10012                    | 0.10672        | 0.80 | -1.0659  | 2.7139              |
| 0.10151                    | 0.12265        | 0.95 | -1.2082  | 2.8562              |
| 0.10601                    | 0.13334        | 1.20 | -1.2578  | 2.9058              |
| 0.09895                    | 0.15012        | 1.46 | -1.5172  | 3.1652              |
| 0.091537                   | 0.17006        | 1.96 | -1.8578  | 3.5058              |
| 0.09782                    | 0.19334        | 2.74 | -1.9765  | 3.6245              |
| 0.09425                    | 0.19569        | 3.48 | -2.0762  | 3.7242              |
| 0.098561                   | 0.18803        | 4.96 | -1.9078  | 3.556               |
| FCA #3 $\delta = 6^\circ$  |                |      |          |                     |
| 0.098515                   | 0.10682        | 0.80 | -1.08431 | 2.7323              |
| 0.10229                    | 0.10466        | 0.95 | -1.0232  | 2.6712              |
| 0.10534                    | 0.14234        | 1.20 | -1.3512  | 2.9992              |
| 0.09965                    | 0.15214        | 1.46 | -1.5268  | 3.1747              |
| 0.095042                   | 0.18345        | 1.96 | -1.9302  | 3.578               |
| 0.085091                   | 0.18459        | 2.74 | -2.1693  | 3.8173              |
| 0.09121                    | 0.20460        | 3.48 | -2.2432  | 3.8912              |
| 0.10423                    | 0.19119        | 4.96 | -1.8343  | 3.4823              |
| FCA #3 $\delta = 15^\circ$ |                |      |          |                     |
| 0.10114                    | 0.11424        | 0.80 | -1.1295  | 2.7775              |
| 0.10089                    | 0.12096        | 0.95 | -1.1989  | 2.8469              |
| 0.10398                    | 0.14853        | 1.20 | -1.4285  | 3.0765              |
| 0.098207                   | 0.16284        | 1.46 | -1.6581  | 3.3061              |
| 0.094495                   | 0.18464        | 1.96 | -1.9539  | 3.6019              |
| 0.095123                   | 0.1783         | 2.74 | -1.8744  | 3.5224              |
| 0.09369                    | 0.20649        | 3.48 | -2.204   | 3.852               |
| 0.09953                    | 0.19622        | 4.96 | -1.9714  | 3.6194              |
| FCA #4 $\delta = 6^\circ$  |                |      |          |                     |
| 0.099154                   | 0.11125        | 0.80 | -1.122   | 2.77                |
| 0.10021                    | 0.11121        | 0.95 | -1.1098  | 2.7578              |
| 0.10492                    | 0.14462        | 1.20 | -1.3784  | 3.0264              |
| 0.09875                    | 0.16031        | 1.46 | -1.6234  | 3.2714              |
| 0.093345                   | 0.1868         | 1.96 | -2.0012  | 3.649               |
| 0.097186                   | 0.1796         | 2.74 | -1.8482  | 3.496               |
| 0.09442                    | 0.20314        | 3.48 | -2.1514  | 3.799               |
| 0.10862                    | 0.20167        | 4.96 | -1.8567  | 3.505               |



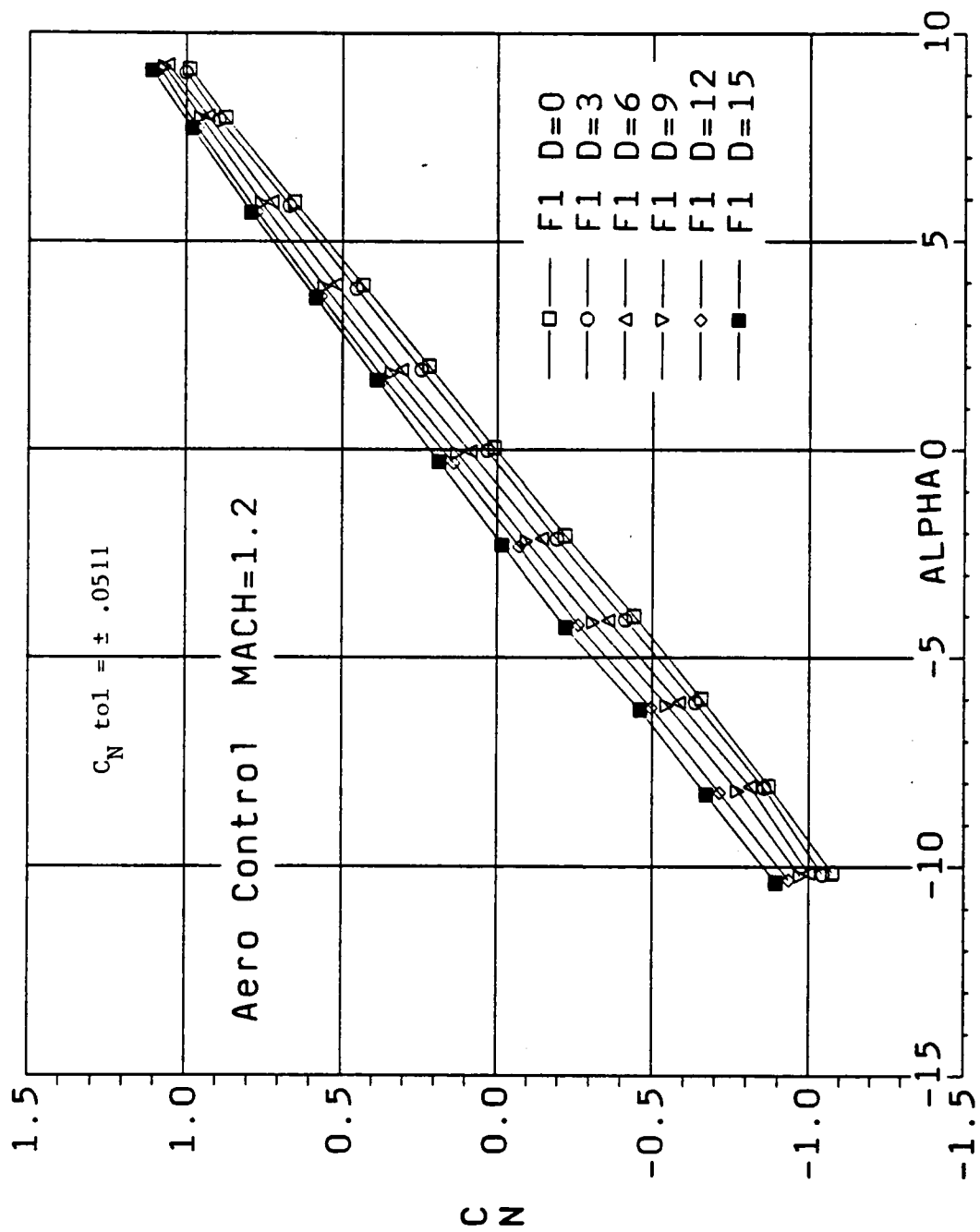


Figure 110.  $C_N$  vs.  $\alpha$  for FCA 1 ;  $\delta = 0^\circ$  to  $15^\circ$ .

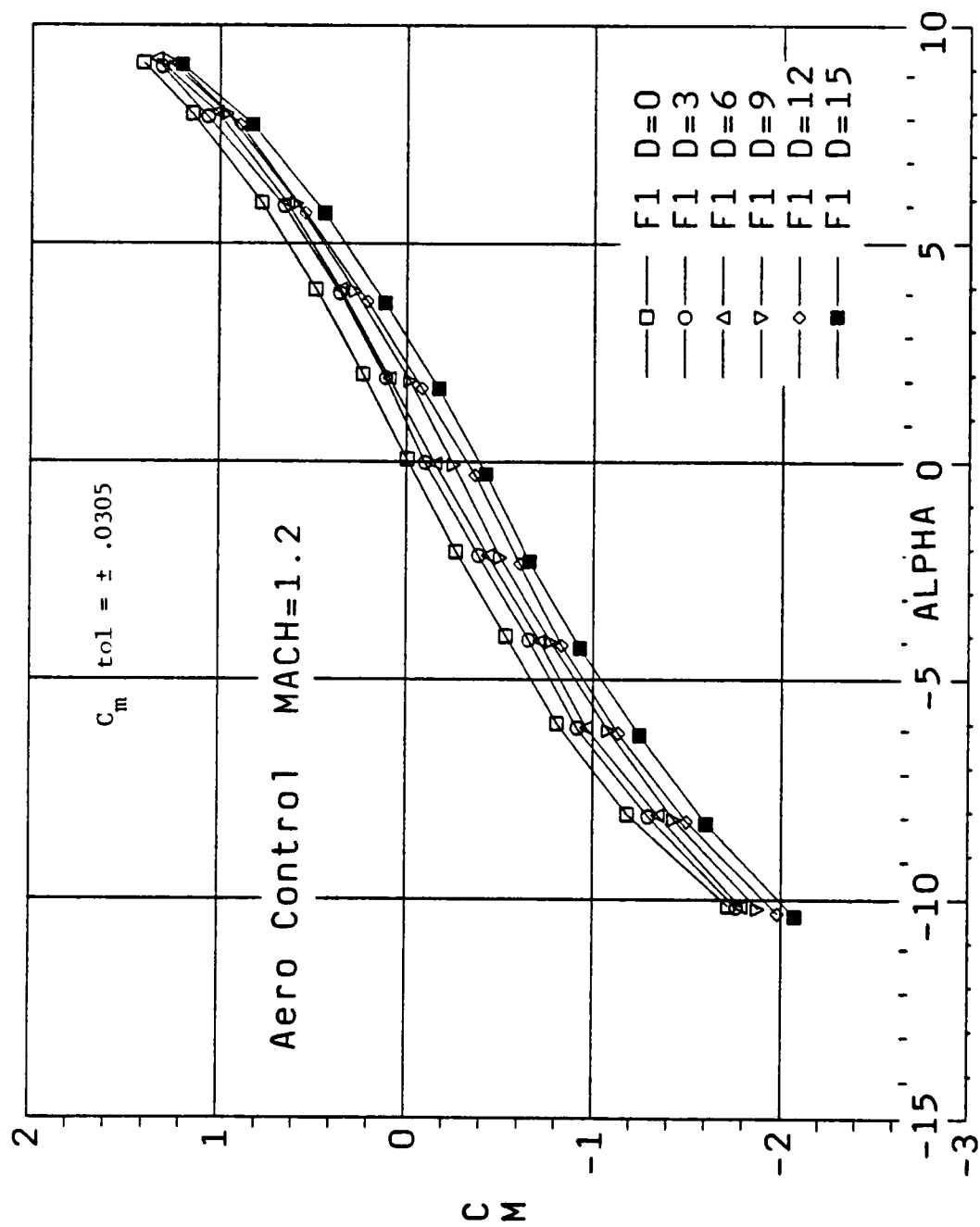


Figure 111.  $C_m$  vs.  $\alpha$  for FCA 1 ;  $\delta = 0^\circ$  to  $15^\circ$ .

these  $C_m$  values vs.  $\delta_F$  were plotted for specific FCA, Mach number, and  $\alpha$  values. The slopes were then taken to arrive at the linearized  $C_{m\delta_F}$  values. In summary, the basic stability derivatives were tabulated as slopes of coefficients vs. aerodynamic angle curves for fixed FCA deflections. The basic control derivatives were tabulated as slopes of coefficients vs. FCA deflection curves for fixed aerodynamic angles.

## 7.2 Stability Analysis

The criteria for longitudinal static stability of a flight vehicle are,

$$C_{m\alpha} < 0 \quad (28)$$

and for  $\alpha_{trim}$  to be in the flight range of the vehicle, where  $\alpha_{trim}$  is the  $\alpha$  where  $C_m = 0$ . A negative  $C_{m\alpha}$  means that as  $\alpha$  increases, e.g., from a disturbance, the pitching moment of the vehicle becomes more negative, thereby tending to decrease  $\alpha$  and stabilize the vehicle by returning it to its undisturbed position. The vehicle  $C_{m\alpha}$  varies with the cg position. As the cg of a stable vehicle moves farther aft,  $C_{m\alpha}$  becomes less and less negative. When the  $C_{m\alpha}$  has become positive, the vehicle becomes longitudinally unstable. That cg location at which  $C_{m\alpha} = 0$  is the Neutral Point (NP). The NP is the aerodynamic center (ac) for the entire flight vehicle, that place where the pitching moment is invariant with  $\alpha$ ,

$$cg \text{ at NP} \Rightarrow C_{m\alpha} = 0 \quad (29)$$

When the vehicle cg is located at the NP, the vehicle is said to have neutral stability. For a symmetric vehicle,

$$ac = cp \quad (30)$$

The longitudinal Static Margin (SM) is a quantitative measure of the amount of longitudinal static stability of a flight vehicle. The SM is defined as the distance forward of the NP that the cg is located. Figure 112 shows the desired positions of the NP and the corresponding amount of SM on the experimental LV for static stability. Of course, when the cg is located aft of the NP, the SM is negative and the vehicle is longitudinally unstable. Today, not all flight vehicles are statically stable. Sometimes more maneuverability can be achieved by flying statically unstable. For example, the X-29A aircraft is 35% unstable, where the SM in percentage is based on the MAC for aircraft and the caliber for LV's. The F-16 is unstable subsonically, neutrally stable transonically, and stable supersonically. Early LV's were statically stable. Today, most are unstable, notable exceptions having been mentioned in Chapters II to IV, and the control systems provide artificial stability. The Space Shuttle has a negative SM in certain regions of the entry flight envelope. As a LV's SM becomes more negative, the requirements for control torque increase.

Using a relationship (Etkin) that facilitates use of the calculated longitudinal static stability parameters in Table 6,

$$SM \equiv - \frac{C_{m\alpha}}{C_{N\alpha}} \quad (\text{in calibers}) \quad (31)$$

the SM's and locations of the vehicle cp ( $X_{cp}$ ) have been determined. For the experimental expendable LV in the ascent trajectory, the static flight stability has been assessed. Only the longitudinal static flight stability is presented, since it is much more germane to the aft cg problem being treated here than the static directional or lateral stability. In fact, for an axisymmetric flight vehicle,

$$C_{N\alpha} = C_{Y\beta} ; C_{m\alpha} = C_{n\beta} ; C_{N\delta} = C_{Y\delta} ; C_{m\delta} = C_{n\delta} \quad (32)$$

## Static Longitudinal Stability

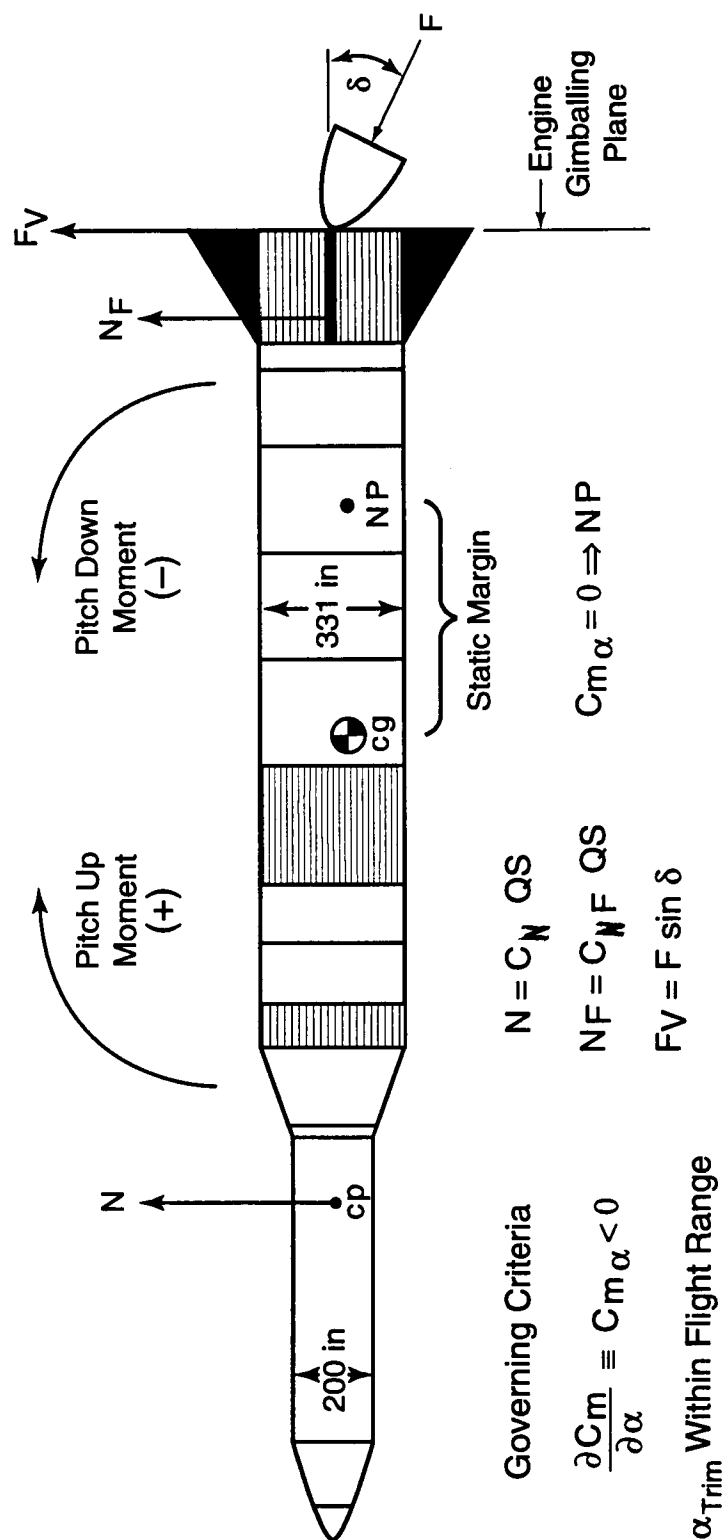


Figure 112. Static longitudinal stability of the experimental LV.

The SM's and the locations of the vehicle cp ( $X_{cp}$ ) have been determined first for the cases of no FCA's at each Mach number, and then for the cases with each of the different FCA's tested at each Mach number but with zero FCA deflection, and finally each FCA with deflection at each Mach number in order to assess the improvement in static flight stability and cp shift provided by each of the FCA designs during ascent.

Figure 113 shows the results for the basic experimental LV with no FCA's (where curves have been faired through the data and  $C_{NA} = C_{N\alpha}$ ;  $C_{mA} = C_{m\alpha}$ ). The cg is aft of the NP and the SM is negative, as expected, which indicates a longitudinally unstable vehicle.  $C_{N\alpha}$  hovers around 0.08 for the entire Mach range of usable Q, i.e., through  $M = 3.8$  as shown in section 5.3 above.  $C_{m\alpha}$  increases to 0.22 at  $M = 3.5$ . The best SM is approximately -2, and the best cp location is approximately 4 calibers from the aft end of the LV throughout the Mach range. Figures 114 through 124 show the results for each of the 11 aft FCA's tested at  $\delta_F = 0^\circ$ . It can be seen that in all cases the use of aft FCA's increases the total normal force slope on the LV and also moves the location of the cp aft. Figure 114, with the use of a FCA #1, shows the overall improved  $C_{N\alpha}$  which hovers around 0.1, has a significant rise during the max Q event, and then drops to approximately 0.09 for flight after  $M = 2$ . It can be seen that the overall SM has dramatically improved with the best value of -0.92. The cp has shifted farther aft overall and is approximately 3 calibers from the aft end. Figure 115 shows an increase of  $C_{N\alpha}$  to 0.113 during max Q and approximately 0.1 thereafter.  $C_{m\alpha}$  ranges from 0.094 to 0.212. The best SM is -0.89 and the best cp location has shifted to 2.53 calibers. Figure 116 shows a  $C_{N\alpha}$  range from 0.105 to 0.09 and a  $C_{m\alpha}$  range from 0.114 to 0.206. The best SM is -1.14 and the best cp shift is to 2.8 calibers. Figure 117 shows  $C_{N\alpha}$  hovers around 0.1;  $C_{m\alpha}$  ranges from 0.12 to 0.2. The best SM is -1.2, and the best cp shift is to 2.83 calibers. Figure 118 shows the  $C_{N\alpha}$  increases to 0.11,  $C_{m\alpha}$  decreases to 0.09, and the SM increases to = -0.77. Figure 119

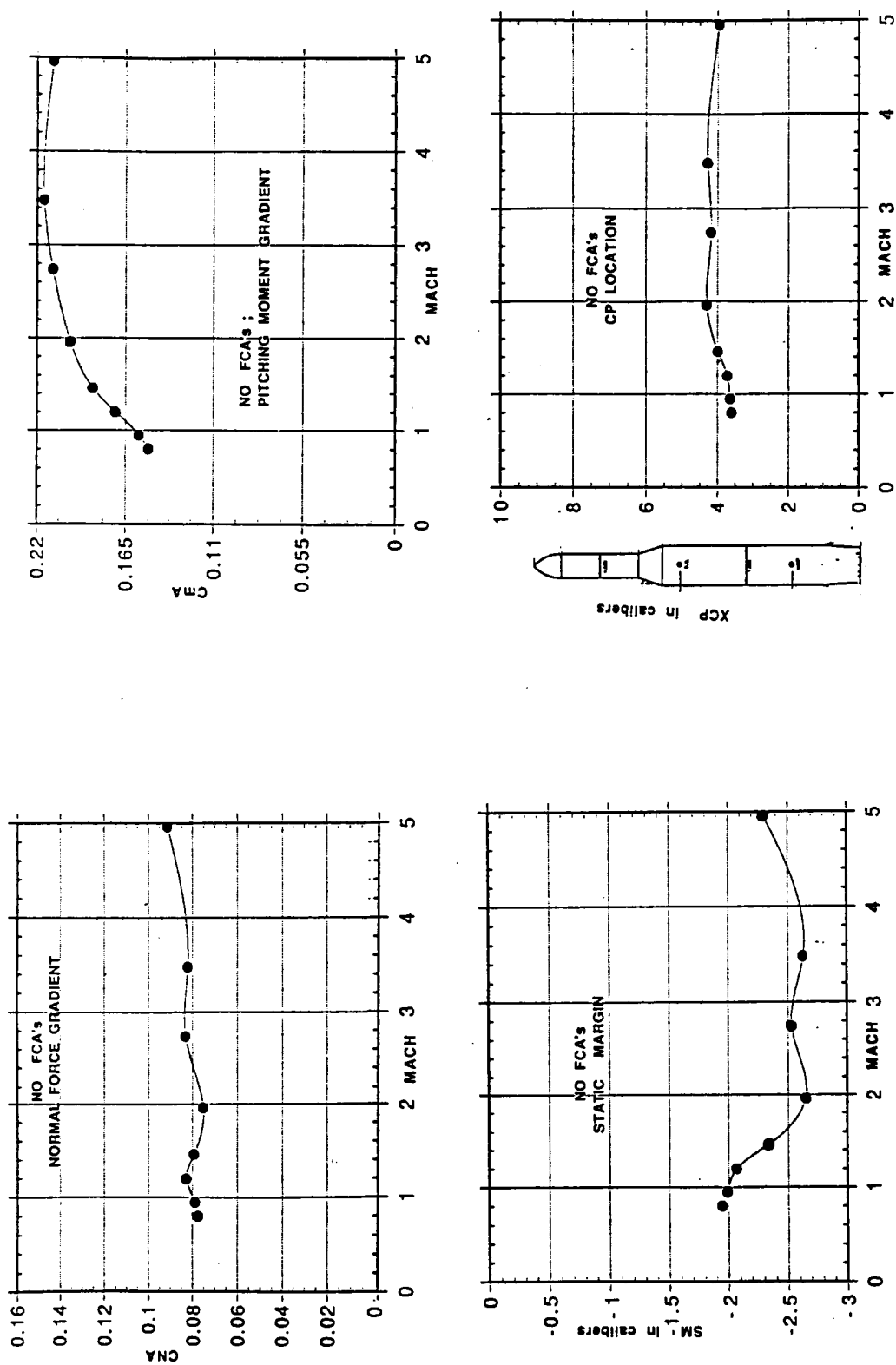


Figure 113. Stability results with no FCA's.

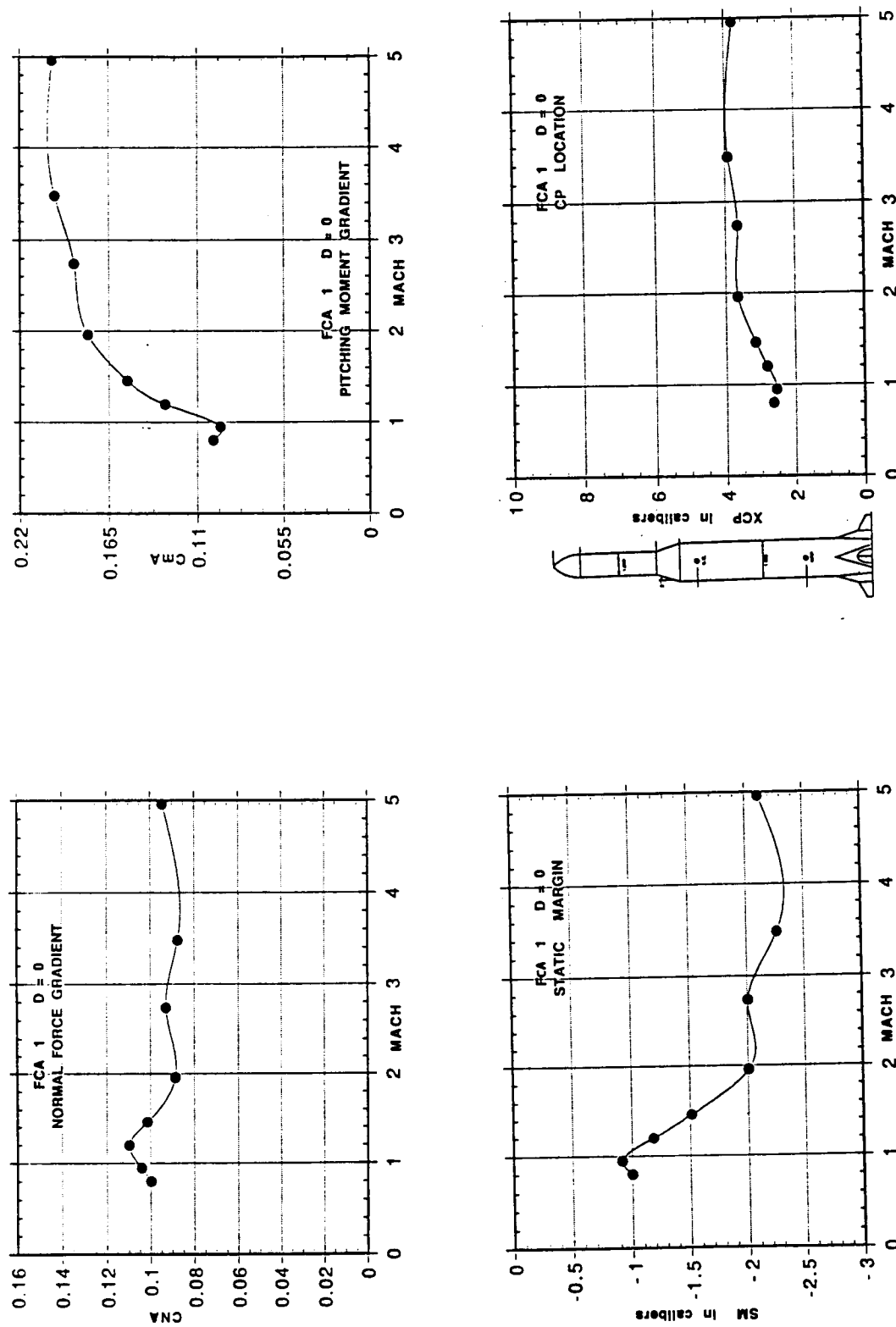


Figure 114. Stability results with FCA 1; deflection  $= 0^\circ$ .



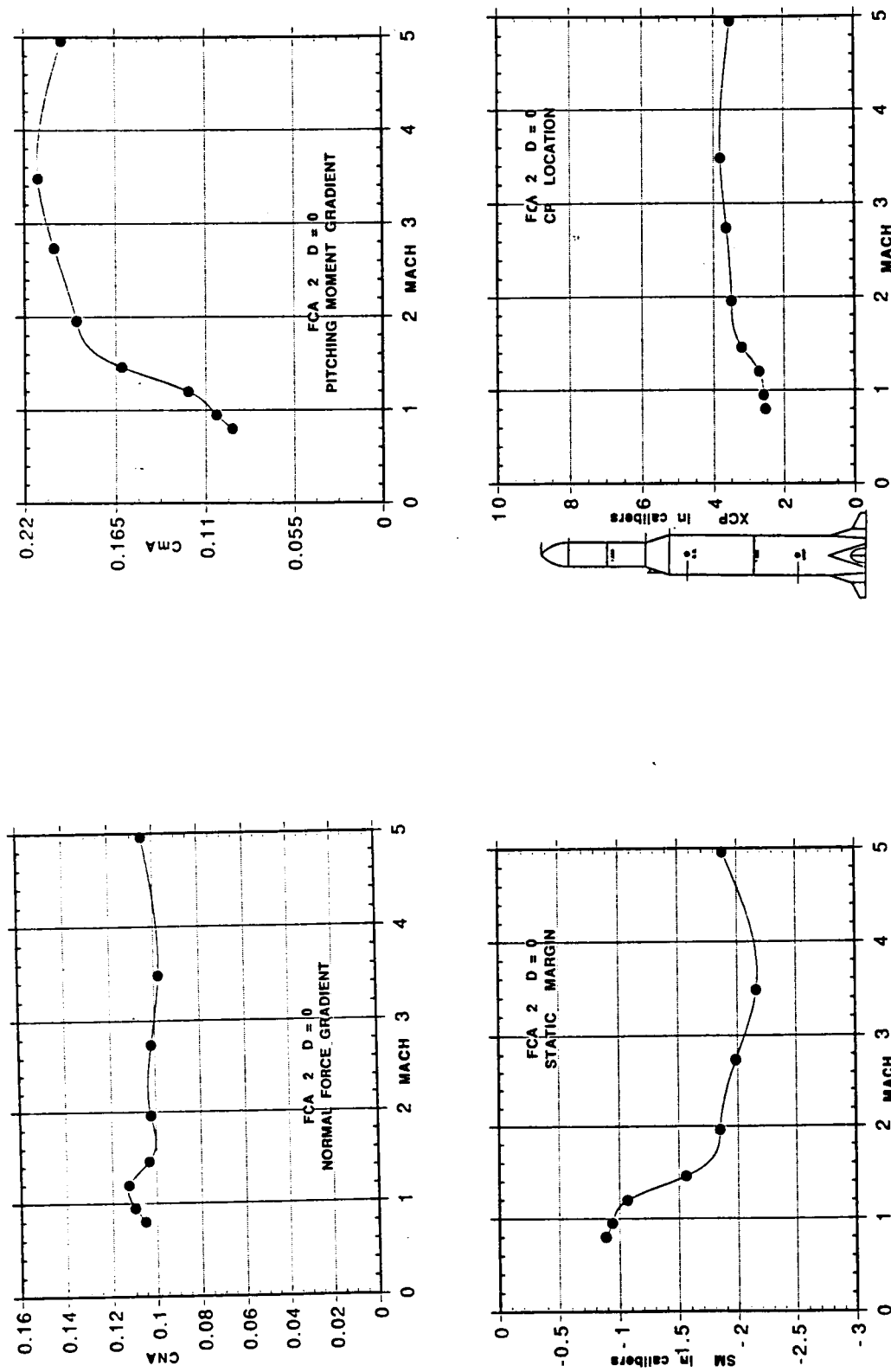


Figure 115. Stability results with FCA 2; deflection = 0°.

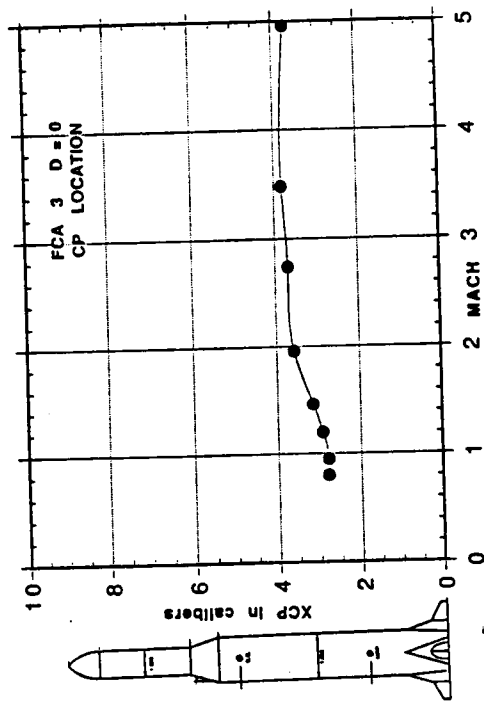
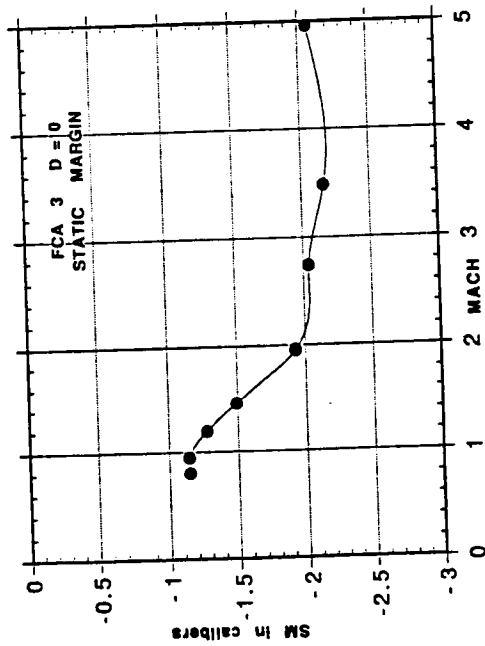
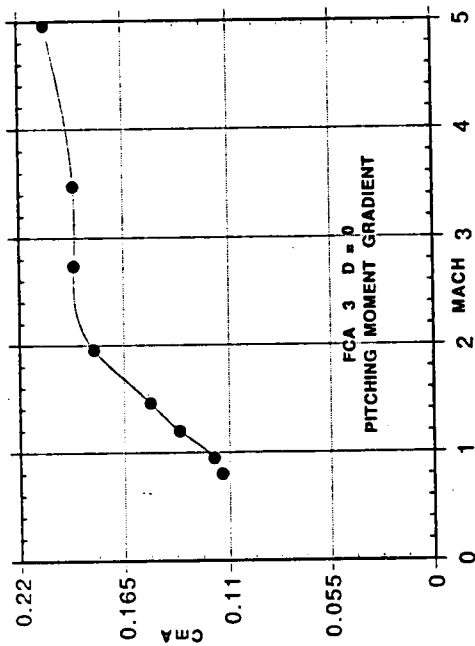
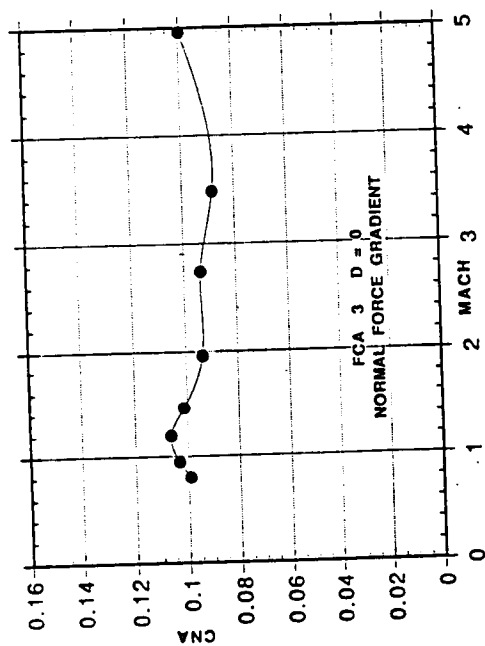


Figure 116. Stability results with FCA 3; deflection = 0°.

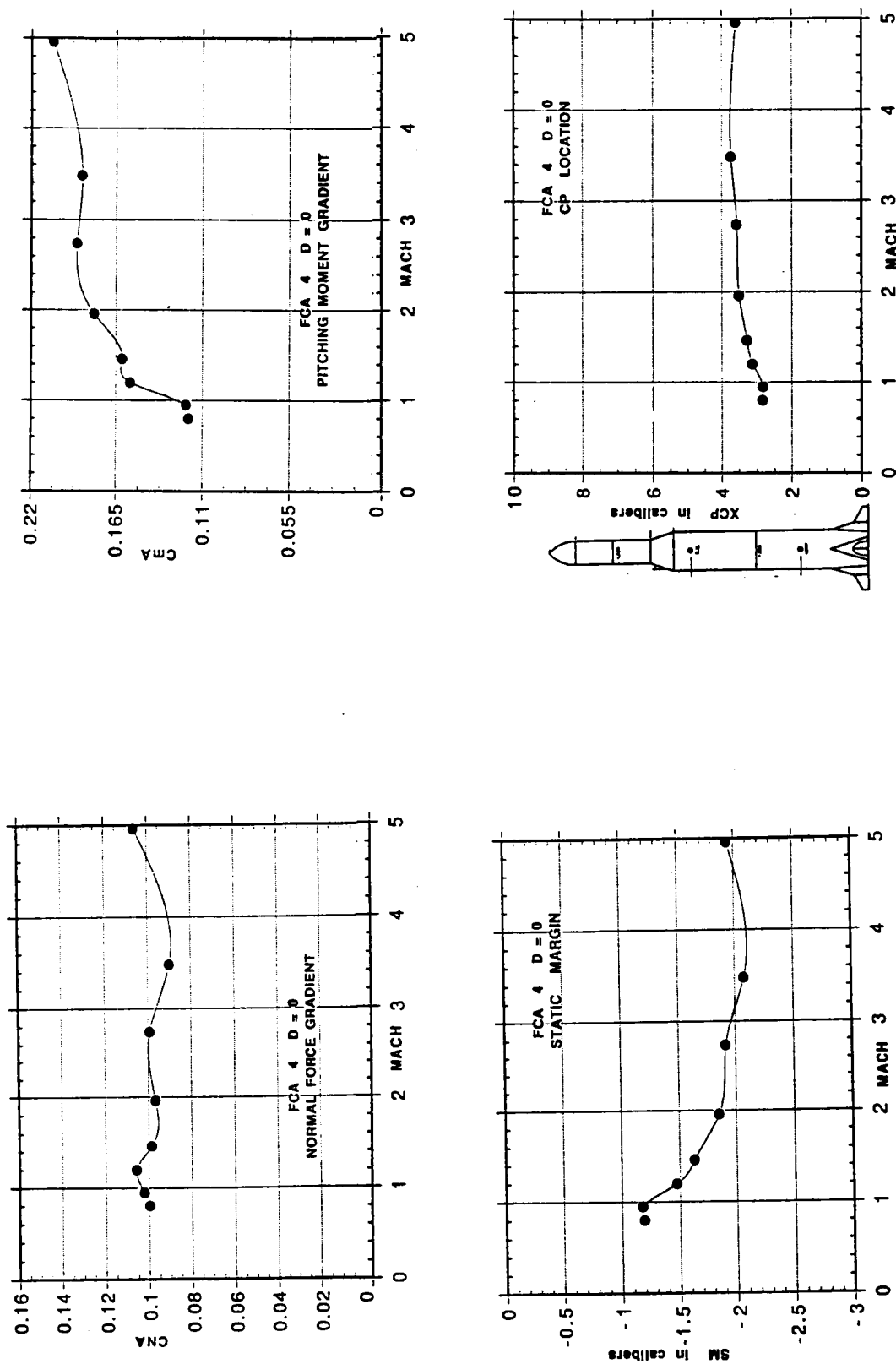


Figure 117. Stability results with FCA 4; deflection = 0°.

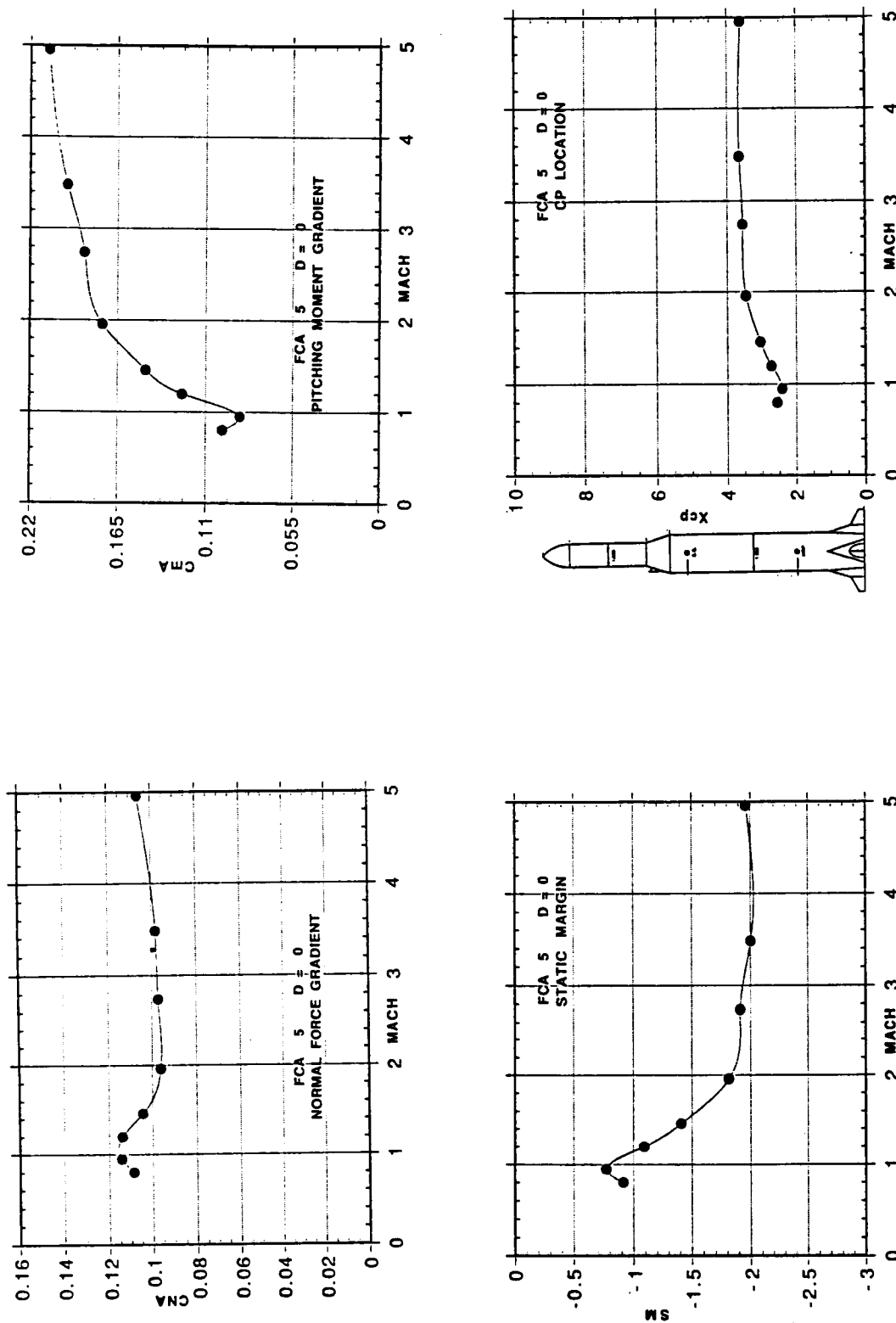


Figure 118. Stability results with FCA 5; deflection =  $0^\circ$ .

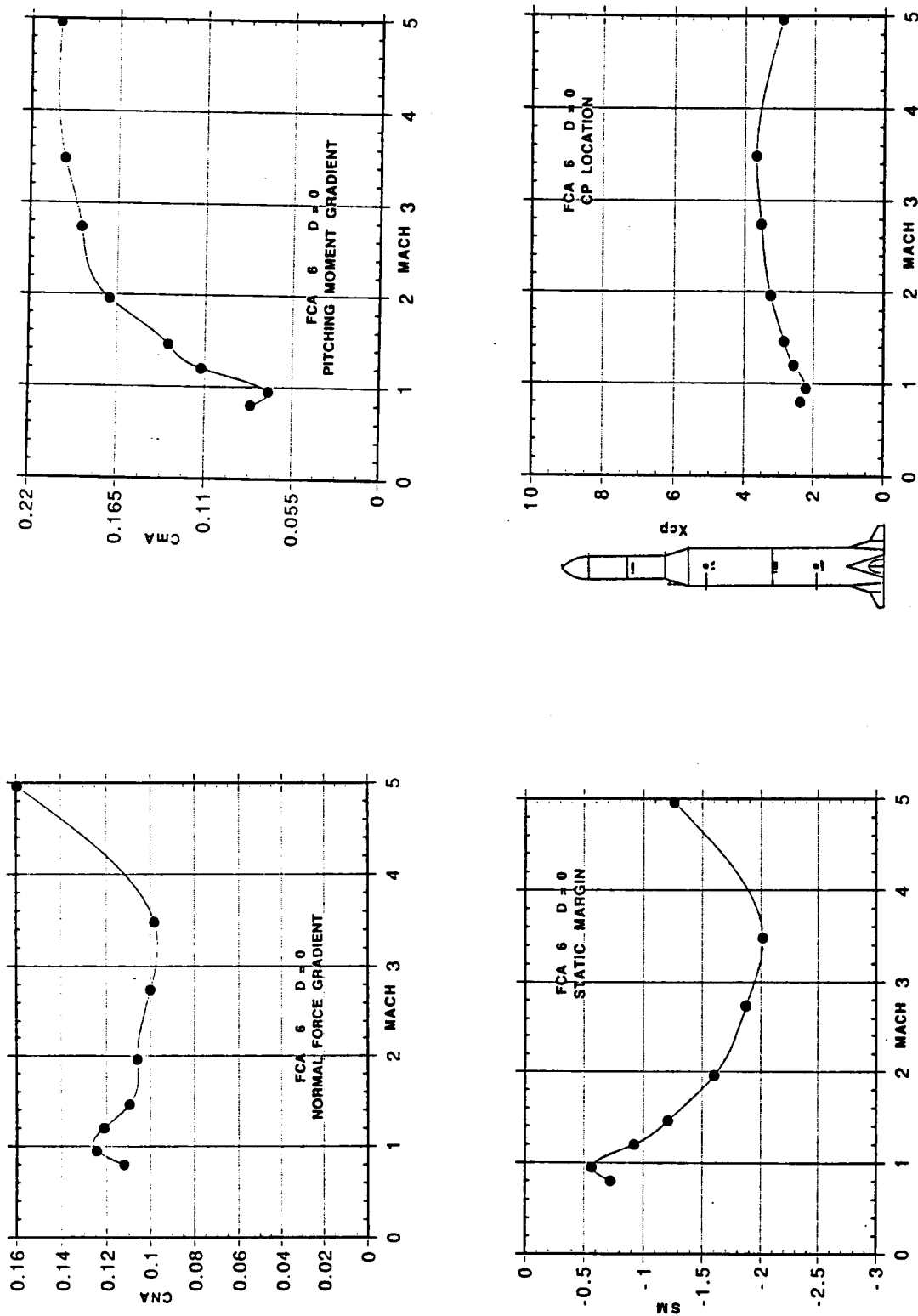


Figure 119. Stability results with FCA 6; deflection = 0°.

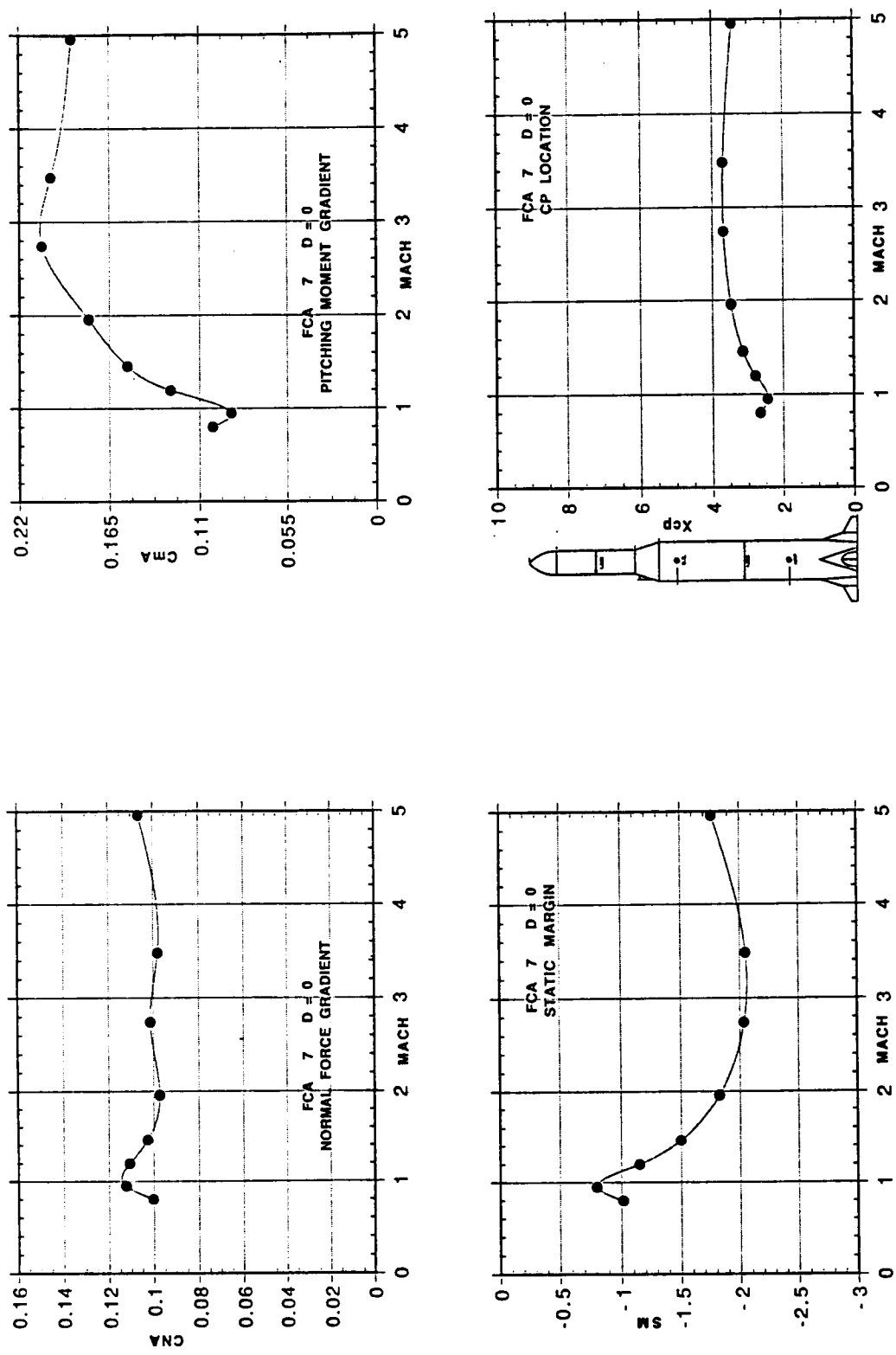


Figure 120. Stability results with FCA 7; deflection = 0°.

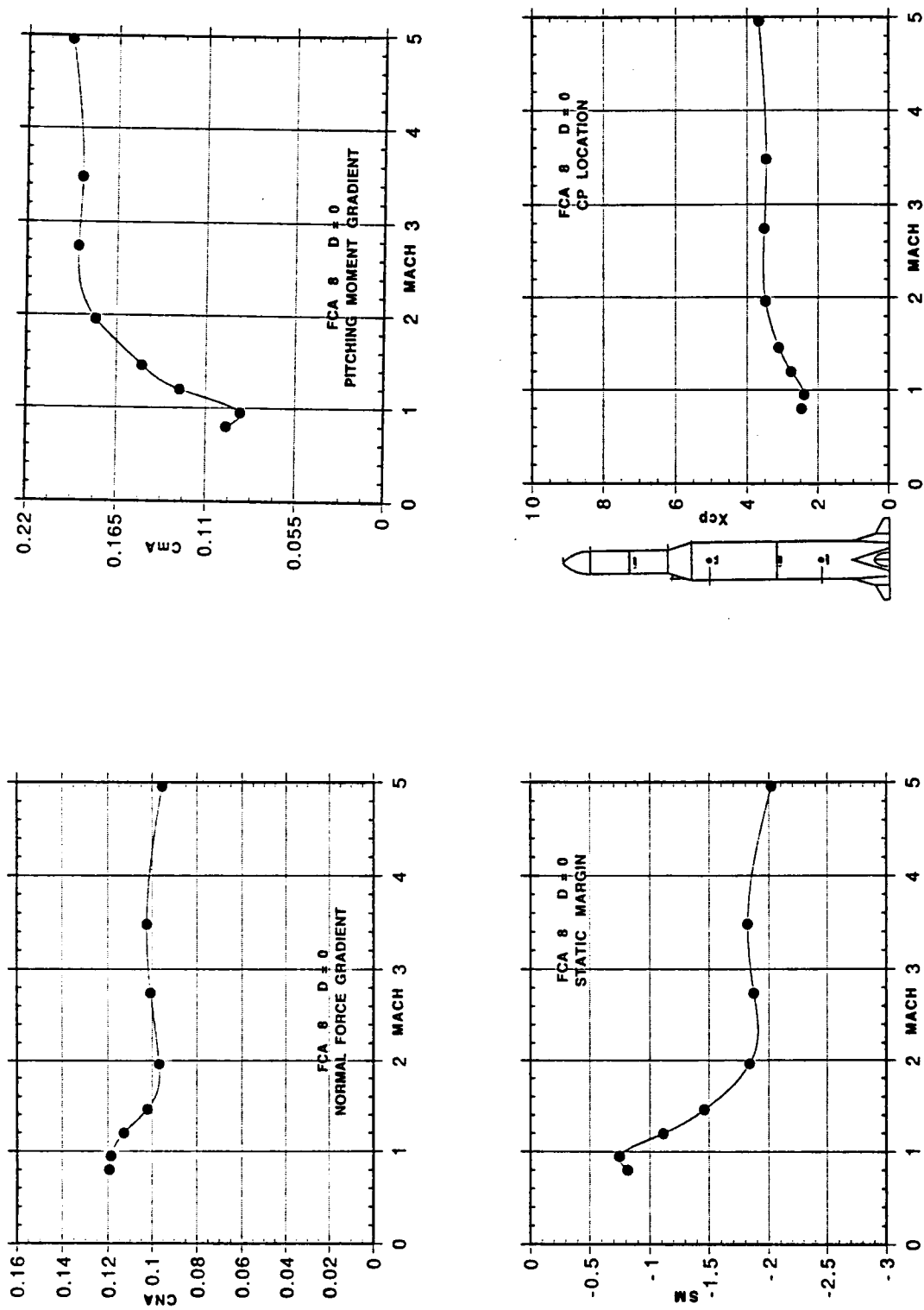


Figure 121. Stability results with FCA 8; deflection = 0°.

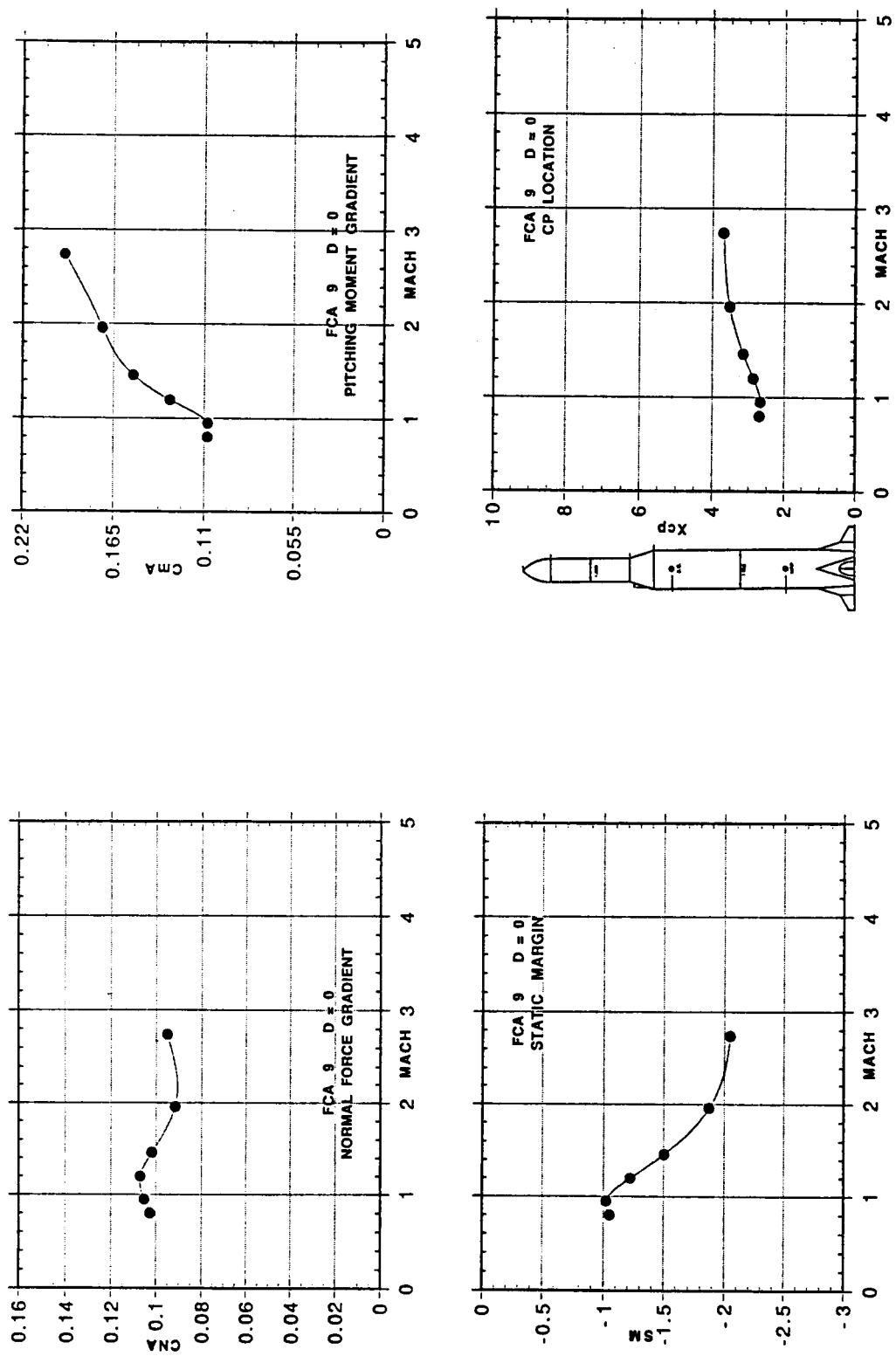


Figure 122. Stability results with FCA 9; deflection = 0°.



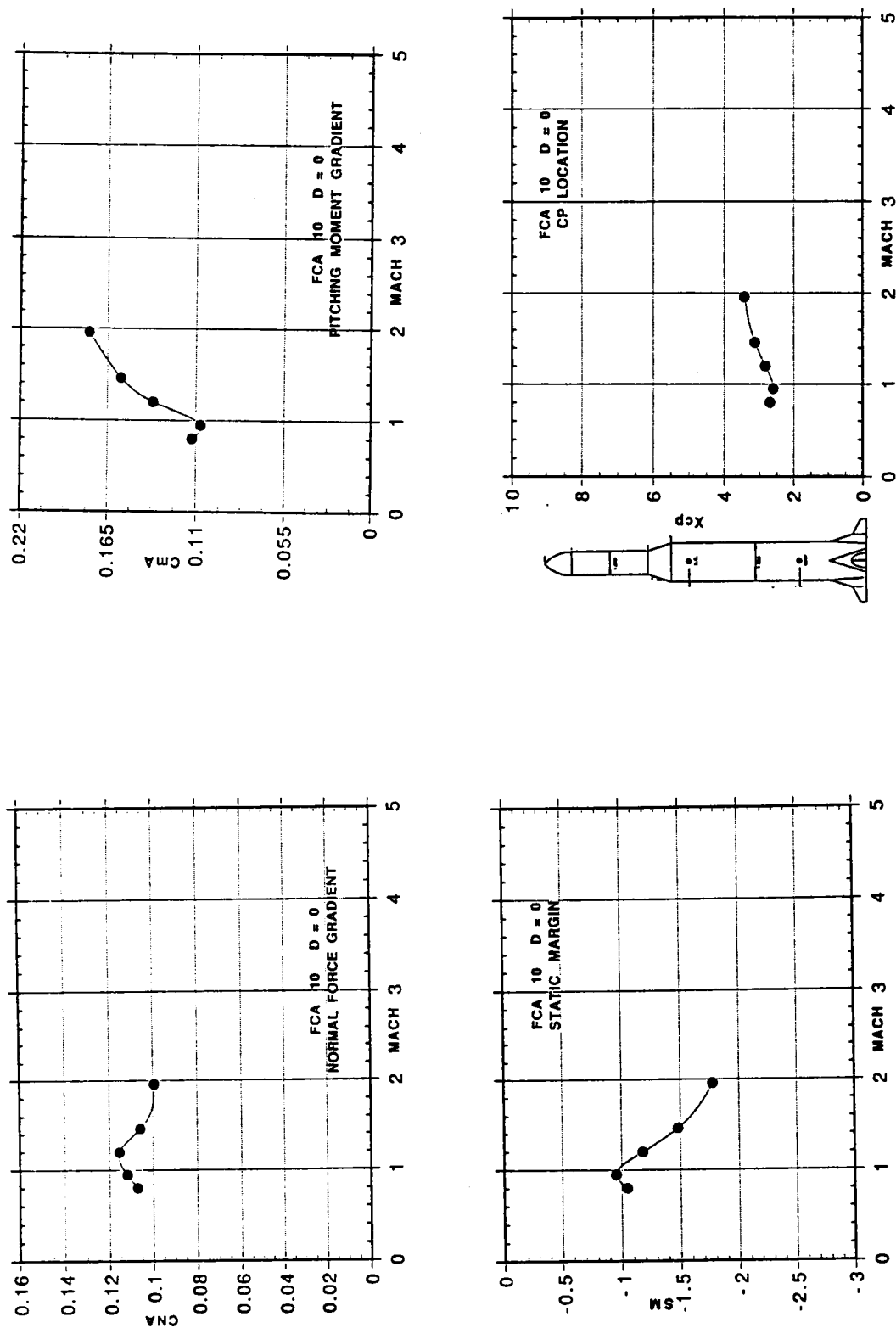


Figure 123. Stability results with FCA 10; deflection  $\approx 0^\circ$ .

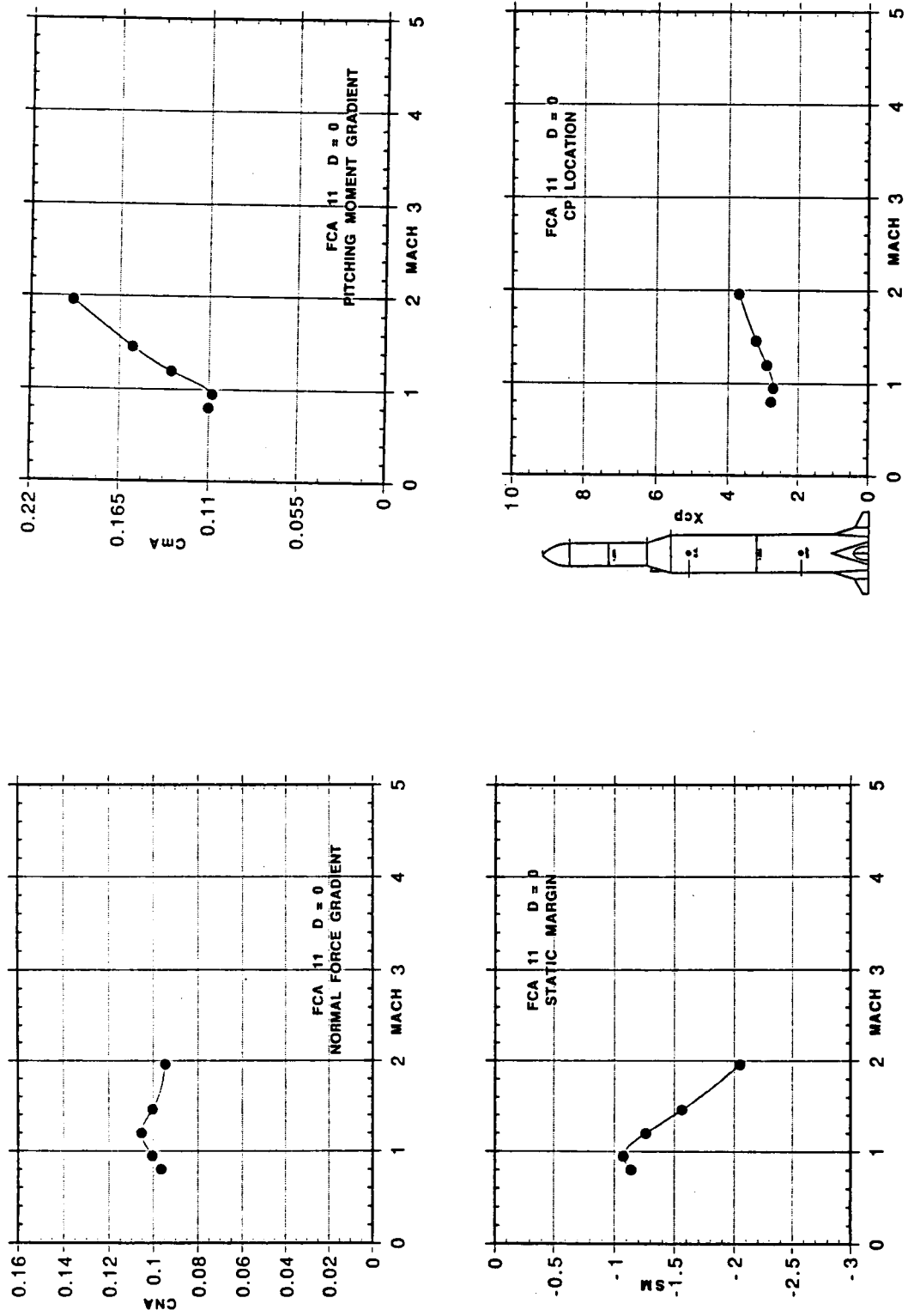


Figure 124. Stability results with FCA 11; deflection =  $0^\circ$ .

shows the  $C_{N\alpha}$  peaks at 0.124,  $C_{m\alpha}$  decreases to 0.07, and the SM increases to a dramatic  $-0.56$ . It can also be seen that the SM is its worst value at  $M = 3.5$  where  $C_{N\alpha}$  is also at its lowest value, and the LV is almost out of usable atmosphere as shown in Figure 100 in section 5.2. Figure 120 shows  $C_{N\alpha}$  increases to 0.113,  $C_{m\alpha}$  decreases to 0.09, and the best SM is  $-0.80$ . Figure 121 shows the  $C_{N\alpha}$  increases to 0.119,  $C_{m\alpha}$  decreases to 0.09, and the best SM is  $-0.75$ . Figure 122 shows  $C_{N\alpha}$  increases to 0.107,  $C_{m\alpha}$  decreases to 0.108, and the best SM is  $-1.02$ . Figure 123 shows  $C_{N\alpha}$  increases to 0.115,  $C_{m\alpha}$  decreases to 0.107, and the best SM is  $-0.96$ . Figure 124 shows  $C_{N\alpha}$  increases to 0.105,  $C_{m\alpha}$  decreases to 0.108, and the best SM is  $-1.08$ .

The FCA's with the greatest aspect ratio had the greatest normal force gradient. Additional plots of these same stability parameters for the cases of deflected fore and aft FCA's can be found in Appendix C.3. Table 7 summarizes the ratings of the aft FCA's #1 through #11 at  $\delta_F = 0^\circ$ , based on the quantitative criterion of longitudinal static stability, the SM, and shows that the  $30^\circ$  swept LE, blunt TE design provides the best flight stability. The same planform in a lenticular airfoil provides the next best flight stability.

Table 7. FCA flight stability ratings based on SM.

| FCA No. $\delta = 0^\circ$ | SM    | Planform Design                                                           |
|----------------------------|-------|---------------------------------------------------------------------------|
| NO FCA's                   | -2.0  |                                                                           |
| 6                          | -0.56 | Triangular, $30^\circ$ LE $\Lambda$ , Blunt TE, Wedge                     |
| 8                          | -0.75 | Triangular, $30^\circ$ LE $\Lambda$ , Lenticular                          |
| 5                          | -0.77 | Triangular, $30^\circ$ LE $\Lambda$ , Diamond                             |
| 7                          | -0.80 | Triangular, $30^\circ$ LE $\Lambda$ , Diamond, Clipped Tip                |
| 2                          | -0.89 | Triangular, $45^\circ$ LE $\Lambda$ , Blunt TE, Wedge                     |
| 1                          | -0.92 | Triangular, $45^\circ$ LE $\Lambda$ , Diamond                             |
| 10                         | -0.96 | Forward Swept, $-45^\circ$ LE $\Lambda$ , Diamond                         |
| 9                          | -1.02 | Aft Swept, $45^\circ$ LE $\Lambda$ , $30^\circ$ TE $\Lambda$ , Diamond    |
| 11                         | -1.08 | Trapezoidal, $45^\circ$ LE $\Lambda$ , $-30^\circ$ TE $\Lambda$ , Diamond |
| 3                          | -1.14 | Triangular, $45^\circ$ LE $\Lambda$ , Diamond, Clipped Tip                |
| 4                          | -1.18 | Triangular, $45^\circ$ LE $\Lambda$ , Lenticular                          |

Figures 125 and 126 show the Saturn V  $X_{cp}$  movements with and without aerodynamic surfaces for comparison to the experimental LV. Figure 127 shows the small axial force increase for the Saturn V using aerodynamic surfaces.

In summary, the static flight stability analysis has shown that dramatic improvements in flight stability are achieved with the use of all the aft FCA designs. The SM was used as quantification of longitudinal static stability. The experimental LV longitudinal stability improvements in SM range from 72% to 41% over that of the vehicle with no FCA's used.

### 7.3 Control Analysis

The control requirement from eq. (1) in Chapter V is to have the control ratio (CR)  $\geq 1.5$  where the CR is the ratio of the control torque (CT) to the aerodynamic torque (AT). With no FCA's the total control torque is that provided by the thrust vector control (TVC) of the gimbaled engines. With the use of aft FCA's the total CT becomes the sum of that provided by both the TVC and the FCA's. From equation (5) in Chapter V,

$$CR = \frac{T \sin \delta_E (E) \ell_E}{C_{N\alpha} \alpha Q S \ell_{cp}} \geq 1.5 \quad (33)$$

where all variables are defined in Chapter V. Since

$$C_N Q S \ell_{cp} = C_m Q S d \quad (34)$$

It follows that,

$$CR = \frac{T \sin \delta_E (E) \ell_E}{C_N Q S \ell_{cp}} = \frac{T \sin \delta_E (E) \ell_E}{C_m Q S d} \geq 1.5 \quad (35)$$

With deflection of the FCA's the pitching moment coefficient involves more terms. Using a relationship developed in the classic work of Etkin [59],

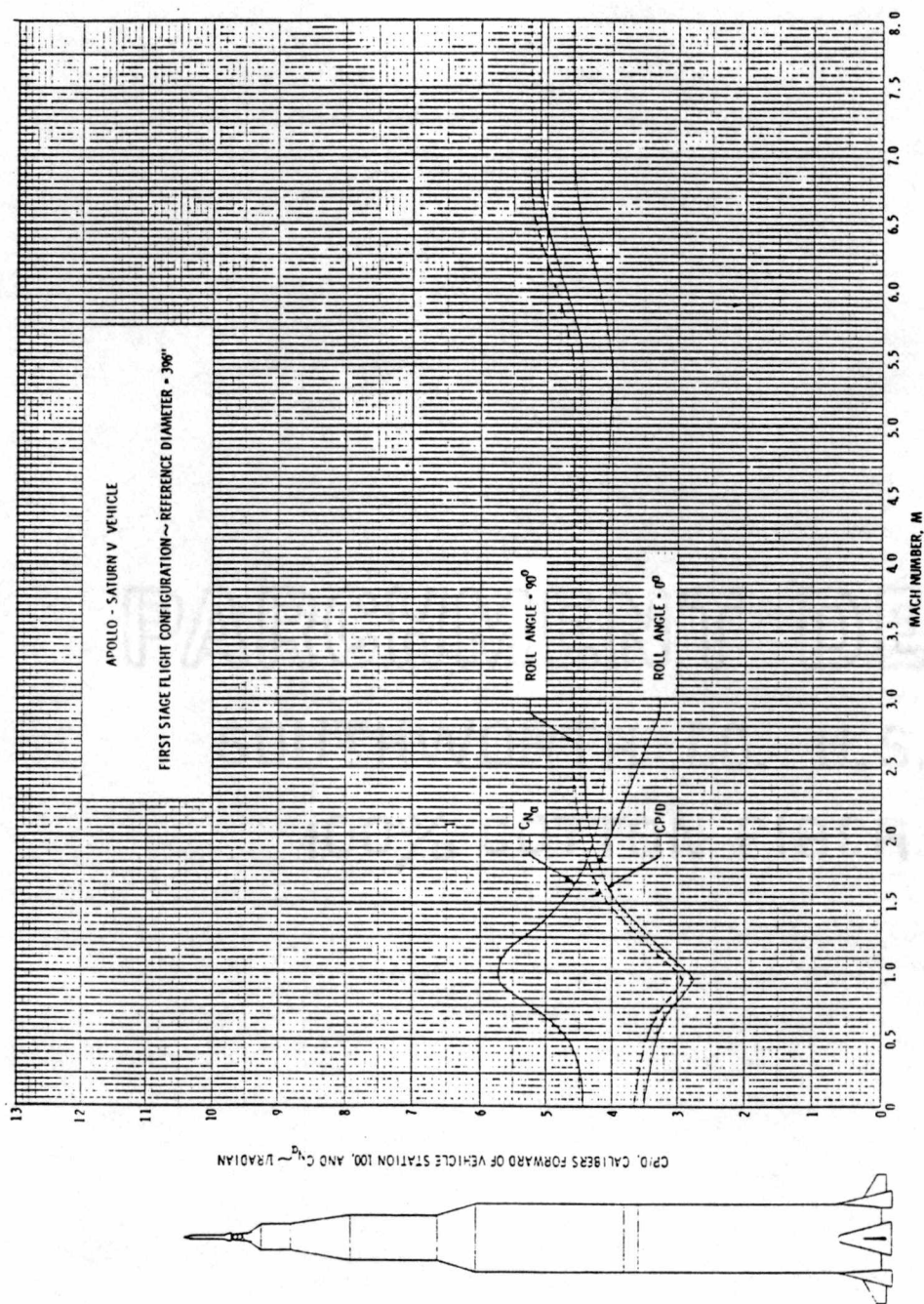
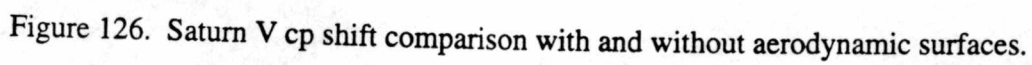


Figure 125. Saturn V cp shift comparison.



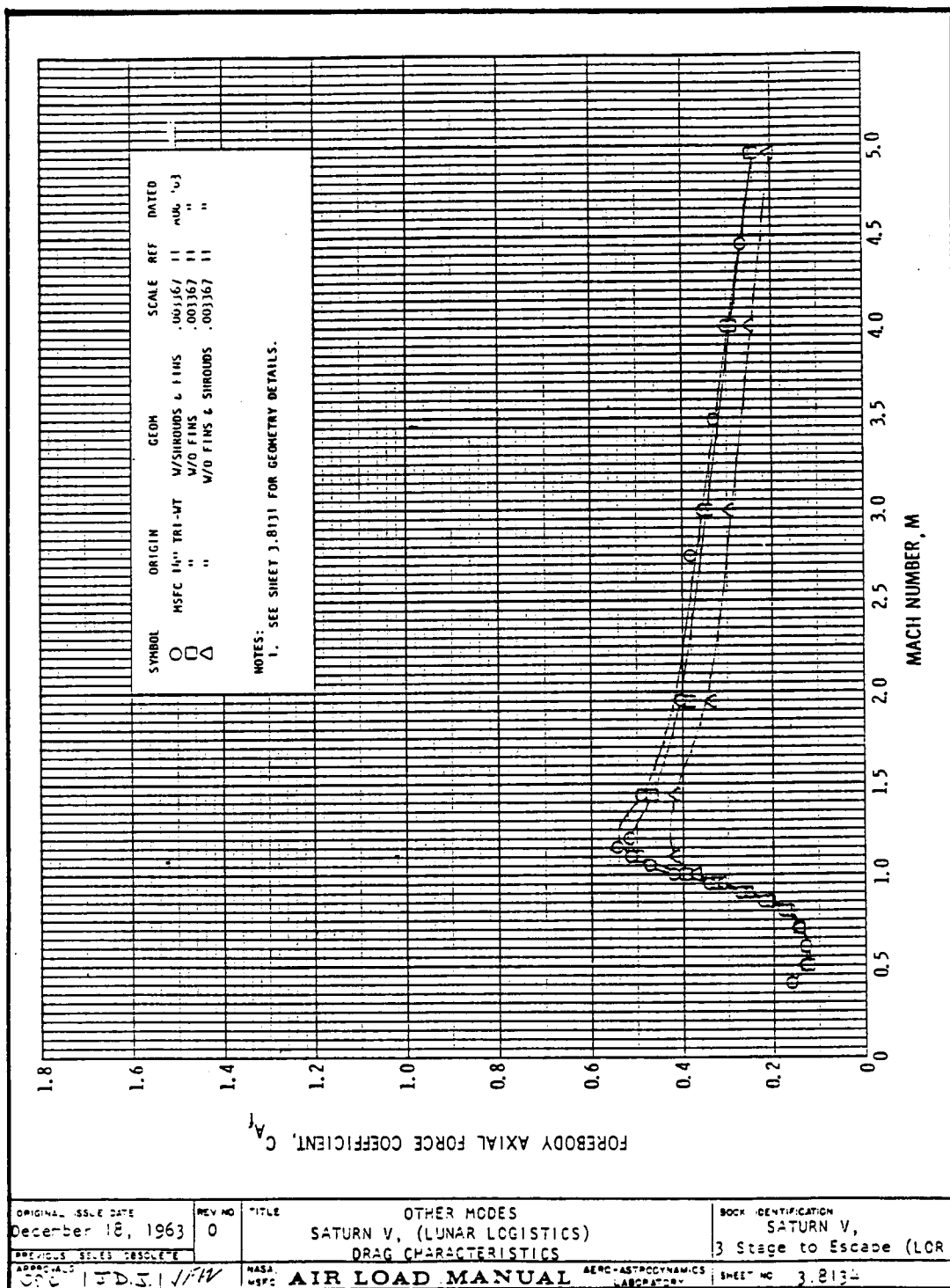


Figure 127. Saturn V axial force comparisons with fins.

$$C_m = C_{m_0} + C_{m_\alpha} \alpha + C_{m_{\delta_F}} \delta_F \quad (36)$$

where  $C_{m_0}$  = pitching moment coefficient at  $\alpha = 0^\circ$ ; when the FCA deflection is utilized to augment the TVC control the control stability derivative is [59],

$$C_{m_{\delta_F}} = \frac{\partial C_m}{\partial \delta_F} \equiv \text{FCA Control Power} \quad (37)$$

The more negative the value of  $C_{m_{\delta_F}}$ , the more effective the FCA control power of the experimental LV. For no deflection of the controls, equation (36) reduces to  $C_m = C_{m_\alpha} \alpha$ .

Thus, equation (35) becomes,

$$CR = \frac{T \sin \delta_E (E) \ell_E}{(C_{m_0} + C_{m_\alpha} \alpha + C_{m_{\delta_F}} \delta_F) Q S d} \geq 1.5 \quad (38)$$

For the experimental LV, the CR is now calculated using equation (33) first with no FCA's, then with FCA's but no deflection, and lastly using equation (38) with FCA's and FCA deflection.

From Chapter V,  $S = 597.56 \text{ ft}^2$ ;  $d = 27.58 \text{ ft}$ ; and  $\delta_E = \alpha = 4^\circ$ . In Chapter V, before the FCA's were designed, the preliminary mass properties and corresponding trajectory data for the experimental aft-cg LV were used. Subsequently, the NASA MSFC Systems Engineering Division has updated and refined the mass properties for the experimental aft-cg LV. The updated far aft-cg mass properties and corresponding trajectory data, which challenge the FCA's to a greater degree and can be found in Appendix A, are used in this analysis. The trajectory data shows that at Max Q conditions, the six engines are only producing 491,806.5 lb of thrust each; max Q = 722.56 psf at  $M = 1.31$ . From Table 6 above, for no FCA's, the interpolated values



corresponding to  $M = 1.31$  are  $C_{N_\alpha} = 0.081208$  and  $C_{m_\alpha} = 0.176911$ ;  $\Rightarrow \ell_{cp} = 60.08$  ft.

Using these values and  $\ell_E = 45.82$  ft in equation (33) with no FCA's,

$$CR = \frac{9,431,594 \text{ ft-lb}}{8,426,557 \text{ ft-lb}} = 1.12 \quad (39)$$

which shows that the FCA's are presented with a greater challenge to augment the control torque than was the case in Chapter V using preliminary data, where the CR was calculated to be 1.30.

Now, using FCA 6 but no deflection, Table 6 above shows that for FCA 6 with  $\delta_F = 0^\circ$  at  $M = 1.31$  (interpolated),  $C_{N_\alpha} = 0.11629$  and  $C_{m_\alpha} = 0.12085$ ;  $\Rightarrow \ell_{cp} = 28.66$  ft.

Substituting these values into equation (33) yields,

$$CR = \frac{9,431,594 \text{ ft-lb}}{5,756,255 \text{ ft-lb}} = 1.64 \quad (40)$$

This shows that the controllability is more than adequate at max Q with only the presence of the FCA's using no deflection at all. It can be seen that the improvement in the CR by the mere presence of the FCA's with no deflection at all is 46%.

Now FCA 6 is used with deflection. Figures 128 and 129 show the control power ( $C_{m_{\delta_F}}$ ) for selected FCA's through the range of Mach numbers. It can be seen that the control power is greatest for most aft FCA's at max Q. Next,  $C_m$  values were plotted against  $\alpha$  for the cases of FCA deflection. The value of  $C_m$  when  $\alpha = 0^\circ$  were tabulated, shifting data curves by a small amount when necessary, i.e., when it was known that a curve should pass through the origin. From the reduced and shift corrected data (using very conservative values of  $C_{m_0}$ ), a sample  $C_{m_0}$  tabulation for the case of a deflected FCA is,

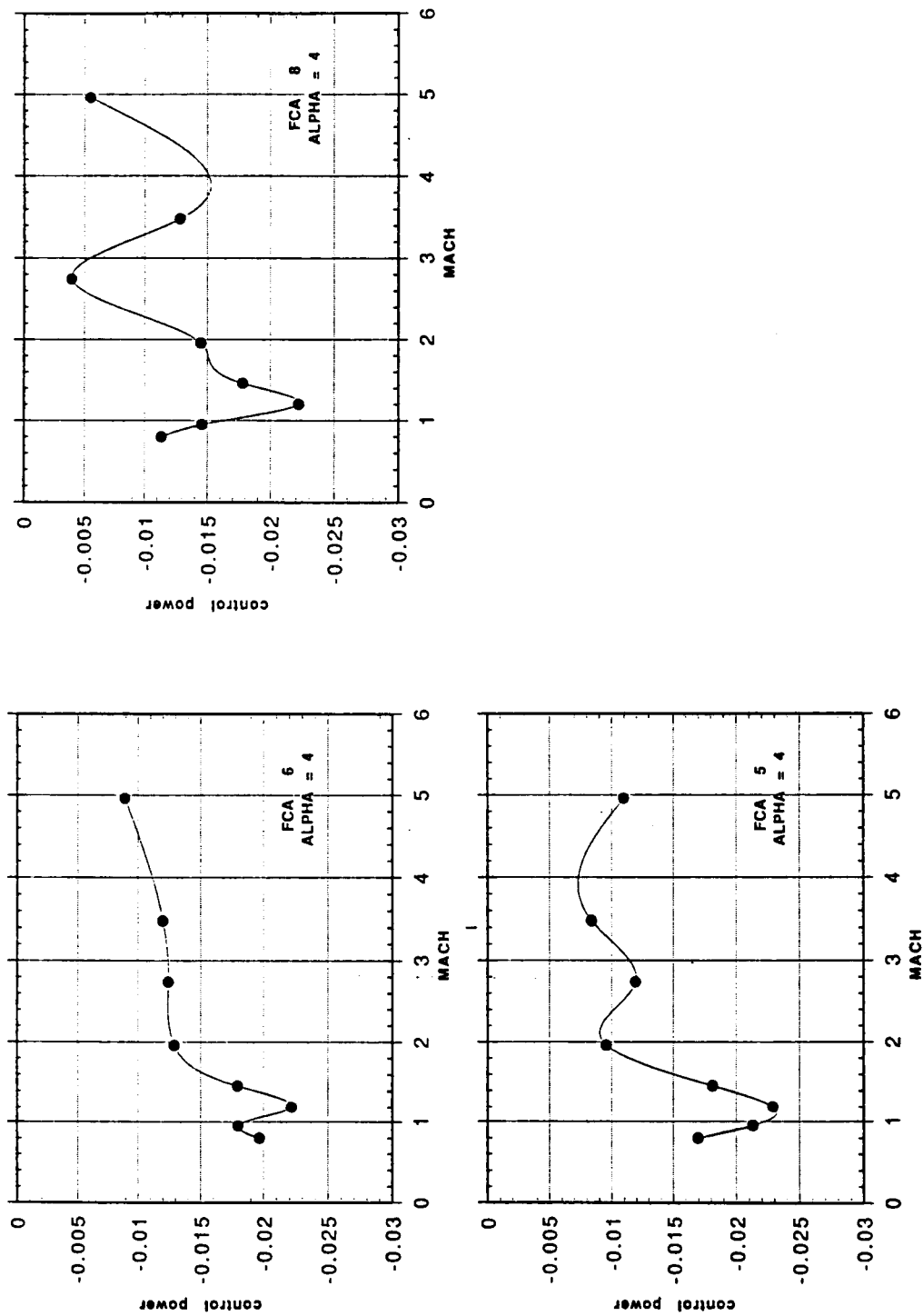


Figure 128. Control power vs. Mach number for FCA 5, 6, and 8.

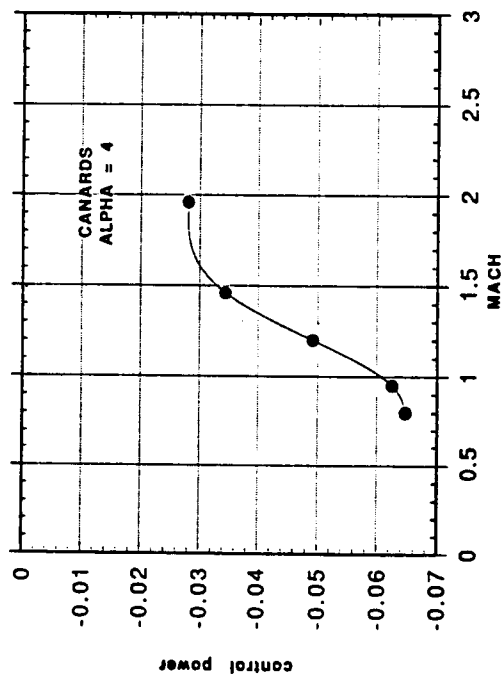
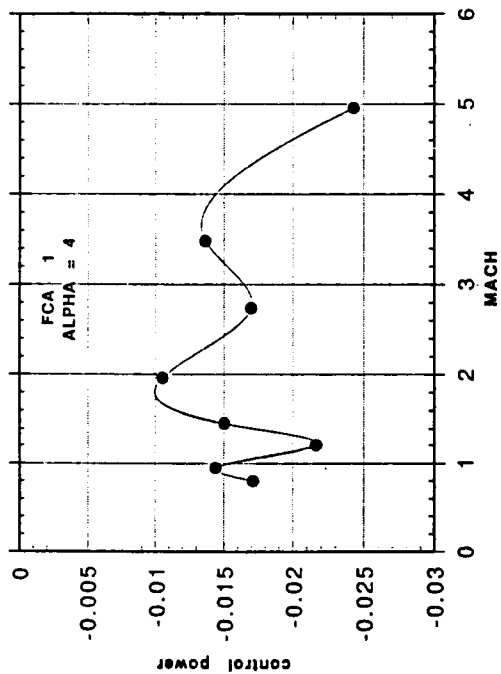
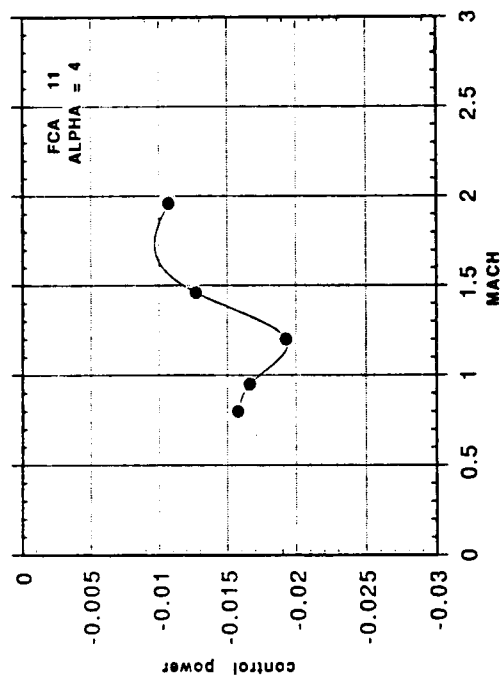


Figure 129. Control power vs. Mach number for FCA 1, 11, and canard.

| FCA 6 | $\delta_F^\circ$ | M = 1.20<br>$C_{m_0}$ | M = 1.46<br>$C_{m_0}$ | M = 1.3<br>$C_{m_0}$ |
|-------|------------------|-----------------------|-----------------------|----------------------|
|       | 0                | 0                     | 0                     | 0                    |
|       | 3                | -0.1073               | -0.047                | -0.8411              |
|       | 6                | -0.2342               | -0.1491               | -0.2015              |
|       | 9                | -0.3284               | -0.2331               | -0.2918              |
|       | 12               | -0.3800               | -0.3135               | -0.3544              |
|       | 15               | -0.4654               | -0.3551               | -0.423               |

$C_m$  was also plotted against  $\delta_F$  for fixed  $\alpha$  and the slopes calculated. Table 8 shows selected calculated longitudinal control derivatives for  $\alpha = 4^\circ$  through the Mach range.

Table 8. Calculated longitudinal control derivatives for  $\alpha = 4^\circ$ .

| M    | FCA 1<br>$C_{m_\delta}$ | FCA 5<br>$C_{m_\delta}$ | FCA 6<br>$C_{m_\delta}$ | FCS 8<br>$C_{m_\delta}$ | FCA 11<br>$C_{m_\delta}$ | Canards<br>$C_{m_\delta}$ |
|------|-------------------------|-------------------------|-------------------------|-------------------------|--------------------------|---------------------------|
| 0.8  | -0.01712                | -0.01699                | -0.0197                 | -0.01136                | -0.01579                 | -0.06477                  |
| 0.95 | -0.01442                | -0.0213                 | -0.018                  | -0.01457                | -0.01665                 | -0.0626                   |
| 1.20 | -0.02167                | -0.02288                | -0.02218                | -0.02222                | -0.0193                  | -0.04922                  |
| 1.46 | -0.01503                | -0.01815                | -0.01795                | -0.01782                | -0.01274                 | -0.03446                  |
| 1.96 | -0.01053                | -0.009573               | -0.0129                 | -0.01448                | -0.01071                 | -0.02804                  |
| 2.74 | -0.01696                | -0.01193                | -0.01243                | -0.003922               |                          |                           |
| 3.48 | -0.01365                | -0.0084                 | -0.012                  | -0.01283                |                          |                           |
| 4.96 | -0.0243                 | -0.01098                | -0.008933               | -0.0055                 |                          |                           |

For FCA 6 at  $M = 1.31$ ,  $C_{m_\alpha} = 0.12085$  (at fixed  $\delta_F$ ) from above, and from Table 6,  $C_{m_{\delta_F}} = -0.020553$  (at fixed  $\alpha$ ). Substituting these values into equation (38), CR is determined as a function of FCA deflection angle. Table 9 shows the calculated control ratio values for various FCA deflection angles, and Figure 130 is the control analysis summary plot which shows the control ratio for no FCA's, the required amount, and that available through the use of the deflected FCA's.

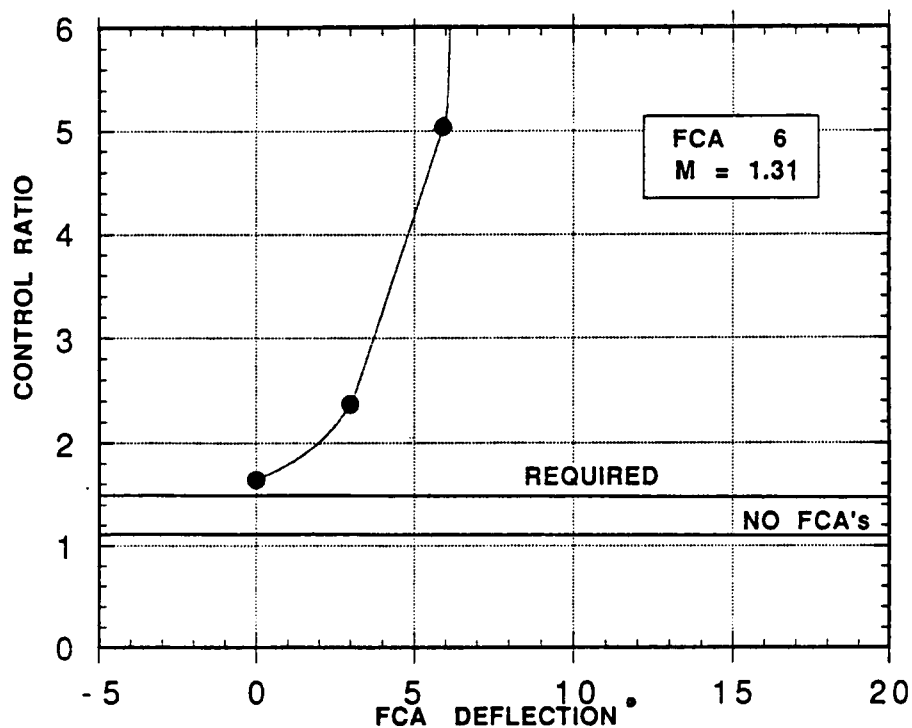


Figure 130. Control analysis summary plot.

Table 9. Calculated control ratio values for FCA deflection angles.

| $\delta_F^\circ$ | Control Ratio |
|------------------|---------------|
| No FCA's         | 1.12          |
| Min Requirement  | 1.50          |
| 0                | 1.64          |
| 3                | 2.35          |
| 6                | 4.99          |
| 9                | 120.          |

When the aerodynamic moments on the vehicle are zero, the vehicle is said to be trimmed. From equation (36)

$$C_m = C_{m_0} + C_{m_\alpha} \alpha + C_{m_{\delta_F}} \delta_F = 0 \text{ for trim} \Rightarrow \delta_{F_{Trim}} = 9.1^\circ \quad (41)$$

Thus, (for FCA 6) when the FCA's are deflected  $9.1^\circ$  there is no resultant aerodynamic moment on the LV for  $\alpha = 4^\circ$ . For deflections above  $9.1^\circ$ , the FCA's would be available to provide excess control along with the TVC.

Even greater results are achievable by using the FCA's and engine shrouds at 45°. It was found that the Saturn V aerodynamic surfaces and engine shrouds became more efficient when the LV was rolled 45° in the wind tunnel tests. [60] With the FCA surfaces aligned with the body axes, i.e., two surfaces in the horizontal position and two in the vertical position, only two are working in the pitch plane. This orientation is referred to in the literature as the (+) configuration. When the FCA surfaces are rolled in orientation w.r.t. the body axes, all four would be working in the pitch plane. This orientation is referred to as the (X) configuration. (See Figure 131.) [61] For example,

$$0^\circ (+): \quad N_F = 2 C_{N_F} Q S_F$$

$$45^\circ (X): \quad N = 4 (\sin 45^\circ) C_{N_F} Q S_F = 2.83 C_{N_F} Q S_F \quad (42)$$

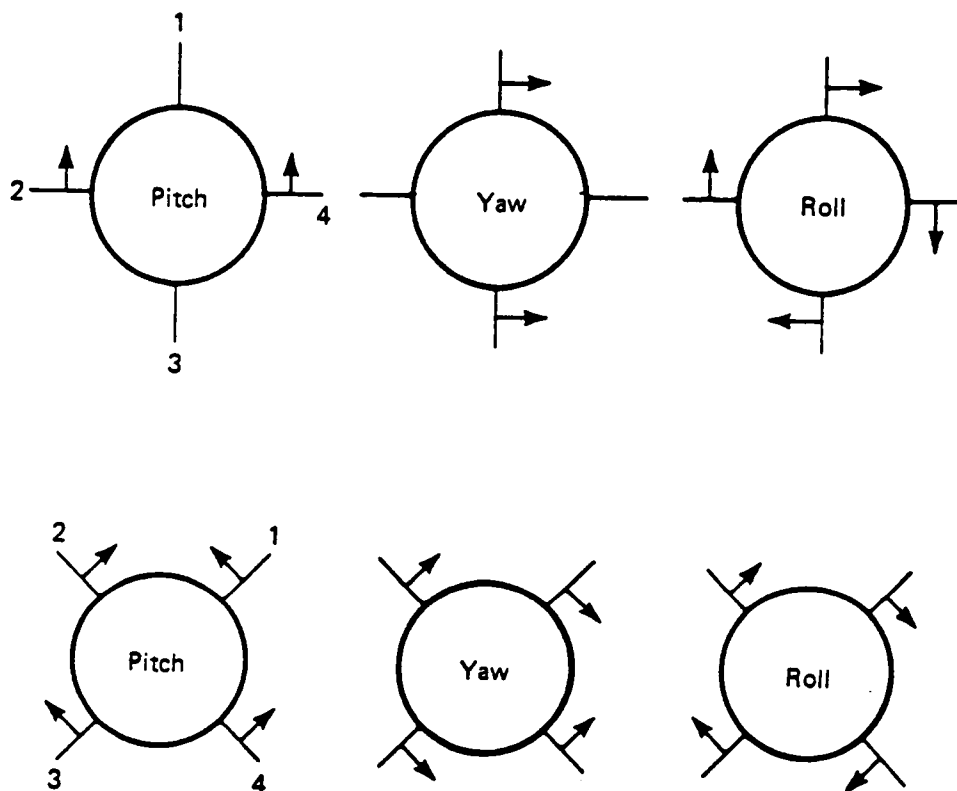


Figure 131. Definitions of positive pitch, yaw, and roll control (arrow shows direction of forces and leading edge of panel, looking upstream).

For canard control, from Table 6 above, for  $\alpha = 4^\circ$  and  $M = 1.31$  (interpolation),  $C_{m_{\delta_C}} = -0.04354$  and from the sample tabulation of reduced and shift corrected data for  $C_{m_0}$  for the canards,

| Canards | $\delta_C^\circ$ | $M = 1.20$<br>$C_{m_0}$ | $M = 1.46$<br>$C_{m_0}$ | $M = 1.3$<br>$C_{m_0}$ |
|---------|------------------|-------------------------|-------------------------|------------------------|
|         | 0                | 0                       | 0                       | 0                      |
|         | 5                | -0.2760                 | -0.2197                 | -0.2543                |
|         | 10               | -0.575                  | -0.4035                 | -0.5090                |
|         | 15               | -0.8103                 | -0.6019                 | -0.7301                |
|         | 20               | -0.9391                 | -0.6528                 | -0.829                 |
|         | 25               | -0.8394                 | -0.7911                 | -0.8208                |
|         | 30               | -1.0719                 | -0.8328                 | -0.9799                |

Using the trim equation (41) for the canards, it can be seen that the canards are capable of trimming out the aerodynamic moment with only  $5^\circ$  deflection.

In summary, the control analysis shows that the mere presence of the FCA's with no deflection increases the control ratio by 46%. When FCA deflection is utilized, the control ratio increases by 110% for  $3^\circ$  FCA deflection, to 346% for  $6^\circ$  FCA deflection, and at  $9.1^\circ$  FCA deflection, the aerodynamic moment is completely trimmed for the  $\alpha = 4^\circ$  case considered. The canards are able to trim the vehicle at  $5^\circ$  deflection. Based on the analysis of this chapter, three FCA designs were selected for the follow-on dynamic wind tunnel test program.

## CHAPTER VIII. DYNAMIC WIND TUNNEL TEST

### 8.1 Dynamic Stability Wind Tunnel Test Program Requirements

The purpose of the Dynamic Stability Wind Tunnel Test Program was to obtain the damping derivatives for the different FCA's tested on the experimental LV, so that the dynamic stability characteristics could be compared. A LV in its ascent trajectory is exposed to unsteady flow fields and asymmetric vortex shedding that may affect the vehicle's motion significantly. Unsteady flowfields affect the dynamic stability derivatives. Whereas the static stability derivatives are a function of aerodynamic angle, control surface deflection, and Mach number, the dynamic stability derivatives are a function of aerodynamic angle rates-of-change, control surface deflection rates, body rates, and Mach number, e.g.,

$$C_{m_{\dot{\alpha}}} , \quad C_{n_{\dot{\beta}}} , \quad C_{m_{\dot{\delta}}} , \quad C_{m_q} , \quad C_{\ell_{\dot{\phi}}}$$

where

$$C_{m_{\dot{\alpha}}} \equiv \frac{\partial C_m}{\partial \dot{\alpha}} \quad \text{etc.}$$

In certain flight vehicles, significant aerodynamic coupling between longitudinal and lateral-directional degrees of motion can occur. Figure 132 [62] shows cross, cross-coupling, and damping dynamic stability derivatives. Damping derivatives are defined such that each vector component of the moment,  $\bar{M} = (l, m, n)$ , is differentiated with respect to the same corresponding vector component of angular velocity,  $\bar{\omega} = (p, q, r)$ , i.e.,

$$C_{\ell_p}, C_{m_q}, C_{n_r} \quad \text{where} \quad C_{\ell_p} \equiv \frac{\partial C_{\ell}}{\partial p} \quad \text{etc.}$$

Table 10 shows some types of moment dynamic stability derivatives. [63]



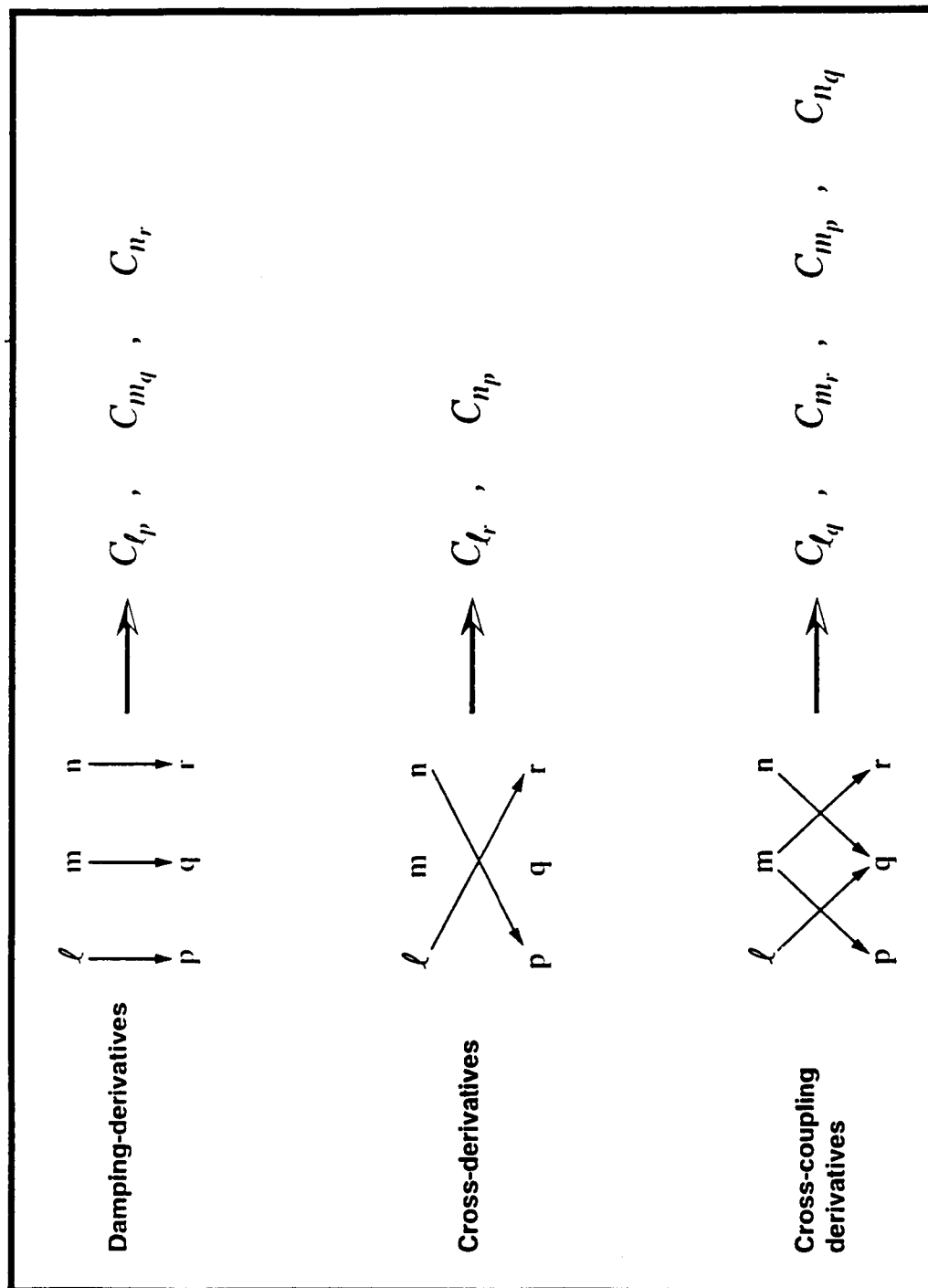


Figure 132. The classification of dynamic stability derivatives.

Table 10. Moment dynamic stability derivatives.

| PURE ROTATION                                                                                                                      |                                                                                                |                                                                                                                                                                        |
|------------------------------------------------------------------------------------------------------------------------------------|------------------------------------------------------------------------------------------------|------------------------------------------------------------------------------------------------------------------------------------------------------------------------|
| $C_{\ell_p} = \frac{\partial C_{\ell}}{\partial \frac{pb}{2V}}$                                                                    | $C_{m_p} = \frac{\partial C_m}{\partial \frac{pb}{2V}}$                                        | $C_{n_p} = \frac{\partial C_n}{\partial \frac{pb}{2V}}$                                                                                                                |
| $C_{\ell_q} = \frac{\partial C_{\ell}}{\partial \frac{qc}{2V}}$                                                                    | $C_{m_q} = \frac{\partial C_m}{\partial \frac{qc}{2V}}$                                        | $C_{n_q} = \frac{\partial C_n}{\partial \frac{qc}{2V}}$                                                                                                                |
| $C_{\ell_r} = \frac{\partial C_{\ell}}{\partial \frac{rb}{2V}}$                                                                    | $C_{m_r} = \frac{\partial C_m}{\partial \frac{rb}{2V}}$                                        | $C_{n_r} = \frac{\partial C_n}{\partial \frac{rb}{2V}}$                                                                                                                |
| TRANSLATIONAL ACCELERATION                                                                                                         |                                                                                                |                                                                                                                                                                        |
| $C_{\ell_{\dot{\alpha}}} = \frac{\partial C_{\ell}}{\partial \frac{\dot{\alpha}c}{2V}}$                                            | $C_{m_{\dot{\alpha}}} = \frac{\partial C_m}{\partial \frac{\dot{\alpha}c}{2V}}$                | $C_{n_{\dot{\alpha}}} = \frac{\partial C_n}{\partial \frac{\dot{\alpha}c}{2V}}$                                                                                        |
| $C_{\ell_{\dot{\beta}}} = \frac{\partial C_{\ell}}{\partial \frac{\dot{\beta}b}{2V}}$                                              | $C_{m_{\dot{\beta}}} = \frac{\partial C_m}{\partial \frac{\dot{\beta}b}{2V}}$                  | $C_{n_{\dot{\beta}}} = \frac{\partial C_n}{\partial \frac{\dot{\beta}b}{2V}}$                                                                                          |
| OSCILLATION AROUND FIXED AXIS                                                                                                      |                                                                                                |                                                                                                                                                                        |
| DAMPING<br>DERIVATIVES                                                                                                             | CROSS DERIVATIVES                                                                              | CROSS-COUPLING<br>DERIVATIVES                                                                                                                                          |
| $C_{\ell_p} + C_{\ell_{\dot{\beta}}} \sin\alpha$<br>$C_{m_q} + C_{m_{\dot{\alpha}}}$<br>$C_{n_r} - C_{n_{\dot{\beta}}} \cos\alpha$ | $C_{\ell_r} - C_{\ell_{\dot{\beta}}} \cos\alpha$<br>$C_{n_p} + C_{n_{\dot{\beta}}} \sin\alpha$ | $C_{\ell_q} + C_{\ell_{\dot{\alpha}}}$<br>$C_{m_r} - C_{m_{\dot{\beta}}} \cos\alpha$<br>$C_{m_p} + C_{m_{\dot{\beta}}} \sin\alpha$<br>$C_{n_q} + C_{n_{\dot{\alpha}}}$ |

In order to expedite the dynamic wind tunnel test program for the experimental LV, the test requirements have been limited to only the most important, the rates of change of moment coefficients with angular velocities, and rates of change of moment coefficients with control surface deflections, for three selected vehicle configurations. The longitudinal dynamic stability derivatives have been obtained in wind tunnel tests and are discussed here. The longitudinal dynamic stability derivatives, i.e., the pitch damping derivatives ( $C_{mq} + C_{m\dot{\alpha}}$ ), and the corresponding  $C_{m\alpha}$  for a Mach range of 0.8 to 2.0 and FCA deflections from  $0^\circ$  to  $15^\circ$ , were the requirements.

Oscillation around a fixed axis results in a sum of a purely rotary derivative and a translational acceleration derivative. In the longitudinal case, these are  $C_{mq}$  and  $C_{m\dot{\alpha}}$ , respectively, which are now explained. For a free-flying vehicle, the variations in angle of pitch,  $q$ , and in  $\alpha$  can occur independently of each other, and each gives rise to a different longitudinal distribution of the normal velocity. The distribution due to the angle-of-pitch variation, i.e., due to the pitching velocity  $q$ , varies along the LV body and intersects zero at the axis of rotation, while the distribution due to the  $\alpha$  variation,  $\dot{\alpha}$  is constant along the fuselage.

In the case of an oscillation about a fixed axis, both variations occur simultaneously, and even if the two variables  $q$  and  $\dot{\alpha}$  are numerically equal, their effects are different and have to be superimposed. Figure 133 [62, 64, 65] shows these different motions. The contribution due to  $\dot{\alpha}$  is equivalent to one due to a vertical acceleration, since  $\dot{\alpha} = \ddot{Z}/V$ .

For more insight on the two different angular rates,  $\dot{\alpha}$  and  $q$ , which produce aerodynamic moments proportional to the damping terms, one or both of the angular velocities must be non-zero. One simple planar motion that produces a  $q \neq 0$  and  $\dot{\alpha} = 0$  is the circular looping motion shown in Figure 133. In this case, the in-plane moment,

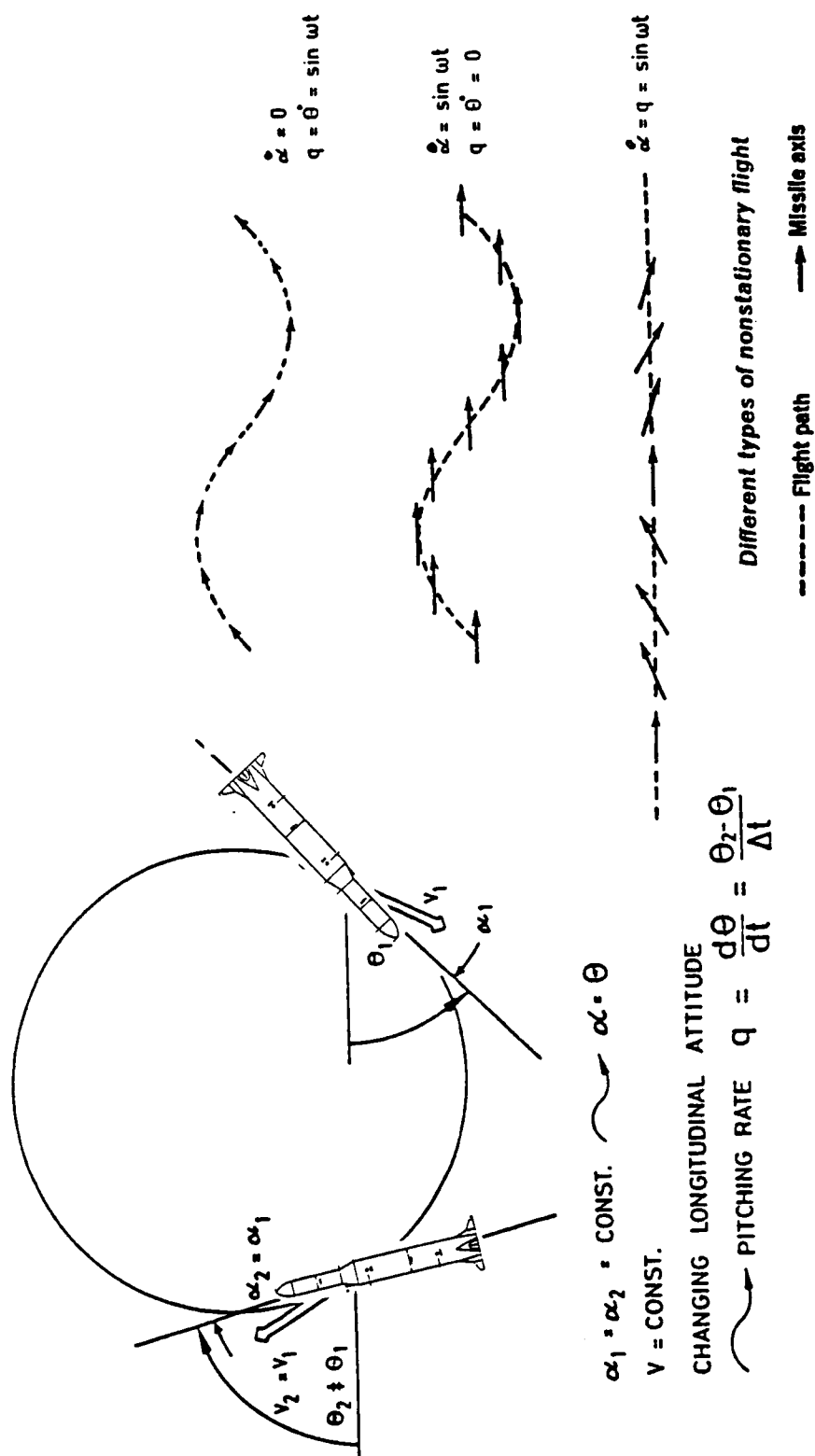


Figure 133. Rates of change of pitch rate vs. rate of change of angle of attack.

$C_m$ , contains contributions from both the pitching moment slope  $C_{m\alpha}$  and  $C_{mq}$ . The contribution from  $C_{m\alpha}$  can be eliminated if the vehicle longitudinal axis is tangent to the radius of curvature of the loop. In this case,  $\alpha = 0$ , and  $C_{mq}$  is directly proportional to the net aerodynamic moment.

Other more complicated helical motions can be used for visualization. Two such motions require the cg of the vehicle to traverse a helical flight path. The first motion requires the vehicle's longitudinal axis to be oriented in the same direction as the center of rotation of the helix but displaced by a constant distance. A 2-D projection of this motion on the vertical plane is what is depicted in Figure 133 in the  $\dot{\alpha} \neq 0, q = 0$  case, where  $\alpha$  and  $\dot{\alpha}$  vary continuously, and produce moment components proportional to  $C_{m\alpha}$  and  $C_{m\dot{\alpha}}$ , respectively. This is referred to as "q = 0 helical motion" because the motion will not produce any moment proportional to  $C_{mq}$ .

For the second motion, the longitudinal axis of the vehicle remains tangent to the helical flight path at each point along the trajectory. A 2-D projection of this motion on the vertical plane is also depicted in Figure 133 in the  $\dot{\alpha} = 0, q \neq 0$  case, where moment components proportional to  $C_{mq}$  are produced. The  $\alpha$  of the incident air-stream is zero, since both the longitudinal axis of the vehicle and the free-stream velocity vector are tangent to the flight path. This motion is referred to as " $\dot{\alpha} = 0$  helical motion" because the motion will not produce any moment proportional to  $C_{m\dot{\alpha}}$ . [65]

The wind tunnel test conducted here involved the combined rate of change of  $\alpha$ , that is  $\dot{\alpha}$ , and the rate of change of the longitudinal attitude,  $q$  and normally it is not possible to separate the  $C_{m\dot{\alpha}}$  and the  $C_{mq}$  in wind tunnel test, so these coefficients were measured as a sum. Similarly, in the lateral-directional case, a rolling velocity ( $p$ ), or a yawing velocity ( $r$ ) about a fixed axis at an  $\alpha$  causes a simultaneous variation in the rate of change of the angle-of-sideslip ( $\dot{\beta}$ ). If a system of body axes is used, the resulting

composite expressions for the lateral-directional damping derivatives include a trigonometric function of  $\alpha$ , such as

$$(C_{n_r} - C_{n_{\dot{\beta}}} \cos \alpha) \quad \text{and} \quad (C_{\ell_p} - C_{\ell_{\dot{\beta}}} \sin \alpha)$$

For an axisymmetric vehicle,

$$C_{N_{\dot{\alpha}}} = C_{Y_{\dot{\beta}}} \quad \text{and} \quad C_{m_{\dot{\alpha}}} = C_{n_{\dot{\beta}}} \quad (43)$$

where  $C_{N_{\dot{\alpha}}} \equiv \frac{\partial C_N}{\partial \dot{\alpha}} =$  normal damping force coefficient

$C_{Y_{\dot{\beta}}} \equiv \frac{\partial C_Y}{\partial \dot{\beta}} =$  side damping force coefficient

$C_{m_{\dot{\alpha}}} \equiv \frac{\partial C_m}{\partial \dot{\alpha}} =$  pitch damping force coefficient

$C_{n_{\dot{\beta}}} \equiv \frac{\partial C_n}{\partial \dot{\beta}} =$  yaw damping force coefficient

## 8.2 Instrumentation

For the dynamic stability wind tunnel testing, the strain gauge balance used was a 1-component, internally mounted, MSFC #141. The dynamic balance was 11/16 inches in diameter by 2 1/2 inches in length, and measured the pitch deflection angle. The balance had an inner flexure beam with the gauges and an outer flexure to support the model. The mechanism was pneumatically activated with 500 psi air which caused a plunger to activate the protruding metal kicker which then deflected the model 2° nose down. The sting used was 7/8 inches in diameter at the model end and 11/16 inches in diameter at the balance end and had an overall length of 14 5/8 inches.

### 8.3 Fabrication of Test Articles

Figure 134 shows the dimensions of the new dynamic stability test model, which was also a 0.5% scale, as the one used for the static testing. A new model was fabricated of lightweight aluminum to facilitate the oscillation after the disturbance. For this test, the stainless steel model of the static stability wind tunnel test program was deemed too heavy. For a heavy model, the inertia forces will dominate the aerodynamic forces and

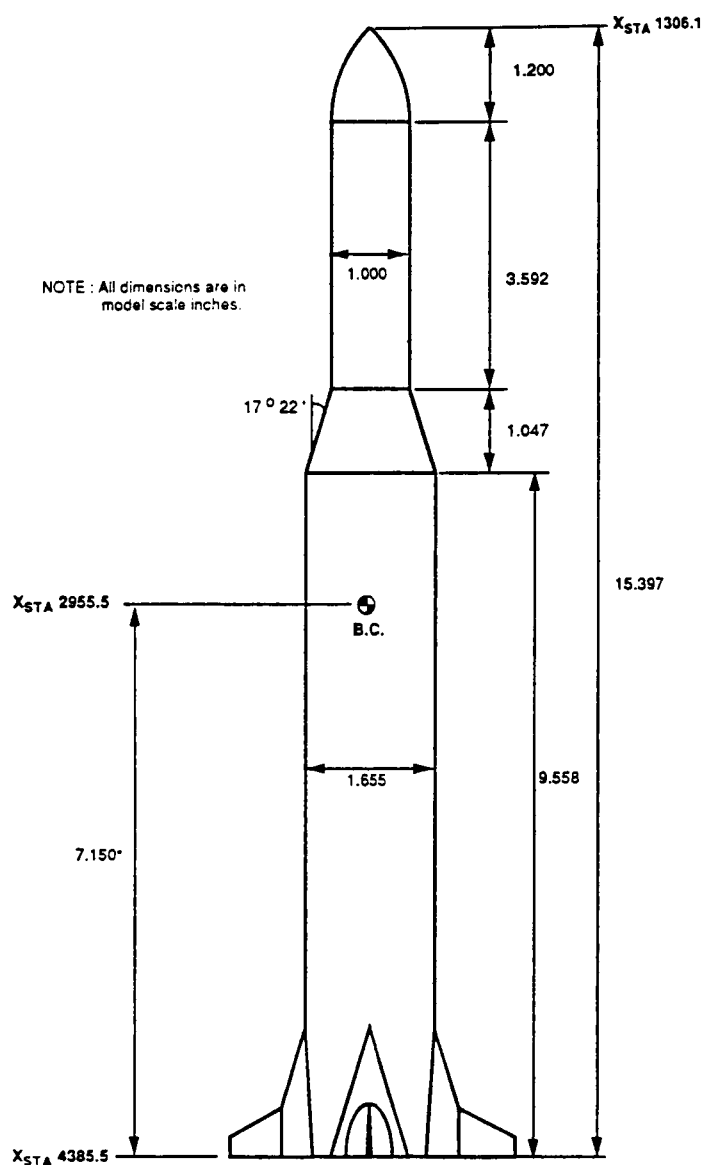


Figure 134. Dynamic stability wind tunnel model (0.005 scale).

make obtaining the aerodynamic damping more difficult. Figure 135 shows the drawing of the fabricated lightweight aluminum dynamic stability wind tunnel test model with the adapter assembly.

#### **8.4 Test Conditions**

Many wind tunnel techniques are available for dynamic stability testing. NASA LaRC has typically used forced oscillation of 0.5 to 1.5 Hz, but their wind tunnels are larger than the one used at MSFC for this test. To facilitate the logistics, a free oscillation with a mechanical release was chosen.

Free oscillation is the simplest of the oscillatory techniques and lends itself well to automatic computer processing. It is usually limited to measurement of direct damping derivatives, and involves the evaluation of a decaying oscillatory motion performed by an elastically suspended model following some initial disturbance. The initial disturbance may include a mechanical release from a displaced position, an application of a mechanical impulse, or an excitation at resonance and sudden interruption of the source of excitation. The decaying oscillations are recorded. From this data, the frequency and the damping, expressed as the log increment of amplitude, can be measured.

The dynamic stability wind tunnel test was conducted in the MSFC 14-inch trisonic wind tunnel described in Chapter VII. The MSFC free oscillation balanced #141 was used. The disturbance for this test was provided pneumatically, as described in the instrumentation section above. The dynamic stability test was conducted first with no FCA's on the experimental LV model, and then with FCA's 1, 6, and 11 for the Mach range of 0.8 to 2 and FCA deflections from 0° to 15°. The oscillation was 2° about  $\alpha = 0^\circ$ . Sample test data can be seen in Appendix C.4. Figure 136 shows the dynamic wind tunnel test model in the MSFC wind tunnel. Figure 137 shows the detail of the dynamic adapter assembly. Figure 138 shows the variable porosity walls used at the transonic speeds and described in Chapter VI.



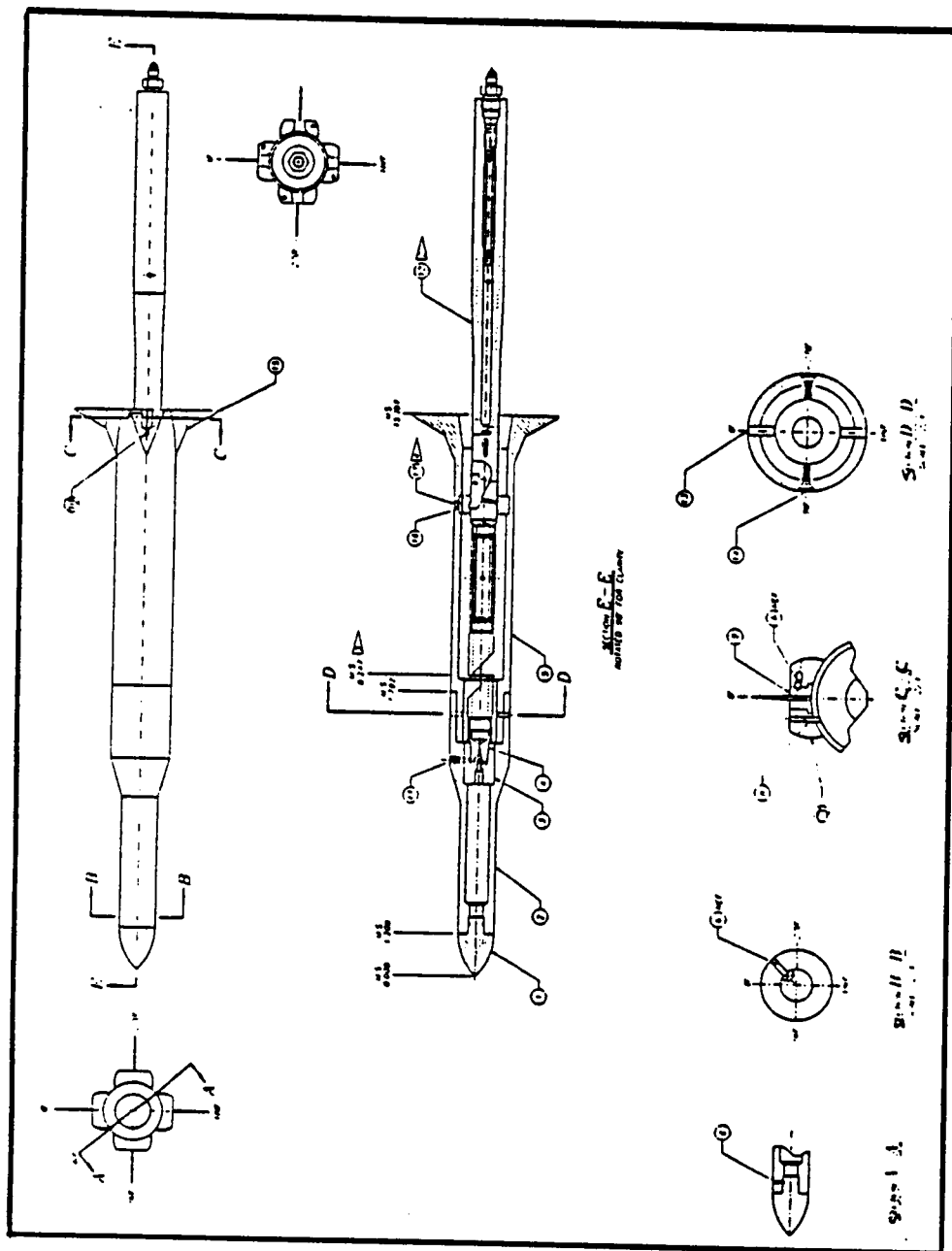


Figure 135. Drawing of dynamic stability wind tunnel test model with adapter assembly.

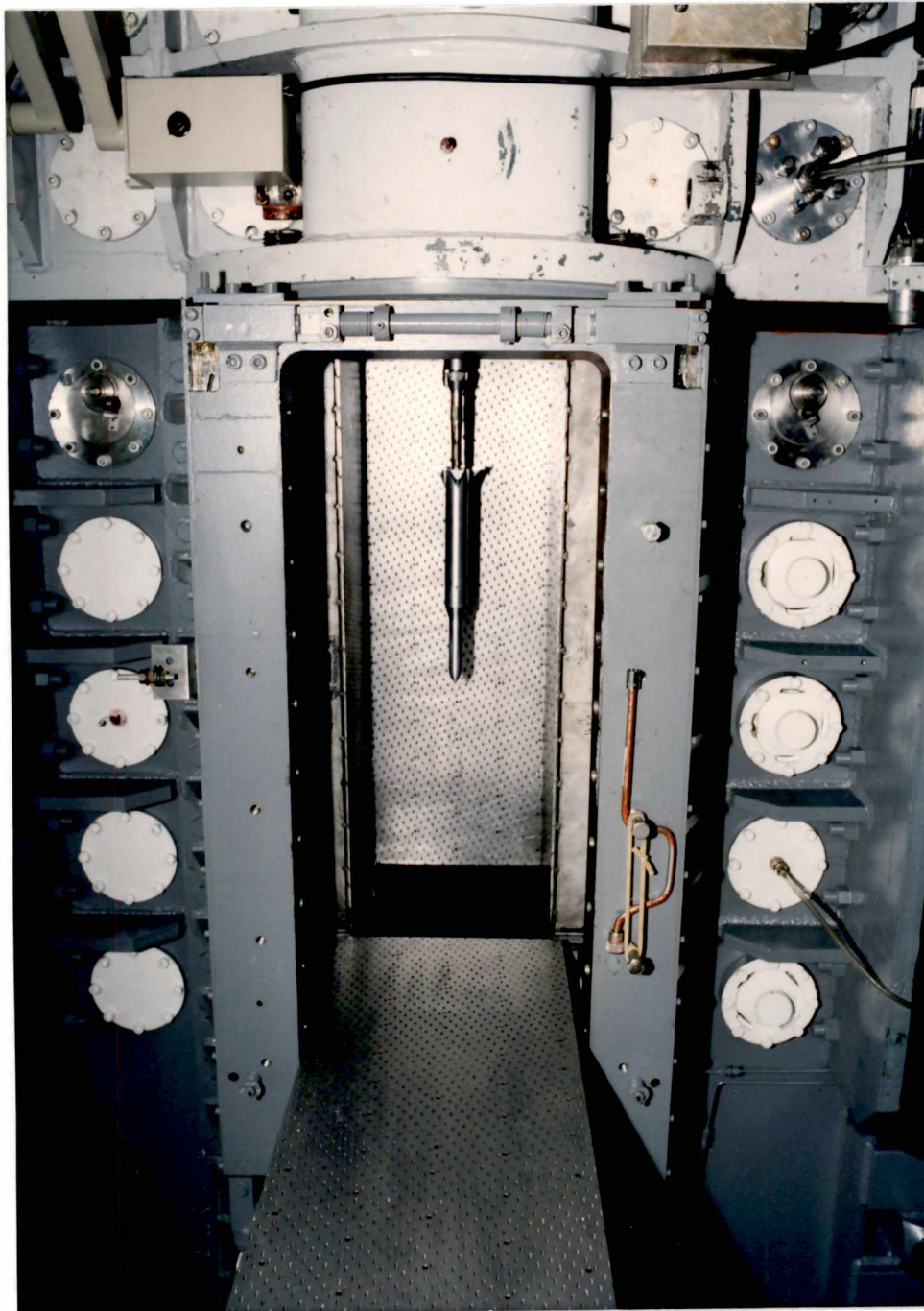


Figure 136. Dynamic wind tunnel test setup.

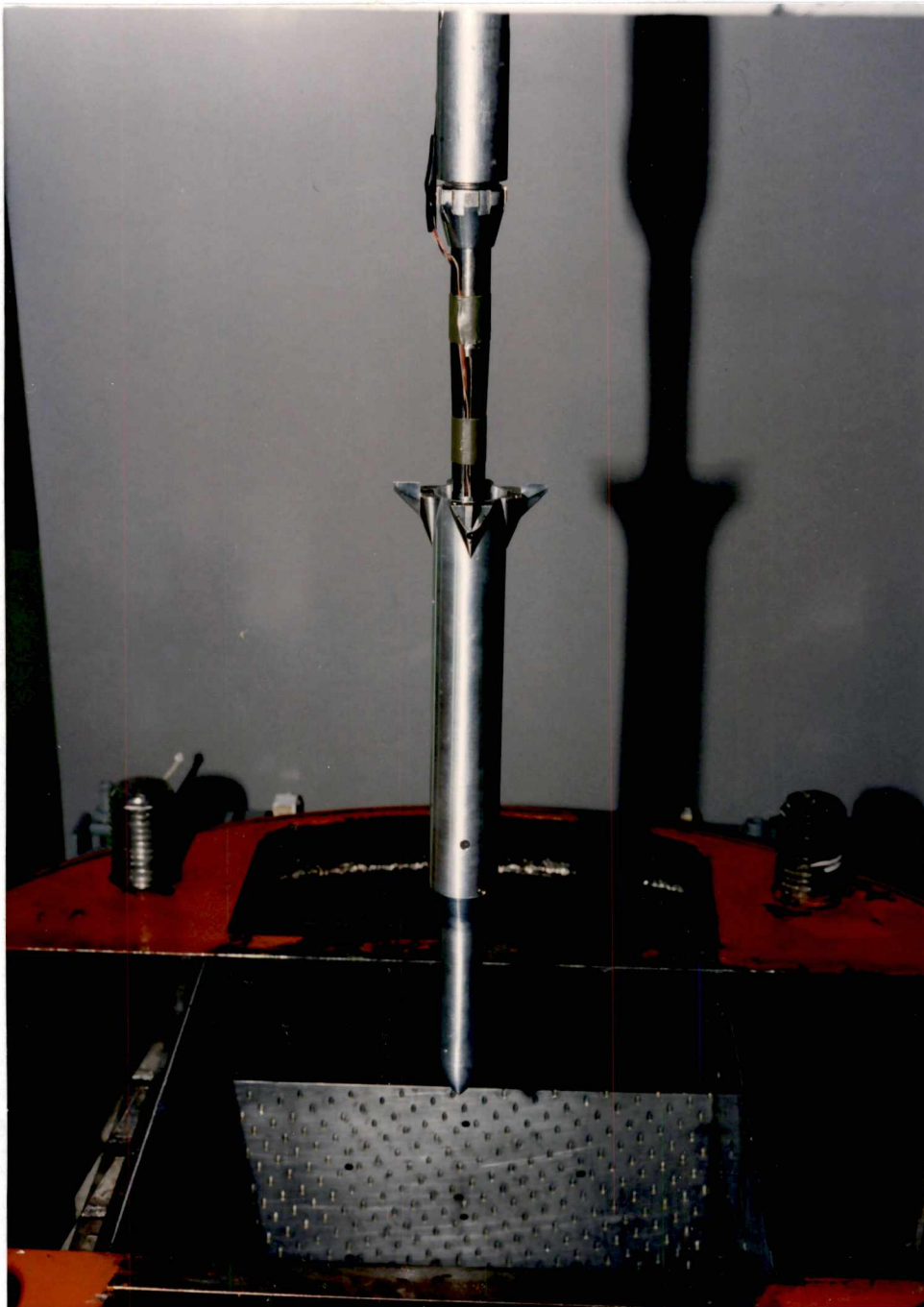


Figure 137. Dynamic wind tunnel test adapter assembly.

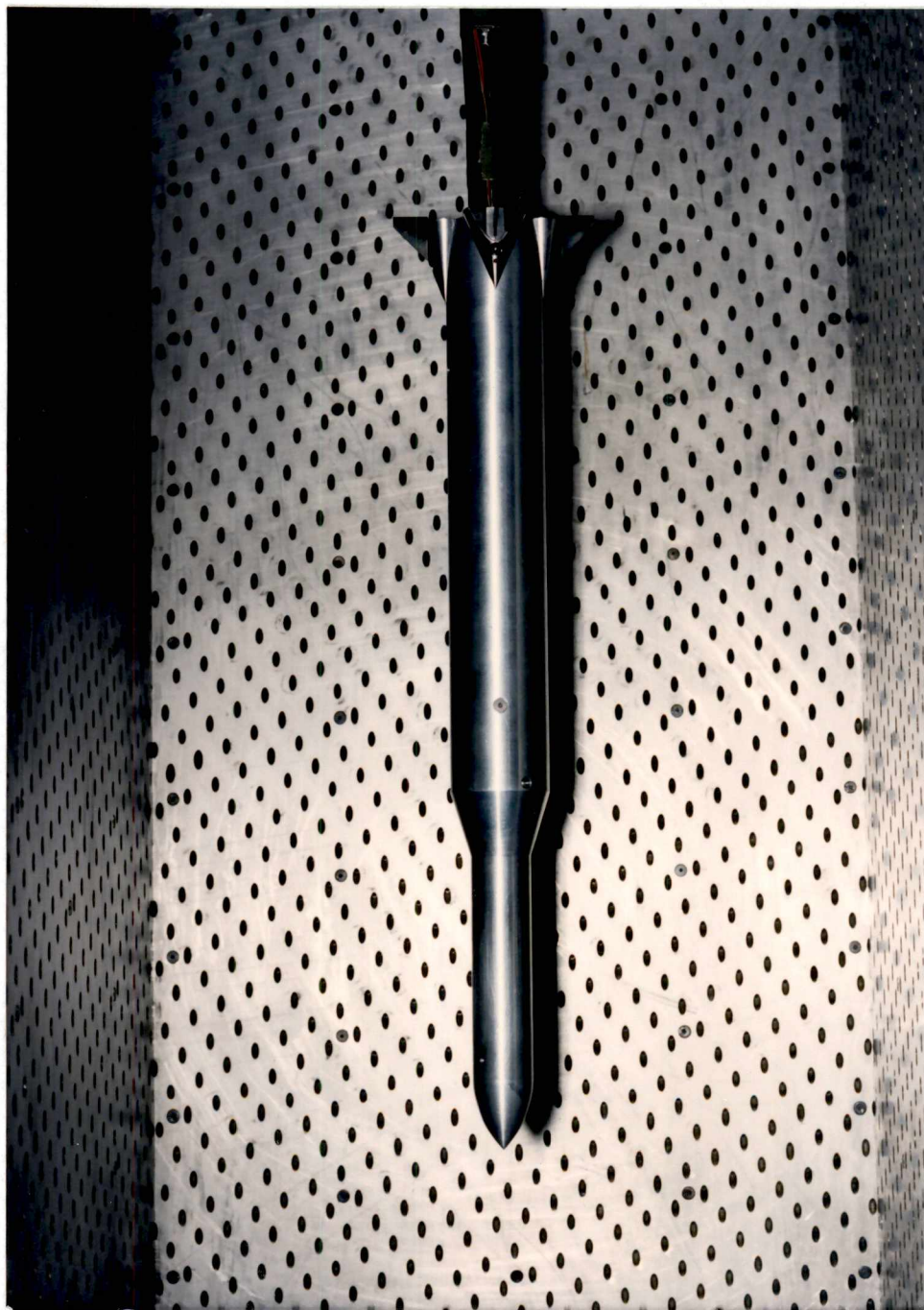


Figure 138. Variable porosity walls used in transonic range.

AERODYNAMIC FLIGHT CONTROL TO PROVIDE FLIGHT CONTROL  
AUGMENTATION FOR AFT CG FUTURE LAUNCH VEHICLES

A Dissertation  
Presented for the  
Doctor of Philosophy  
Degree  
The University of Tennessee, Knoxville

Christian Barret  
December 1996



## CHAPTER IX. DYNAMIC STABILITY ANALYSES

A LV can be considered dynamically stable if, after being disturbed from a nominal trajectory, the vehicle oscillates about the nominal condition with an amplitude of oscillation that decreases with time. A quantitative measure of dynamic stability or instability is the time it takes a disturbance motion to damp to half of its initial amplitude in a stable vehicle, or the time it takes for the initial amplitude to double in an unstable vehicle. Also of interest is the natural frequency and the period of oscillatory motion. The requirement for dynamic stability is typically expressed in terms of natural frequency ( $\omega_n$ ) and damping ratio ( $\zeta$ ) of a natural mode.

### 9.1 Rigid Body Equations of Motion

Using a body axes system fixed to the LV, and an inertial axes fixed to the Earth, the rigid body equations of motion can be obtained from Newton's second law. Figure 139 [66] shows the Earth-fixed frame of reference used here for the analysis of the experimental LV in its ascent trajectory. It has been shown in Reference 66 (from an acceleration expansion) that the Earth's rotation and curvature terms can be neglected in this development when  $M < 3$ . From lift-off to the maximum dynamic pressure point of the ascent trajectory of the experimental LV, the Mach number increases to 1.3, therefore,  $M < 3$  for the major atmospheric portion of the ascent.

The two vector equations of motion of the LV, equivalent to six scalar equations, are:

$$\sum \bar{F} = \frac{d}{dt} (\mu \bar{V}) ; \quad \text{and} \quad \sum \bar{M} = \frac{d}{dt} \bar{h} = \dot{\bar{h}} \Rightarrow (44)$$

$$F_x = \frac{d}{dt} (\mu u) ; \quad F_y = \frac{d}{dt} (\mu v) ; \quad F_z = \frac{d}{dt} (\mu w) \quad \text{and} \quad (45)$$

$$l = \frac{d}{dt} h_x ; \quad m = \frac{d}{dt} h_y ; \quad n = \frac{d}{dt} h_z$$

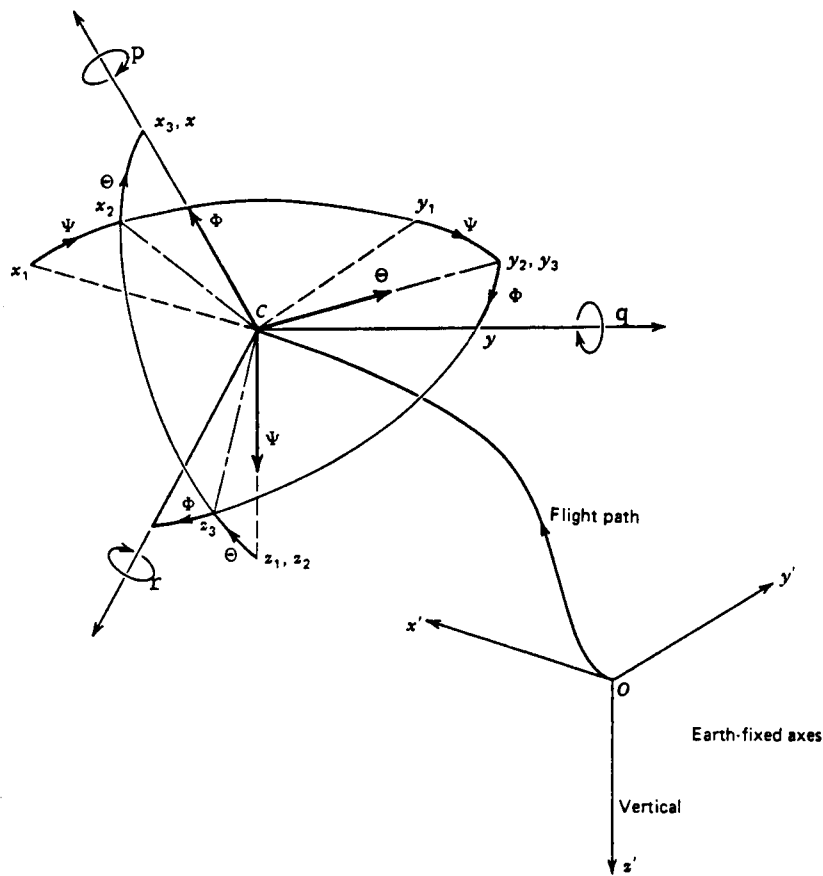


Figure 139. Earth-fixed reference frame.

where  $\bar{V} = (u, v, w)$  ;  $\bar{\omega} = (p, q, r)$  ;  $\bar{M} = (1, m, n)$  ;

$\mu = \text{mass}$  ;  $\bar{r} = (x, y, z)$  ;  $\bar{h} = (h_x, h_y, h_z)$

If  $\delta \mu$  is a differential element of mass and  $V_c$  is the velocity of the center of mass, then following traditional detailed derivations such as found in References 59, and 66 to 68:

$$\bar{h} = \sum (\bar{r} \times \bar{V}) \delta \mu \quad \text{and} \quad \bar{V} = \bar{V}_c + \bar{\omega} \times \bar{r} \Rightarrow$$

$$\begin{aligned}
\bar{h} &= \sum \bar{r} \times (\bar{V}_c + \bar{\omega} \times \bar{r}) \delta \mu = \sum \bar{r} \times (\bar{\omega} \times \bar{r}) \delta \mu = \\
&\bar{\omega} \sum r^2 \delta \mu - \sum \bar{r} (\bar{\omega} \cdot \bar{r}) \delta \mu = \\
&\bar{\omega} \sum (x^2 + y^2 + z^2) \delta \mu - \sum \bar{r} (p x + q y + r z) \delta \mu \quad (46)
\end{aligned}$$

From the expansion of

$$\sum \bar{r} \times (\bar{\omega} \times \bar{r}) \delta \mu = \sum \begin{vmatrix} i & j & k \\ x & y & z \\ (z q - y r) & (x r - z p) & (y p - x q) \end{vmatrix} \delta \mu \quad (47)$$

the scalar components of the angular momentum are

$$\begin{aligned}
h_x &= p \sum (y^2 + z^2) \delta \mu - q \sum x y \delta \mu - r \sum x z \delta \mu \\
h_y &= -p \sum x y \delta \mu + q \sum (x^2 + z^2) \delta \mu - r \sum y z \delta \mu \quad (48) \\
h_z &= -p \sum x z \delta \mu - q \sum y z \delta \mu + r \sum (x^2 + y^2) \delta \mu
\end{aligned}$$

where the summations are the moments and products of inertia, i.e.,

$$\begin{aligned}
h_x &= p I_x - q I_{xy} - r I_{xz} \\
h_y &= -p I_{xy} + q I_y - r I_{yz} \quad (49) \\
h_z &= -p I_{xz} - q I_{yz} + r I_z
\end{aligned}$$

In a rotating body frame,

$$\bar{F} = \mu \bar{V}_c + \mu \bar{\omega} \times \bar{V}_c \quad \text{and} \quad \bar{M} = \dot{\bar{h}} + \bar{\omega} \times \bar{h} \Rightarrow$$

$$F_x = \mu (\dot{u} + q w - r v) \quad ; \quad F_y = \mu (\dot{v} + r u - p w) \quad ;$$

$$F_z = \mu (\dot{w} + p v - q u)$$

and

$$l = \dot{h}_x + q h_z - r h_y; \quad m = \dot{h}_y + r h_x - p h_z; \quad n = \dot{h}_z + p h_y - q h_x \quad (50)$$



These are the Euler equations of motion, where the forces and moments acting on the LV consist of aerodynamic, propulsive, and gravitational components. Substituting for  $\bar{h}_i$  and  $\dot{\bar{h}}_i$  during a small time interval,

$$\begin{aligned} l &= \dot{p} I_x + q r (I_z - I_y) + (p r - \dot{q}) I_{xy} - (\dot{r} + p q) I_{xz} + (r^2 - q^2) I_{yz} \\ m &= p r (I_x - I_z) + \dot{q} I_y - (\dot{p} + q r) I_{xy} + (p^2 - r^2) I_{xz} + (p q - \dot{r}) I_{yz} \\ n &= p q (I_y - I_x) + \dot{r} I_z + (q^2 - p^2) I_{xy} + (q r - \dot{p}) I_{xz} - (\dot{q} + p r) I_{yz} \end{aligned} \quad (51)$$

For the experimental LV treated here, during atmospheric flight, the three cross products of inertia,  $I_{xy}$ ,  $I_{xz}$ ,  $I_{yz}$ , range from 0.24% to 1.2% of the smallest moment of inertia, as can be seen in Appendix A.2. Neglecting these, equations (51) become,

$$\begin{aligned} l &= \dot{p} I_x + q r (I_z - I_y) \quad ; \\ m &= p r (I_x - I_z) + \dot{q} I_y \quad ; \quad n = p q (I_y - I_x) = \dot{r} I_z \end{aligned} \quad (52)$$

Due to the axial symmetry of the experimental LV,  $I_z \approx I_y$  and the second term of the first equation can be neglected.

The equations of motion have been derived for a body-fixed axis system. The position and orientation of the LV can be conveniently expressed in terms of a fixed reference frame whose angular displacement from the body axes are the Euler angles. The LV is rotated through yaw, pitch, and roll angles in that order, known as the 3, 2, 1 sequence. (See Figure 140 [66]). It should be mentioned that in general there are 12 possible ways to define Euler angles. For other sequences and their applications see Hughes [69]; in addition, 24 sets of body and space rotation sequences are detailed in Kane [70].

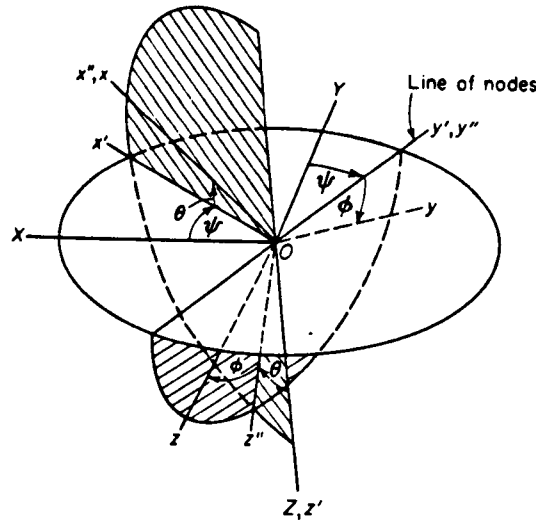


Figure 140. The Eulerian angles.

In terms of the Euler angles  $\psi$ ,  $\theta$ , and  $\phi$ ,

$$\begin{bmatrix} \dot{x} \\ \dot{y} \\ \dot{z} \end{bmatrix} = \begin{bmatrix} \cos \theta \cos \psi & (\sin \phi \sin \theta \cos \psi - \cos \phi \sin \psi) & (\cos \phi \sin \theta \cos \psi + \sin \phi \sin \psi) \\ \cos \theta \sin \psi & (\sin \phi \sin \theta \sin \psi + \cos \phi \cos \psi) & (\cos \phi \sin \theta \sin \psi - \sin \phi \cos \psi) \\ -\sin \theta & \sin \phi \cos \theta & \cos \phi \cos \theta \end{bmatrix} \begin{bmatrix} u \\ v \\ w \end{bmatrix} \quad (53)$$

The relationship between angular velocities in the body frame and Euler rates can be expressed as,

$$\begin{bmatrix} p \\ q \\ r \end{bmatrix} = \begin{bmatrix} 1 & 0 & -\sin \theta \\ 0 & \cos \phi & \cos \theta \sin \phi \\ 0 & -\sin \phi & \cos \theta \cos \phi \end{bmatrix} \begin{bmatrix} \dot{\phi} \\ \dot{\theta} \\ \dot{\psi} \end{bmatrix} \quad (54)$$

The force equations from equation (50) with inclusion of gravitational terms become,

$$\begin{aligned} X - \mu g \sin \theta &= \mu (\dot{u} + q w - r v) \\ Y + \mu g \cos \theta \sin \phi &= \mu (\dot{v} + r u - p w) \\ Z + \mu g \cos \theta \cos \phi &= \mu (\dot{w} + p v - q u) \end{aligned} \quad (55)$$

where  $\bar{F} = (X, Y, Z)$  = force due to aerodynamics and propulsion. [67]

Etkin, Nelson, et al. linearize the equations of motion by application of small disturbance theory about a steady flight condition for aircraft. For a LV in ascent, there is no unaccelerated flight. The equations of motion are not time invariant because the LV mass, moments and products of inertia, thrust, and the atmospheric density all vary with time for the LV. However, the ascent trajectory can be analyzed by small "time slices" where linearized equations do hold. The above equations can be linearized by assuming the LV motion consists of small deviations about a nominal ascent trajectory, that the mass remains constant for a small "time slice," and by using small-disturbance theory. All the variables in the equations are replaced by a reference value plus a perturbation or disturbance:

$$\begin{aligned}
 u &= u_0 + \Delta u, & v &= v_0 + \Delta v, & w &= w_0 + \Delta w \\
 p &= p_0 + \Delta p, & q &= q_0 + \Delta q, & r &= r_0 + \Delta r \\
 X &= X_0 + \Delta X, & Y &= Y_0 + \Delta Y, & Z &= Z_0 + \Delta Z \\
 m &= m_0 + \Delta m, & n &= n_0 + \Delta n, & l &= l_0 + \Delta l \\
 \delta_i &= \delta_{0i} + \Delta \delta_i
 \end{aligned} \tag{56}$$

where  $\delta_i$  are the deflection angles of the gimbaled engines and the FCA's.

The change in forces and moments then become,

$$\Delta X = \frac{\partial X}{\partial u} \Delta u + \frac{\partial X}{\partial w} \Delta w + \frac{\partial X}{\partial \delta_{FP}} \Delta \delta_{FP} + \frac{\partial X}{\partial \delta_{EP}} \Delta \delta_{EP}$$

$$\Delta Y = \frac{\partial Y}{\partial v} \Delta v + \frac{\partial Y}{\partial p} \Delta p + \frac{\partial Y}{\partial r} \Delta r + \frac{\partial Y}{\partial \delta_{FY}} \Delta \delta_{FY} + \frac{\partial Y}{\partial \delta_{EY}} \Delta \delta_{EY}$$

$$\Delta Z = \frac{\partial Z}{\partial u} \Delta u + \frac{\partial Z}{\partial w} \Delta w + \frac{\partial Z}{\partial \dot{w}} \Delta \dot{w} + \frac{\partial Z}{\partial q} \Delta q + \frac{\partial Z}{\partial \delta_{FP}} \Delta \delta_{FP} + \frac{\partial Z}{\partial \delta_{EP}} \Delta \delta_{EP}$$

$$\begin{aligned}
\Delta l &= \frac{\partial l}{\partial v} \Delta v + \frac{\partial l}{\partial p} \Delta p + \frac{\partial l}{\partial r} \Delta r + \frac{\partial l}{\partial \delta_{FR}} \Delta \delta_{FR} + \frac{\partial l}{\partial \delta_{ER}} \Delta \delta_{ER} \\
\Delta m &= \frac{\partial m}{\partial u} \Delta u + \frac{\partial m}{\partial w} \Delta w + \frac{\partial m}{\partial \dot{w}} \Delta \dot{w} + \frac{\partial m}{\partial q} \Delta q + \frac{\partial m}{\partial \delta_{FP}} \Delta \delta_{FP} + \frac{\partial m}{\partial \delta_{EP}} \Delta \delta_{EP} \\
\Delta n &= \frac{\partial n}{\partial v} \Delta v + \frac{\partial n}{\partial p} \Delta p + \frac{\partial n}{\partial r} \Delta r + \frac{\partial n}{\partial \delta_{FY}} \Delta \delta_{FY} + \frac{\partial n}{\partial \delta_{EY}} \Delta \delta_{EY}
\end{aligned} \tag{57}$$

where

- $\delta_{FP}$  = pitch deflection of FCA
- $\delta_{FR}$  = roll deflection of FCA
- $\delta_{EY}$  = yaw deflection of engine
- $\delta_{FY}$  = yaw deflection of FCA
- $\delta_{EP}$  = pitch deflection of engine

The linearized longitudinal and lateral rigid body equations then become,

#### Longitudinal

$$\begin{aligned}
(d/dt - X_u) \Delta u - X_w \Delta w + (g \cos \theta_0) \Delta \theta &= X_{\delta_{FP}} \Delta \delta_{FP} + X_{\delta_{EP}} \Delta \delta_{EP} \\
-Z_u \Delta u + ((1 - Z_{\dot{w}}) d/dt - Z_w) \Delta w - ((u_0 + Z_q) d/dt - g \sin \theta_0) \Delta \theta &= Z_{\delta_{FP}} \Delta \delta_{FP} + Z_{\delta_{EP}} \Delta \delta_{EP} \\
-m_u \Delta u - (m_{\dot{w}} d/dt + m_w) \Delta w + (d^2/dt^2 - m_q d/dt) \Delta \theta &= m_{\delta_{FP}} \Delta \delta_{FP} + m_{\delta_{EP}} \Delta \delta_{EP}
\end{aligned} \tag{58}$$

#### Lateral-Directional

$$\begin{aligned}
(d/dt - Y_v) \Delta v - Y_p \Delta p + (u_0 - Y_r) \Delta r - (g \cos \theta_0) \Delta \phi &= Y_{\delta_{FY}} \Delta \delta_{FY} + Y_{\delta_{EY}} \Delta \delta_{EY} \\
-l_v \Delta v + (d/dt - l_p) \Delta p - ((I_{xz}/I_x) (d/dt) + l_r) \Delta r &= l_{\delta_{FR}} \Delta \delta_{FR} + l_{\delta_{ER}} \Delta \delta_{ER} \\
-n_v \Delta v - ((I_{xz}/I_z) (d/dt) + n_p) \Delta p + (d/dt - n_r) \Delta r &= n_{\delta_{FY}} \Delta \delta_{FY} + n_{\delta_{EY}} \Delta \delta_{EY}
\end{aligned} \tag{59}$$

where

$$X_u \equiv \frac{\partial x}{\partial u} / \mu \quad ; \quad X_w = \frac{\partial X}{\partial w} / \mu \quad ; \quad \text{etc.}$$

The longitudinal equations are comprised of the X-force, Z-force, and pitching moment equations. The lateral equations are comprised of the Y-force, rolling moment, and yawing moment equations.

The transfer function (TF) gives the relationship between the motion variables and the control input. The longitudinal TF's are obtained from the equations of motion by assuming  $\theta_o = 0 \Rightarrow \cos \theta_o = 1$  ;  $\sin \theta_o = 0 = Z_q = Z_{\dot{w}}$ . [67] This yields,

$$\begin{aligned} (d/dt - X_u) \Delta u - X_w \Delta w + g \Delta \theta &= X_\delta \Delta \delta \\ - Z_u \Delta u + (d/dt - Z_w) \Delta w - u_o (d/dt) \Delta \theta &= Z_\delta \Delta \delta \\ - m_u \Delta u - (m_{\dot{w}} (d/dt) + m_w) \Delta w + d/dt (d/dt - m_q) \Delta \theta &= m_\delta \Delta \delta \end{aligned} \quad (60)$$

Taking the Laplace transformation of the equations and then dividing by the control deflection gives,

$$\begin{aligned} (s - X_u) \frac{\Delta u}{\Delta \delta} - X_w \frac{\Delta w}{\Delta \delta} + g \frac{\Delta \theta}{\Delta \delta} &= X_\delta \\ - Z_u \frac{\Delta u}{\Delta \delta} + (s - Z_w) \frac{\Delta w}{\Delta \delta} - u_o s \frac{\Delta \theta}{\Delta \delta} &= Z_\delta \\ - m_u \frac{\Delta u}{\Delta \delta} - (m_{\dot{w}} s + m_w) \frac{\Delta w}{\Delta \delta} + s(s - m_q) \frac{\Delta \theta}{\Delta \delta} &= m_\delta \end{aligned} \quad (61)$$

Using Cramer's rule,

$$\frac{\Delta \theta}{\Delta \delta} = \frac{\begin{vmatrix} s - X_u & -X_w & X_\delta \\ -Z_u & s - Z_w & Z_\delta \\ -m_u & -(m_w s + m_w) & m_\delta \end{vmatrix}}{\begin{vmatrix} s - X_u & -X_w & g \\ -Z_u & s - Z_w & -u_0 s \\ -m_u & -(m_w s + m_w) & s(s - m_q) \end{vmatrix}} \quad (62)$$

and  $\Delta u / \Delta \delta$  and  $\Delta w / \Delta \delta$  are found similarly. By expanding the determinants, the TF's can be expressed as a ratio of two polynomials. For example,

$$\begin{aligned} \frac{\Delta u}{\Delta \delta} &= \frac{N_\delta^u(s)}{\Delta_{\text{long}}} = \frac{A_u s^3 + B_u s^2 + C_u s + D_u}{\Delta_{\text{long}}} \\ \frac{\Delta w}{\Delta \delta} &= \frac{N_\delta^w(s)}{\Delta_{\text{long}}} = \frac{A_w s^3 + B_w s^2 + C_w s + D_w}{\Delta_{\text{long}}} \\ \frac{\Delta \theta}{\Delta \delta} &= \frac{N_\delta^\theta(s)}{\Delta_{\text{long}}} = \frac{A_\theta s^2 + B_\theta s + C_\theta}{\Delta_{\text{long}}} \end{aligned} \quad (63)$$

where  $\Delta_{\text{long}} = A s^4 + B s^3 + C s^2 + D s + E$ .

The longitudinal control TF's, and the lateral TF's which are handled in the same manner, are presented in Table 11. [67]

The concept of added mass (or apparent mass) is mentioned here, since it is sometimes used in the theoretical determination of stability and control derivatives. When a body accelerates in a fluid, not only the body is accelerated, but also the surrounding fluid.

$$F > \mu a \quad ; \quad F = (\mu + \mu') a \quad (64)$$

Table 11. Longitudinal and lateral control transfer function.

| Longitudinal control transfer functions |                                                 |                                                                                                                                      |                                                                                                                                                                                        |                        |
|-----------------------------------------|-------------------------------------------------|--------------------------------------------------------------------------------------------------------------------------------------|----------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------|------------------------|
| A                                       | B                                               | C                                                                                                                                    | D                                                                                                                                                                                      | E                      |
| $\Delta_{long}$                         | 1                                               | $-Z_w M_q - u_0 M_{\dot{q}} - Z_w X_w$                                                                                               | $-X_w(Z_w M_q - u_0 M_{\dot{q}}) + Z_w(X_w M_q + g M_w) - M_w(u_0 X_w - g)$                                                                                                            | $g(Z_w M_w - M_w Z_w)$ |
| $N_q^w$                                 | $M_q + Z_w M_{\dot{q}}$                         | $X_q(Z_w M_q + M_{\dot{q}}) + Z_q(M_w X_w - M_q X_w) - M_q(X_w + Z_w)$                                                               | $X_q(u_0 M_w - Z_w M_q) + Z_q X_w M_q - u_0 M_q X_w$                                                                                                                                   | $g(Z_w M_w - M_q Z_w)$ |
| $N_z^w$                                 | $Z_q$                                           | $X_q Z_w - Z_q(X_w + M_q) + M_q u_0$                                                                                                 | $X_q(Z_w M_q - u_0 M_w) - Z_q(X_w M_q + g M_w) + M_q(u_0 X_w - g)$                                                                                                                     | $g(M_q Z_w - Z_q M_w)$ |
| $N_x^w$                                 | $X_q$                                           | $-X_q(Z_w + M_q + u_0 M_w) + Z_q X_w$                                                                                                |                                                                                                                                                                                        |                        |
| Lateral control transfer functions      |                                                 |                                                                                                                                      |                                                                                                                                                                                        |                        |
| A                                       | B                                               | C                                                                                                                                    | D                                                                                                                                                                                      | E                      |
| $\Delta$                                | $1 - \frac{l_{11}^2}{l_r l_i}$                  | $-Y_v \left(1 - \frac{l_{11}^2}{l_r l_i}\right) - L_p - N_r$                                                                         | $u_0 N_v + L_p(Y_v + N_r) + N_p \left(\frac{l_{11}}{l_r} Y_v - L_r\right) + Y_v \left(\frac{l_{11}}{l_r} L_r + N_r\right) + u_0 N_p L_v - g \left(L_v + \frac{l_{11}}{l_r} N_v\right)$ | $g(L_v N_r - N_v L_r)$ |
| $N_r^w$                                 | $Y_v \left(1 - \frac{l_{11}^2}{l_r l_i}\right)$ | $-Y_v \left[L_r + N_r + \frac{l_{11}}{l_r} N_p + \frac{l_{11}}{l_r} L_r\right] \times u_0 \left(\frac{l_{11}}{l_r} L_v + N_v\right)$ | $Y_v(L_p N_r - N_r L_r) + u_0(N_v L_p - L_v N_r) + g \left(L_v + \frac{l_{11}}{l_r} N_p\right)$                                                                                        |                        |
| $N_z^w$                                 | $L_v + \frac{l_{11}}{l_r} N_v$                  | $Y_v \left(L_v + \frac{l_{11}}{l_r} N_v\right) - L_v(N_r + Y_v) + N_v \left(L_r - \frac{l_{11}}{l_r} Y_v\right)$                     | $Y_v(L_v N_v - L_v N_r) + L_v(Y_v N_r + u_0 N_v) - N_v(u_0 L_v + Y_v L_r)$                                                                                                             |                        |
| $N_x^w$                                 | $N_v + \frac{l_{11}}{l_r} L_v$                  | $Y_v \left(N_v + \frac{l_{11}}{l_r} L_v\right) + L_v \left(N_v - \frac{l_{11}}{l_r} Y_v\right) - N_v(Y_v + L_r)$                     | $Y_v(L_v N_r - N_r L_r) - L_v Y_v N_r + N_v Y_v L_r$                                                                                                                                   | $g(L_v N_v - N_v L_v)$ |

where  $\mu'$  is the added mass and  $(\mu + \mu')$  is the virtual mass. Added masses are dependent only on the shape and orientation of a body and on the mode of motion, not on its linear or angular velocity or acceleration. The Air Force Stability and Control DATCOM [45] can be used to calculate apparent mass factors for control surfaces in the presence of the fuselage and in the presence of each other. Nielsen [64] presents a method by which stability derivatives can be calculated using apparent mass coefficients, if they are known for the vehicle. Blakelock [71] points out that the compressibility effects and the apparent-mass effects are taken into account by the use of stability derivatives that are wind tunnel test determined at specific Mach numbers. In the theoretical treatment, quasi-steady flow is usually assumed in flight mechanics stability derivative derivations. The result of this assumption is that  $\dot{u}$  and  $\dot{v}$  are normally neglected and the rates of change with respect to  $\dot{w}$  are retained to account for downwash or upwash effects.

## 9.2 Pitch Damping Assessment

When the LV model is constrained to a pure pitching motion as in the dynamic stability wind tunnel tests, the equations of motion can be developed from the rigid body equations of the section above,

$$\sum m_{cg} = I_y \ddot{\theta} \Rightarrow \Delta m = I_y \Delta \ddot{\theta} \quad (65)$$

where

$$\begin{aligned} \Delta m &= f(\Delta \alpha, \Delta \dot{\alpha}, \Delta q, \Delta \delta_F) \quad \text{and} \quad \alpha \equiv \tan^{-1}(w/u) ; \\ \dot{\alpha} &= (u\dot{w} - \dot{u}w) / (u^2 + w^2) ; \quad (\beta \equiv \sin^{-1} v / \bar{V} ; \quad \bar{V} = (u, v, w)) \end{aligned} \quad (66)$$

This can be expanded in terms of perturbation variables by a Taylor series:

$$\Delta m = \frac{\partial m}{\partial \alpha} \Delta \alpha + \frac{\partial m}{\partial \dot{\alpha}} \Delta \dot{\alpha} + \frac{\partial m}{\partial q} \Delta q + \frac{\partial m}{\partial \delta_F} \Delta \delta_F \quad (67)$$



The  $m_q$  and  $m_{\dot{\alpha}}$  each oppose the LV disturbing motion and thus are considered damping terms. If the body and the fixed frames are aligned so they coincide at  $t = 0$ ,

$$\Delta \alpha = \Delta \theta \quad \text{and} \quad \Delta \dot{\theta} = \Delta q = \Delta \dot{\alpha}$$

This is true only for the special case where the cg is constrained. Substituting this into equation (65),

$$\Delta \ddot{\alpha} - (m_q + m_{\dot{\alpha}}) \Delta \dot{\alpha} - m_{\alpha} \Delta \alpha = m_{\delta_F} \Delta \delta_F \quad (68)$$

where

$$m_q = \frac{\partial m}{\partial q} / I_y, \quad m_{\dot{\alpha}} = \frac{\partial m}{\partial \dot{\alpha}} / I_y, \quad \text{etc.}$$

This nonhomogeneous, second-order, differential equation has the characteristic equation,

$$\lambda^2 - (m_q + m_{\dot{\alpha}}) \lambda - m_{\alpha} = 0 \quad (69)$$

which when compared to a standard equation of a second-order system

$$\begin{aligned} \lambda^2 + 2 \zeta \omega_n \lambda + \omega_n^2 &= 0 \quad \Rightarrow \\ \lambda_{1,2} &= -\zeta \omega_n \pm i \omega_n \sqrt{1 - \zeta^2}; \quad \omega = \omega_n \sqrt{1 - \zeta^2} \end{aligned} \quad (70)$$

This yields

$$\omega_n = \sqrt{-m_{\alpha}}; \quad \zeta = \frac{-(m_q + m_{\dot{\alpha}})}{2 \sqrt{-m_{\alpha}}} \quad (71)$$

where  $\zeta$  is the damping ratio and  $\omega_n$  is the natural frequency. The frequency is related to the LV's static longitudinal stability, and the damping ratio is a function of the aerodynamic damping and the static stability.

$$\lambda_{1,2} = 1/2 (m_q + m_{\dot{\alpha}}) \pm i \sqrt{-4 m_{\alpha} - (m_q + m_{\dot{\alpha}})^2} \quad (72)$$

The general solution to eq. (68) for a step change  $\Delta \delta_F$  in the FCA angle can be expressed as

$$\Delta \alpha(t) = \Delta \alpha_{\text{trim}} \left[ 1 - \frac{e^{-\zeta \omega_n t}}{\sqrt{1 - \zeta^2}} \sin(\omega_n t \sqrt{1 - \zeta^2} + \tilde{\phi}) \right] \quad (73)$$

where

$$\Delta \alpha_{\text{trim}} = \text{change in } \alpha_{\text{trim}} = - (m_{\delta_F} \Delta \delta_F) / m_{\alpha}$$

$$\tilde{\phi} = \text{phase angle} = \tan^{-1} \sqrt{1 - \zeta^2} / \zeta.$$

The solution is a damped sinusoid with the frequency being a function of  $C_{m_{\alpha}}$  and the damping ratio a function of  $(C_{m_q} + C_{m_{\dot{\alpha}}})$  and  $C_{m_{\alpha}}$ . Figure 141 [67] shows the  $\alpha$  time history of a pitching model in wind tunnel test for various damping ratios.

The period of oscillation is a function of the imaginary part of the root, while the time to half amplitude or double amplitude is a function of the real part of the root.

$$P = \frac{2\pi}{\omega} \quad ; \quad t_{\text{double}} \text{ or } t_{\text{half}} = \frac{\ln 2}{|1 - \zeta \omega_n|} \quad (74)$$

The rate of growth or decay of the oscillation is determined by the sign of the real part of the complex root. A negative real part produces a decaying oscillation, whereas a positive real part causes the motion to grow. The time to half or double the initial amplitude of the

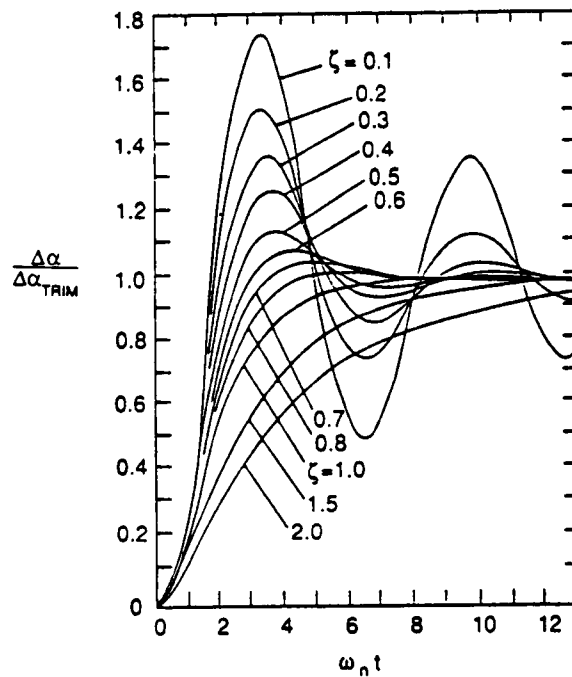


Figure 141. Time history of pitching model as a function of damping ratios.

disturbance is a measure of the rate of growth or decay. The number of cycles for halving or doubling the amplitude is a function of the imaginary part divided by the real part. [67]

$$N_{\text{double or half(cycles)}} = 0.110 \frac{|\omega|}{|-\zeta \omega_n|} \quad (75)$$

### 9.3 Longitudinal Short Period Mode Assessment

The longitudinal motion of a LV disturbed from its nominal ascent trajectory is characterized by two oscillatory modes, the phugoid and the short period modes. The phugoid mode of a stable vehicle is lightly damped, has a long period, and is characterized by changes in pitch attitude at a nearly constant  $\alpha$ . The short period mode is heavily damped in a stable vehicle, has a very short period, and is characterized by change in pitch rate at small  $\Delta u$ .

For a LV in ascent, the phugoid mode is not very important. It can be shown [67] that if compressibility effects are neglected, the phugoid natural frequency and damping ratio can be approximated by

$$\omega_n = \sqrt{2} \frac{g}{u_o} ; \quad \zeta = \frac{1}{\sqrt{2} (L/D)} \quad (76)$$

Thus, the frequency of oscillation and the damping ratio are inversely proportional to the ascent velocity and the vehicle  $L/D$  (lift/drag) ratio, respectively. Since the  $L/D$  for a LV is poor, the phugoid mode is well damped, and since the ascent velocity starts from zero and slowly increases, the phugoid frequency is high. In addition, as the velocity increases, the period of the phugoid increases.

If only the short period mode is considered, the full set of longitudinal state equations can be reduced [67] by assuming  $\Delta u = 0$ , dropping the X-force equation, and using

$$\Delta \alpha = \frac{\Delta w}{u_o} ; \quad m_\alpha = u_o m_w ; \quad m_{\dot{\alpha}} = u_o m_{\dot{w}} ; \quad Z_\alpha = u_o Z_w \quad (77)$$

Solving the resulting short period mode characteristic equation,

$$\lambda_{1,2} = \left[ \left( m_q + m_{\dot{\alpha}} + \frac{Z_\alpha}{u_o} \right) / 2 \right] \pm \left[ \left( m_q + m_{\dot{\alpha}} + \frac{Z_\alpha}{u_o} \right)^2 - 4 \left( m_q \frac{Z_\alpha}{u_o} - m_\alpha \right) \right]^{1/2} / 2$$

$$\omega_n = \left( m_q \frac{Z_\alpha}{u_o} - m_\alpha \right)^{1/2} ; \quad \zeta = - \left( m_q + m_{\dot{\alpha}} + \frac{Z_\alpha}{u_o} \right) / 2 \omega_n \quad (78)$$

The short period mode is by far the most important for the experimental LV. If this mode has a high  $\omega_n$  and is heavily damped, then the LV will respond rapidly to the FCA input without any undesirable overshoot. As the sum  $(m_q + m_{\dot{\alpha}})$  increases, the pitch damping increases. As the static stability increases, the vehicle natural frequency

increases. As the cg is moved rearward, the longitudinal modes become aperiodic and eventually unstable. Figure 142 [67] shows the longitudinal response as a function of cg position and FCA planform area.

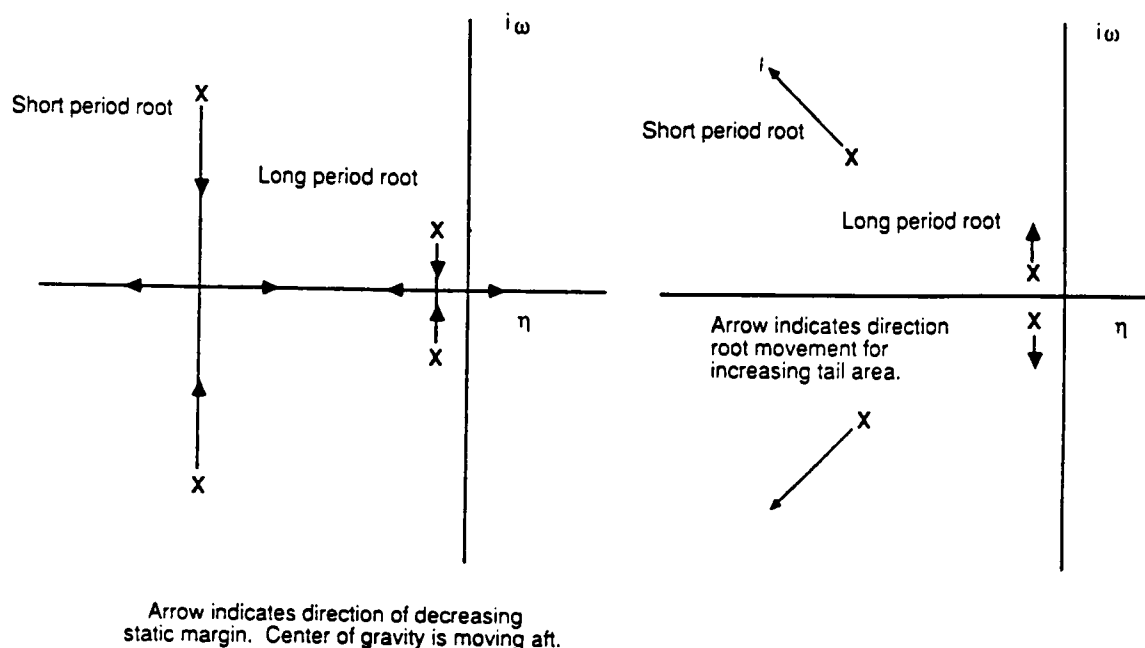


Figure 142. Longitudinal response as a function of cg position and FCA planform size.

#### 9.4 Dynamic Stability Results

After wind tunnel test, the semiempirical computer codes are useful for comparison, and for interpolation of data. The U.S. Air Force Missile DATCOM simulation program [72–79] is a semiempirical flight stability and control prediction code, developed by the Flight Dynamics Directorate of the Wright-Patterson Air Force Base in Ohio, and tailored to conceptual and preliminary design applications. The version used here is the latest as of this writing, the June 1993 revision. The program uses a component build-up approach based on the equivalent  $\alpha$  concept (also called total  $\alpha$  in the literature). The body-axis  $\alpha$  and  $\beta$  are transformed to a total  $\alpha$  and roll angle through the following relations [73]

$$\alpha = \tan^{-1} [ (\tan \alpha_b)^2 + (\tan \beta_b)^2 ]^{1/2}$$

$$\phi = \tan^{-1} [ \tan \beta_b / \tan \alpha_b ]$$
(79)

All internal calculations of the Missile DATCOM are performed in the equivalent or total  $\alpha$  (aeroballistic system), and simple vortex models are employed to allow for rapid and economical analyses. See Figures 143 and 144 [73], and Reference 73 for a more detailed explanation.

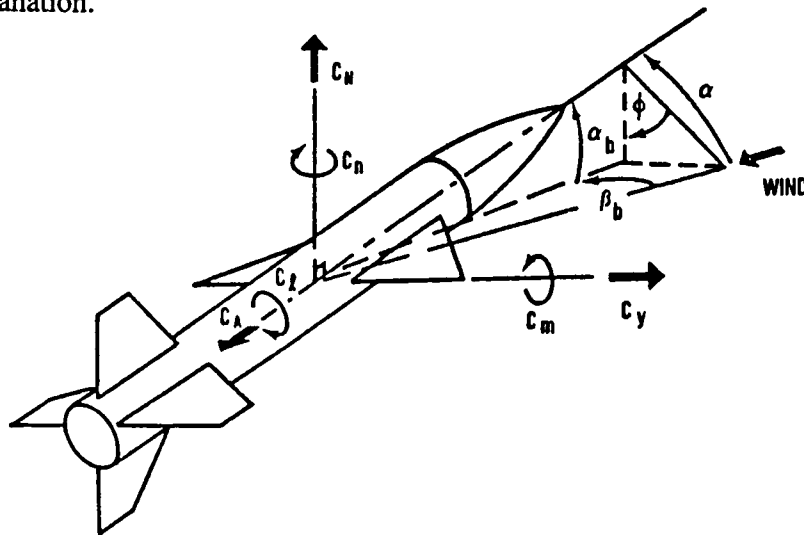


Figure 143. Aeroballistic axis system.

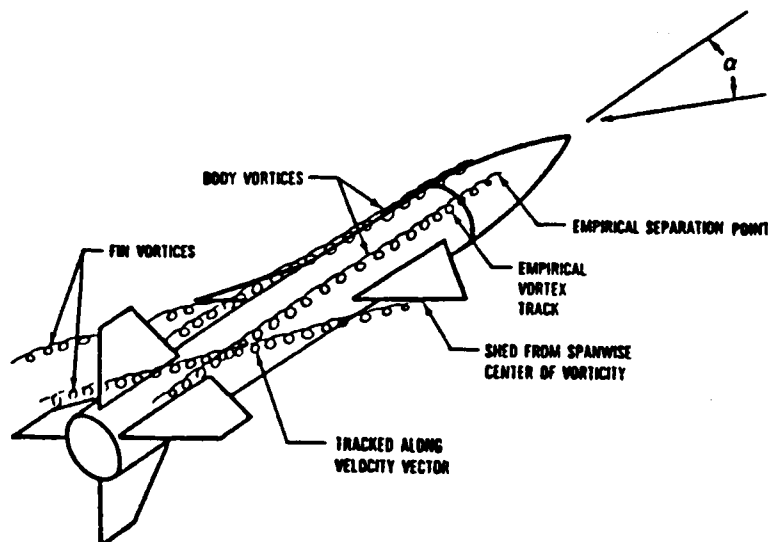


Figure 144. Missile DATCOM vortex model.

Current semiempirical computer codes, applicable to the experimental LV of this research, are discussed in References 54 and 74 to 77. Of the 27 empirical and semiempirical codes for missiles that were surveyed in Reference 54 and shown in Table 12, Missile DATCOM is one of the most widely used, and is one of the few which predicts the dynamic stability derivatives. White [77] states in a 1995 AIAA Conference

Table 12. Capabilities of empirical and semiempirical codes for missiles.

| CODE NAME            | MACH RANGE            | INCIDENCE RANGE                      | ROLL RANGE<br>( $\phi = 0^\circ$ for 4 configuration) | FINS    |                | CONTROL DEFLECTIONS<br>(A1, A2, A3, A4) | AERODYNAMIC COEFFICIENTS                                        |                                            |                                                |
|----------------------|-----------------------|--------------------------------------|-------------------------------------------------------|---------|----------------|-----------------------------------------|-----------------------------------------------------------------|--------------------------------------------|------------------------------------------------|
|                      |                       |                                      |                                                       | IN-LINE | INTERDIGITATED |                                         | AXIAL FORCE<br>CA TOTAL<br>CAW WAVE<br>CAI FRICTION<br>Cdb BASE | STATIC STABILITY<br>CN, Cm<br>CY, Cn<br>CI | DYNAMIC DERIVATIVES<br>Cmq, Cmr<br>Cnr, Cnp... |
| ABACUS               | $M \leq 5.0$          | $90^\circ \leq \alpha \leq 90^\circ$ | $0^\circ \leq \phi \leq 360^\circ$                    |         |                | PITCH/YAW                               | /                                                               |                                            |                                                |
| AERAM                | $1.2 \leq M \leq 5$   | $\alpha \leq 30^\circ$               | $\phi = 0^\circ, 45^\circ$                            |         | /              | $A \leq 40^\circ$                       |                                                                 | CN, Cm                                     | Cmq, Cmr                                       |
| AM300                | $M \geq 3.0$          | $0 \leq \alpha \leq 180^\circ$       | $\phi = 0^\circ$                                      |         | /              | $A \leq 15^\circ$                       | /                                                               | CN, Cm                                     | /                                              |
| BAKER                | $0.6 \leq M \leq 3.0$ | $0 \leq \alpha \leq 180^\circ$       | $\phi = 0^\circ$                                      | /       | /              | /                                       | /                                                               | CN, Cm                                     | /                                              |
| BDV/NCV              | $4 \leq M \leq 12$    | $\alpha \leq 2^\circ$                | $\phi = 0^\circ$                                      | /       | /              | /                                       |                                                                 |                                            | /                                              |
| CARENG               | $M \leq 4$            | $\alpha \leq 20^\circ$               | $\phi = 0^\circ$                                      |         | /              | /                                       | /                                                               | CN, Cm                                     | /                                              |
| CASAERO              | $M \leq 4.0$          | $\alpha \leq 30^\circ$               | $\phi = 0^\circ$                                      |         |                | $A \leq 25^\circ$                       | /                                                               | CN, Cm                                     |                                                |
| DOPRAM               | $M \leq 4.0$          | $\alpha \leq 90^\circ$               | $0^\circ \leq \phi \leq 360^\circ$                    |         |                | PITCH/YAW                               |                                                                 | CN, Cm<br>Cy, Cn                           |                                                |
| DRAG                 | $M \leq 7.0$          | $\alpha = 0^\circ$                   | /                                                     |         |                | /                                       |                                                                 | /                                          | /                                              |
| DTMB                 | $0.6 \leq M \leq 3.0$ | $\alpha \leq 20^\circ$               | $\phi = 0^\circ$                                      |         | /              | $A \leq 10^\circ$                       |                                                                 |                                            | /                                              |
| MAP                  | $M \leq 8.0$          | $\alpha \leq 15^\circ$               | $0^\circ \leq \phi \leq 90^\circ$                     |         |                | $A \leq 15^\circ$                       |                                                                 |                                            | /                                              |
| MASP                 | $M \leq 3.0$          | $\alpha \leq 15^\circ$               | $\phi = 0^\circ, 45^\circ$                            |         | /              | $A$ pitch $\leq 15^\circ$               |                                                                 |                                            |                                                |
| MISSILE DATCOM       | $M \leq 8.0$          | $\alpha \leq 30^\circ$               | $0^\circ \leq \phi \leq 360^\circ$                    |         |                | $A \leq 20^\circ$                       |                                                                 |                                            |                                                |
| MISSILE aerospatiale | $M \leq 4$            | $\alpha \leq 25^\circ$               | $0^\circ \leq \phi \leq 360^\circ$                    |         |                | $A \leq 15^\circ$                       |                                                                 |                                            |                                                |
| MISSILE ONERA        | $M \leq 4$            | $\alpha \leq 25^\circ$               | $0^\circ \leq \phi \leq 360^\circ$                    |         |                | $A \leq 15^\circ$                       | /                                                               |                                            | /                                              |
| MISSILE 1 NEAR       | $M \leq 4$            | $\alpha \leq 45^\circ$               | $0^\circ \leq \phi \leq 360^\circ$                    |         | /              |                                         | CAw, CAI                                                        |                                            | /                                              |
| MISSILE 2A NEAR      | $M \leq 5$            | $\alpha \leq 45^\circ$               | $0^\circ \leq \phi \leq 90^\circ$                     |         |                | $A \leq 15^\circ$                       | /                                                               |                                            | /                                              |
| MISSILE 3 NEAR       | $M \leq 4.5$          | $\alpha \leq 45^\circ$               | $0^\circ \leq \phi \leq 90^\circ$                     |         |                | $A \leq 40^\circ$                       | /                                                               |                                            | /                                              |
| MRRT                 | $M \leq 5$            | $\alpha = 0^\circ$                   | $\phi = 0^\circ$                                      | /       | /              | /                                       |                                                                 | /                                          | /                                              |
| NFR                  | $M \leq 5$            | $90^\circ \leq \alpha \leq 90^\circ$ | $180^\circ \leq \phi \leq 180^\circ$                  |         | /              |                                         | /                                                               | CN, Cm<br>Cy, Cn                           | /                                              |
| NSWC AP              | $M \leq 8.0$          | $\alpha \leq 45^\circ$               | $0^\circ \leq \phi \leq 360^\circ$                    |         | /              | $A$ pitch $\leq 20^\circ$               |                                                                 | CN, Cm                                     |                                                |
| PORTANCE             | $M \leq 4.0$          | $\alpha \leq 2^\circ$                | /                                                     |         | /              | /                                       | /                                                               | CN, Cm                                     | /                                              |
| SEA DATCOM           | $M \leq 8.0$          | $\alpha \leq 30^\circ$               | $\phi = 0^\circ$                                      |         | /              | $\alpha \leq 30^\circ$                  |                                                                 |                                            |                                                |
| S/HARP               | $M \leq 2.0$          | $90^\circ \leq \alpha \leq 90^\circ$ | $0^\circ \leq \phi \leq 360^\circ$                    |         |                | $A \leq 20^\circ$                       | CAw, CAI                                                        |                                            | /                                              |
| TAD                  | $M \leq 8.0$          | $\alpha \leq 2^\circ$                | $\phi = 0^\circ$                                      |         | /              | /                                       | /                                                               | CN, Cm                                     | Cmq, Cmr                                       |
| TASK2                | $M \leq 4.2$          | $\alpha \leq 20^\circ$               | $\phi = 0^\circ, 45^\circ$                            |         |                | $A \leq 15^\circ$                       |                                                                 |                                            | Cmq, Cmr                                       |
| TRAFIRO              | $M \leq 4.0$          | $\alpha = 0^\circ$                   | /                                                     |         |                | /                                       |                                                                 | /                                          | /                                              |

Computed

/ For no

presentation "Missile DATCOM is by far the most powerful and comprehensive nonproprietary aerodynamics prediction tool available to today's missile designer. It enjoys wide and varied application as a preliminary design tool throughout the aerospace industry, government, and academia." Reference 75 has presented the prediction accuracy summary shown in Figures 145 to 147. Because of its good accuracy in pitching-moment predictions in the linear range (Fig. 147), its capability to predict dynamic stability derivatives, and its compatibility with the author's UNIX based workstation, Missile DATCOM was chosen for this research.

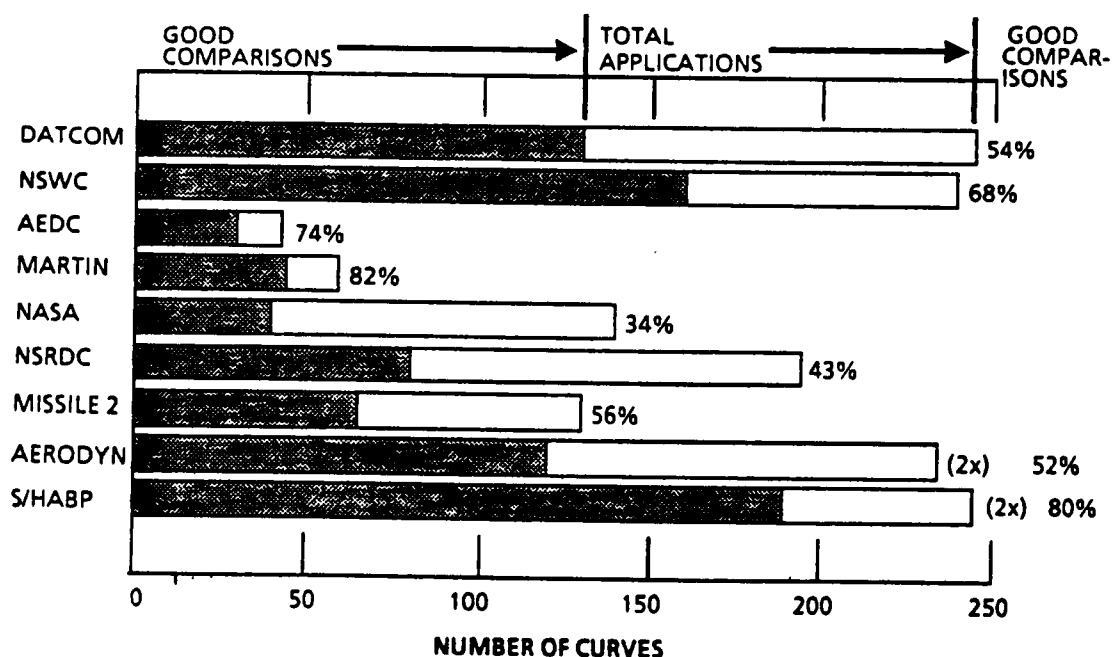


Figure 145. Code prediction accuracy summary.



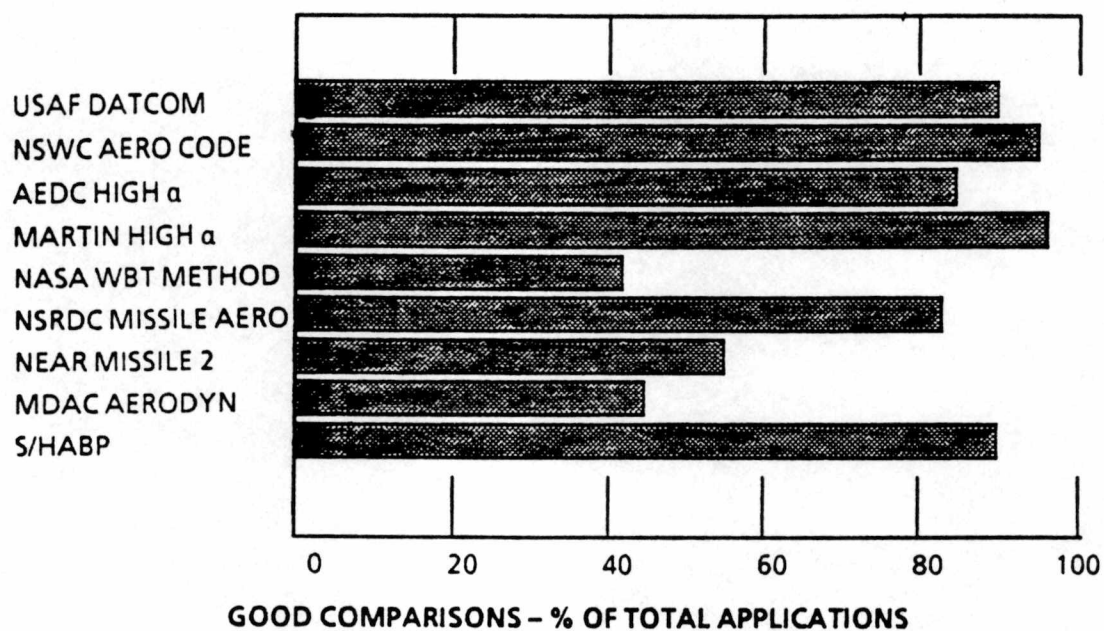


Figure 146. Code normal-force accuracy.

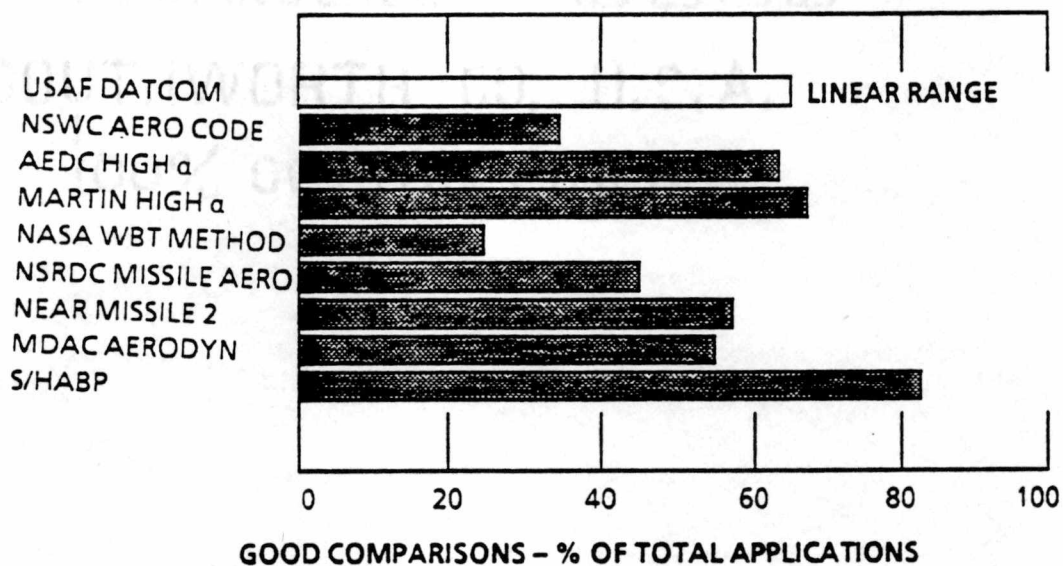


Figure 147. Code pitching-moment accuracy.

Missile DATCOM is composed of a large number of different empirical and theoretical techniques including slender-body, lifting surface, second order shock expansion, and Newtonian flow. Ample use is made of the U.S. Air Force Stability and Control DATCOM used in Chapter V. Different methods are used in each Mach range. Fin interference methods include empirical for viscous lift and theoretical for inviscid lift. The accuracy criteria used in the development were  $C_N \pm 20\%$ ;  $C_m \pm 20\%$ . [73] Typical precision is about  $\pm 10\%$  for lift and  $\pm 20\%$  for control effectiveness [76, 77], although Baker [51] and Nielsen [76] found  $\pm 20\%$  for both, and  $C_m$  has been over predicted by as much as 40% [76, 77]. The damping derivatives are calculated by empirical techniques and typical precision is  $\pm 25\%$ . For a complete listing of the methods incorporated in the Missile DATCOM program subroutines, the reader is referred to Reference 72. A detailed discussion of the methods used can be found in Reference 79.

The input for the Missile DATCOM program can accommodate several varieties of configuration. For the experimental LV of this research, the input variables included: Flight conditions, e.g.,  $\alpha$ ,  $\beta$ ,  $M$ ,  $R_e$ ,  $\phi$  altitude, etc.; Reference quantities, e.g.,  $X_{cg}$ ,  $l$ ,  $d$ ,  $S$ , etc.; Axisymmetric body or contour definition, e.g., longitudinal fuselage stations and corresponding fuselage radii, type of nose, nose geometry, etc.; FCA geometries by set, e.g., number, cross-sectional shapes, airfoil, LE  $\Lambda$ , TE  $\Lambda$ ,  $A$ ,  $c$ ,  $b$ ,  $\tau$ , orientation and position on fuselage, etc.; and FCA deflection values, e.g., incidence, cant,  $\delta$ ,  $\delta_{MIN}$ ,  $\delta_{MAX}$ , FCA number, asymmetric or symmetric deflection, etc. The  $R_e$  is always required. Three types of inputs are permitted to satisfy the Reynolds number requirement:  $R_e$  / length can be specified; altitude and speed can be input, and the  $R_e$  is computed using the Standard Atmosphere; or pressure, temperature, and speed can be input. The author has input the altitude (from the generated trajectories of the experimental LV discussed in Chapter V) together with the speeds for the  $R_e$  calculations.

In order to run the Missile DATCOM simulation program for dynamic wind tunnel test data comparisons in the selected ascent trajectory range, the following were obtained from the experimental LV mass properties and trajectory data (Appendix A. 2 and A. 3). The dynamic test Mach number interval was from  $M = .8$  to  $M = 1.96$ . Changes of consideration during this ascent time interval are indicated in percent at the bottoms of the columns.

| Mach Number | Time (sec) | Altitude (ft) | Q (psf) | Weight (lbs) | cg Station (in) | I (slug-ft <sup>2</sup> ) |
|-------------|------------|---------------|---------|--------------|-----------------|---------------------------|
| .80         | 34         | 13,260        | 575.51  | 1,721,249    | 3860.15         | 132,714,436               |
| .95         | 39         | 17,700        | 691.46  | 1,677,553    | 3863.15         | 131,179,500               |
| 1.20        | 49         | 27,897        | 720.90  | 1,604,655    | 3867.65         | 128,768,416               |
| 1.31        | 53         | 32,165        | 722.57  | 1,577,027    | 3869.20         | 127,902,088               |
| 1.46        | 57         | 37,171        | 718.23  | 1,546,350    | 3870.76         | 126,980,902               |
| 1.96        | 68         | 51,714        | 630.19  | 1,466,188    | 3874.38         | 124,675,602               |
|             |            |               |         | 14.8%        | .37%            | 6.06%                     |

The Missile DATCOM generated results are presented in Table 13. In the original runs, DATCOM discrepancies were found and attributed to: (1) the engine shrouds are not modeled in the Missile DATCOM; (2) the multiple fuselage diameters tax the DATCOM models; and (3) the blunt TE's, i.e., wedge airfoils cannot be accommodated in the Missile DATCOM. To overcome these limitations, the DATCOM cg input was adjusted 9% so that the body-alone DATCOM data (LV fuselage with no FCA's) matched the wind tunnel test data at the critical point of maximum Q in the governing static stability parameter  $C_{m_\alpha}$ . It should be mentioned that discrepancies in general, between wind tunnel test data and Missile DATCOM data, can be attributed to: interference effects between the LV fuselage and the FCA's, interference effects between the FCA's, and the blanking of part of the FCA's by the fuselage; actual 3-D conical flow; viscous effects in the boundary layer; shock wave interference; FCA-body junction gaps; fuselage slots;

nonuniform body upwash; code limitations; simplifying assumptions; limited code data bases; and simplified configurations. White [77] has studied Missile DATCOM / wind tunnel test data discrepancy correlation factors, and Ericsson [80] has found that in dynamic wind tunnel testing the support interference effects are much more severe than in static testing. Sample DATCOM output can be seen in Appendix D. It can be seen

Table 13. DATCOM generated aft-cg dynamic data.

| FCA  | Mach | $(C_{m_q} + C_{m_{\dot{\alpha}}})$<br>per deg | $C_{L_{\alpha}}$<br>per deg | $C_{D_0}$ | $u_0$<br>ft/sec |
|------|------|-----------------------------------------------|-----------------------------|-----------|-----------------|
| None | .8   | -.231970                                      | .06344                      | .100      | 851.36          |
|      | .95  | -.502378                                      | .06459                      | .166      | 993.89          |
|      | 1.20 | -.387560                                      | .06891                      | .270      | 1204.4          |
|      | 1.31 | -.496316                                      | .08606                      | .516      | 1290.8          |
|      | 1.46 | -.460187                                      | .10320                      | .504      | 1413.2          |
|      | 1.96 | -.399210                                      | .08438                      | .456      | 1897.1          |
| # 1  | .8   | -.446937                                      | .1074                       | .102      | 851.36          |
|      | .95  | -.517632                                      | .1179                       | .179      | 993.89          |
|      | 1.20 | -.478618                                      | .1042                       | .288      | 1204.4          |
|      | 1.31 | -.577614                                      | .1189                       | .531      | 1290.8          |
|      | 1.46 | -.544398                                      | .1283                       | .518      | 1413.2          |
|      | 1.96 | -.448576                                      | .1030                       | .465      | 1897.1          |
| # 6  | .8   | -.403317                                      | .1348                       | .103      | 851.36          |
|      | .95  | -.323805                                      | .1517                       | .203      | 993.89          |
|      | 1.20 | -.495690                                      | .1204                       | .330      | 1204.4          |
|      | 1.31 | -.584902                                      | .1278                       | .577      | 1290.8          |
|      | 1.46 | -.528136                                      | .1325                       | .567      | 1413.2          |
|      | 1.96 | -.443366                                      | .1021                       | .523      | 1897.1          |
| # 11 | .8   | -.49699                                       | .09977                      | .102      | 851.36          |
|      | .95  | -.638456                                      | .1025                       | .169      | 993.89          |
|      | 1.20 | -.893780                                      | .1122                       | .279      | 1204.4          |
|      | 1.31 | -.995325                                      | .1290                       | .525      | 1290.8          |
|      | 1.46 | -.760333                                      | .1257                       | .513      | 1413.2          |
|      | 1.96 | -.453556                                      | .1018                       | .463      | 1897.1          |

that both the drag and the pitch damping increase in magnitude up to the max Q point, then both decrease for all cases considered, and that FCA # 6 has the best overall lift coefficient gradient, as in the static wind tunnel test results of Chapter VII.

Next, the DATCOM output was used to calculate the pertinent dynamic stability parameters for the short period mode assessment, eq. (78), where

$$C_{Z\alpha} = - (C_{L\alpha} + C_{D0})$$

$$Z_{\alpha} = - (C_{L\alpha} + C_{D0}) Q S / \mu \quad ; \quad m_q = C_{m_q} \left( \frac{d}{u_o} \right) Q S d / I_y$$

$$m_{\alpha} = C_{m_{\alpha}} Q S d / I_y \quad ; \quad m_{\dot{\alpha}} = C_{m_{\dot{\alpha}}} \left( \frac{d}{u_o} \right) Q S d / I_y$$

(79)

and the actual mass and moments of inertia were used at each point. It should be mentioned here that in the literature and in computer codes, the  $C_{m_q}$  and  $C_{m_{\dot{\alpha}}}$  terms are nondimensionalized by either  $u_o$  or  $2 u_o$ . Table 14 shows the short period mode results for the aft-cg experimental LV for the series of FCA's and Mach numbers tested.

Table 14. Short period mode dynamic stability results for aft-cg LV.

| FCA  | Mach Number | $C_{Z_\alpha}$<br>(rad) | $\omega_n$<br>(rad/sec) | $\zeta$ | t<br>(sec) | P<br>(sec) | N<br>(cycle) |
|------|-------------|-------------------------|-------------------------|---------|------------|------------|--------------|
| None | .80         | -3.735                  | .7849                   | .03758  | 23.5       | 8.01       | 2.92         |
|      | .95         | -3.867                  | .8824                   | .05681  | 13.8       | 7.13       | 1.93         |
|      | 1.20        | -4.218                  | .9506                   | .04061  | 18         | 6.62       | 2.71         |
|      | 1.31        | -5.447                  | .9706                   | .04831  | 14.8       | 6.48       | 2.27         |
|      | 1.46        | -6.417                  | .9933                   | .04457  | 15.7       | 6.33       | 2.46         |
|      | 1.96        | -4.835                  | .9744                   | .02605  | 27.3       | 6.45       | 4.22         |
| # 1  | .80         | -6.256                  | .6388                   | .08341  | 13.0       | 9.87       | 1.31         |
|      | .95         | -6.934                  | .6880                   | .09218  | 10.9       | 9.17       | 1.19         |
|      | 1.20        | -6.258                  | .8287                   | .06206  | 13.5       | 7.6        | 1.77         |
|      | 1.31        | -7.343                  | .8634                   | .06718  | 11.9       | 7.3        | 1.63         |
|      | 1.46        | -7.869                  | .9047                   | .05887  | 13.0       | 6.96       | 1.86         |
|      | 1.96        | -6.366                  | .9214                   | .03196  | 23.5       | 6.82       | 3.44         |
| # 6  | .80         | -7.826                  | .5757                   | .09784  | 12.3       | 11         | 1.12         |
|      | .95         | -8.895                  | .5900                   | .09806  | 12         | 10.7       | 1.12         |
|      | 1.20        | -7.228                  | .7686                   | .07279  | 12.4       | 8.2        | 1.51         |
|      | 1.31        | -7.899                  | .8018                   | .07522  | 11.5       | 7.86       | 1.46         |
|      | 1.46        | -8.159                  | .8415                   | .06337  | 13         | 7.48       | 1.73         |
|      | 1.96        | -6.373                  | .8998                   | .03253  | 23.7       | 6.99       | 3.88         |
| # 11 | .80         | -5.818                  | .6704                   | .08197  | 12.6       | 9.40       | 1.34         |
|      | .95         | -6.042                  | .7316                   | .09323  | 10.2       | 8.63       | 1.17         |
|      | 1.20        | -6.708                  | .8365                   | .09345  | 8.87       | 7.54       | 1.17         |
|      | 1.31        | -7.916                  | .8716                   | .09610  | 8.27       | 7.24       | 1.14         |
|      | 1.46        | -7.715                  | .9141                   | .07004  | 10.8       | 6.89       | 1.57         |
|      | 1.96        | -6.296                  | .961                    | .03150  | 22.9       | 6.53       | 3.20         |

Table 14 shows that: the maximum damping ratio occurs before or at max Q (i.e., at  $M = 1.31$ ) for each case considered; the maximum damping ratio in the Mach range tested improved over that with no FCA's by 62.3% for FCA #1; improved by 72.6% for FCA #6; and improved by 69.2% for FCA #11. Pitch damping and damping ratio plots for all cases tested are shown in Figures 148 and 149. As in the static stability analyses of Chapter VI, FCA #6 again is the best. Here, FCA #6 provides the best dynamic stability based on the short period mode damping ratio. FCA #6 also has the best (lowest) hinge moment since it has the smallest chord length (as shown in Chapter V).

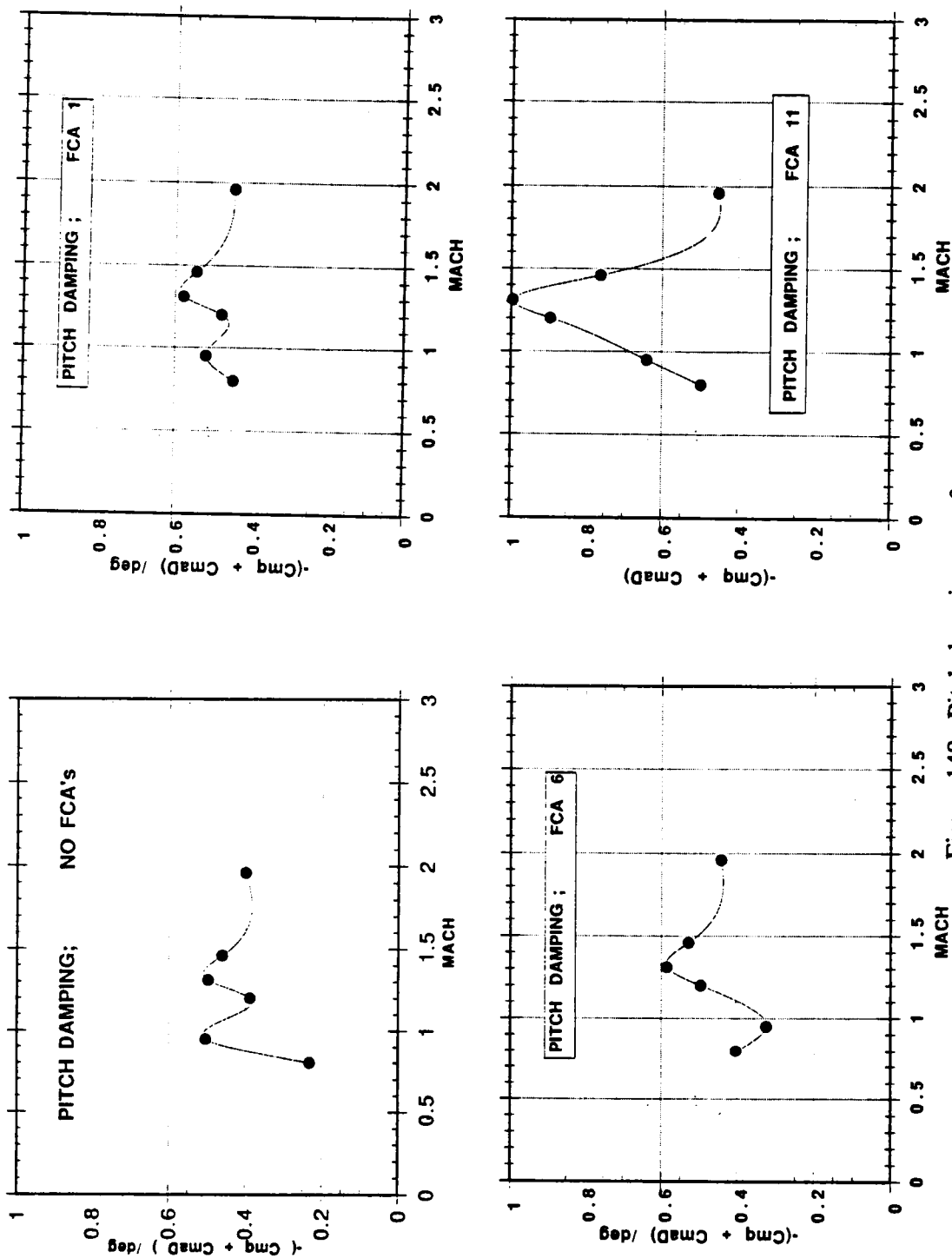


Figure 148. Pitch damping; aft-cg.

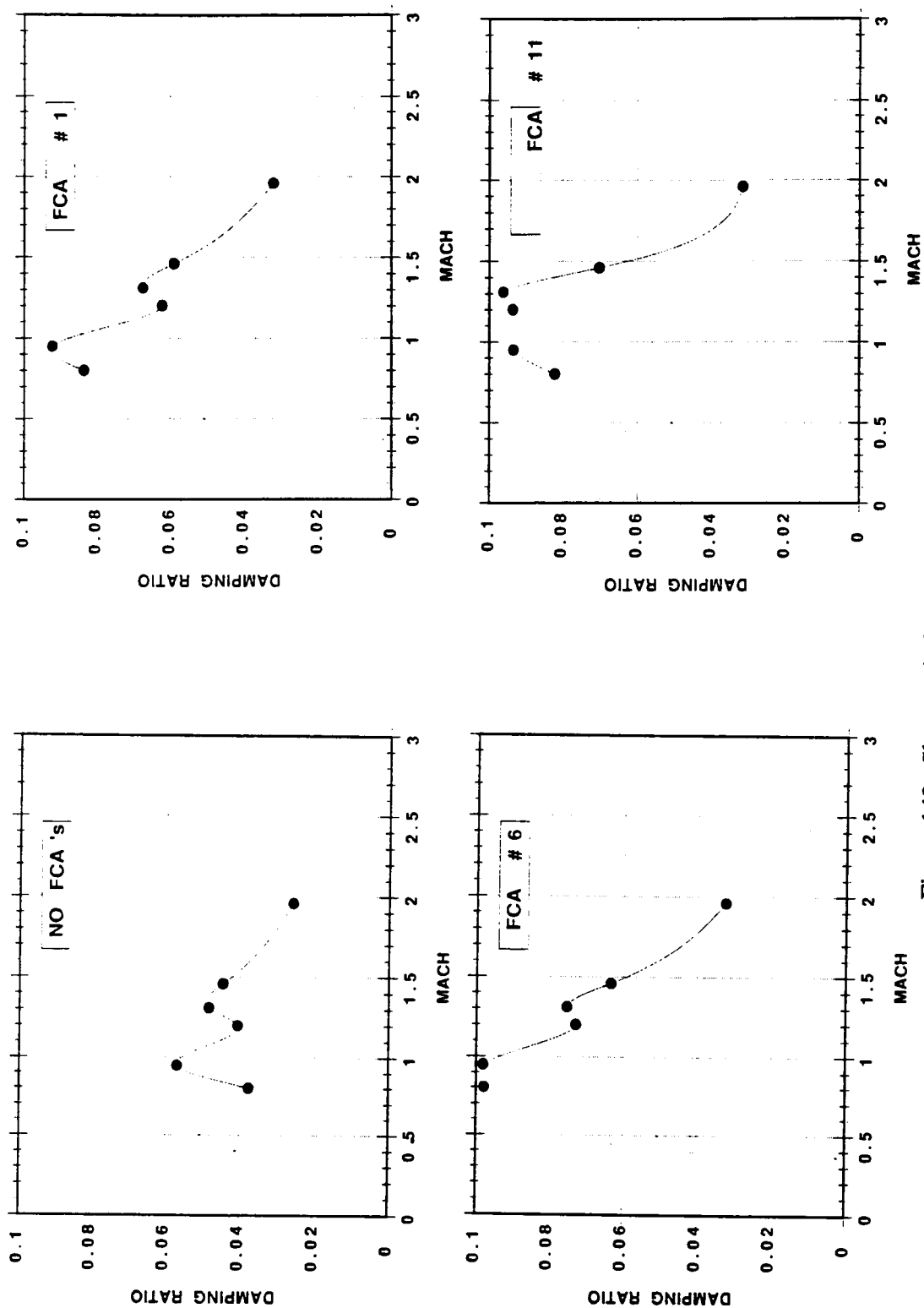


Figure 149. Short period mode; aft-cg.



Figure 150 shows sample raw data from the dynamic stability wind tunnel test program which was run for the forward-cg case to facilitate the free oscillation test method described in Chapter VIII. Figures 151 and 152 show the dynamic stability wind tunnel test results obtained from the tedious data reduction process which included counting the cycles / sec, periods, and times-to-half-amplitude. The data reduction equations incorporated the wind-on and the wind-off wind tunnel test values to arrive at the dynamic stability derivatives which were then substituted into the short period mode equations. More dynamic stability raw wind tunnel test data and the resulting numerical tabulations can be found in Appendix D.

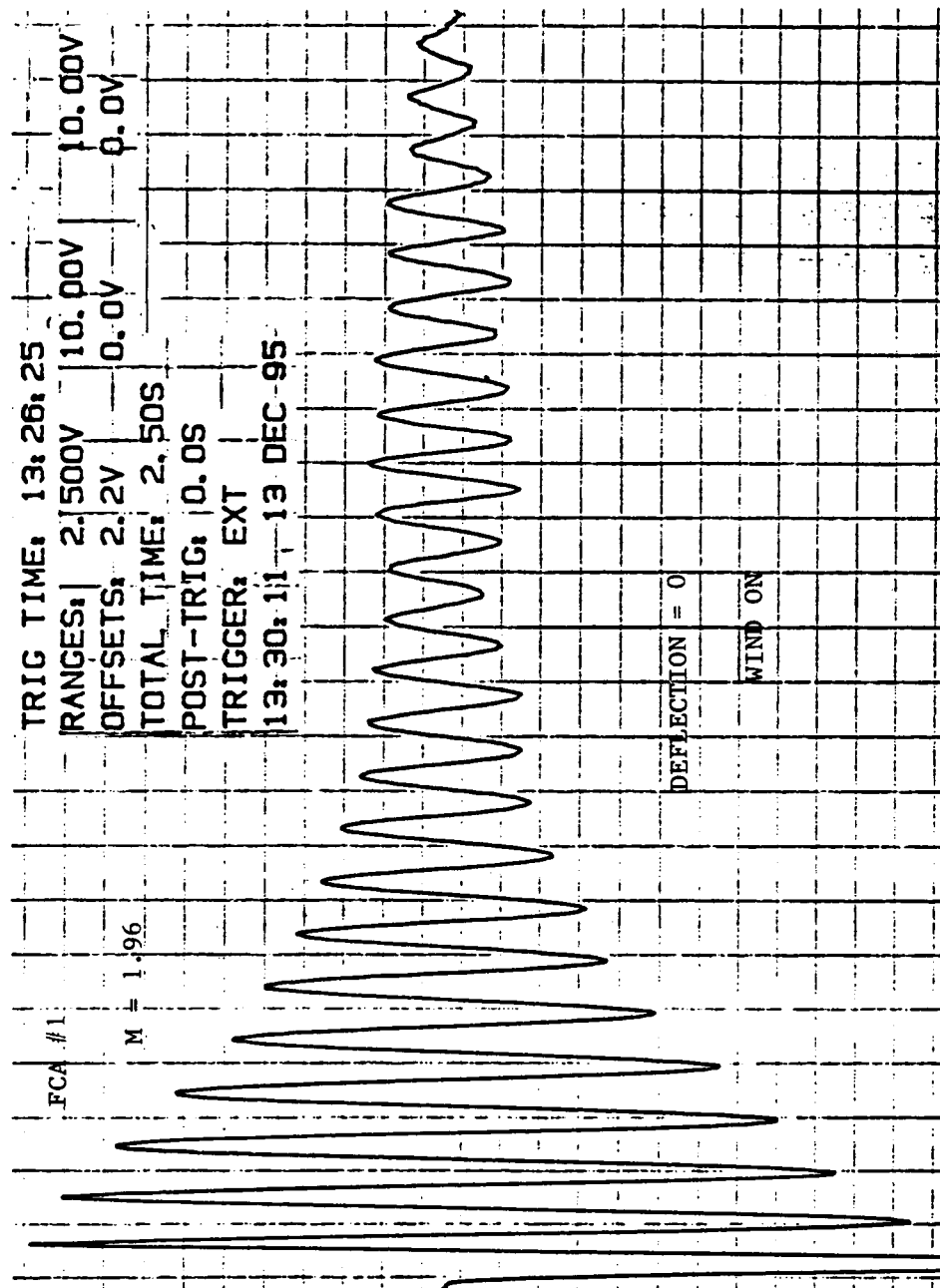


Figure 150. Sample dynamic wind tunnel test raw data.

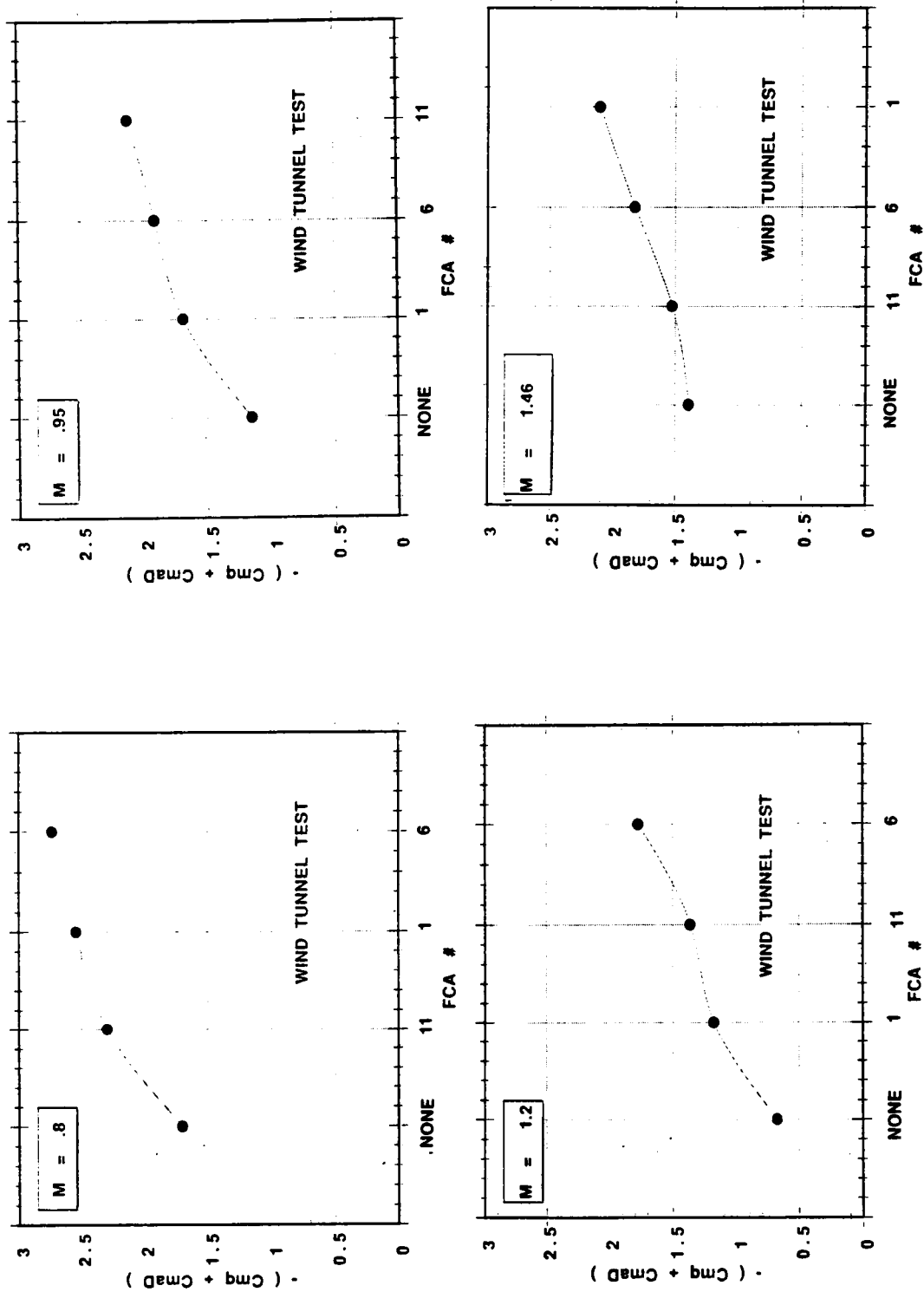


Figure 151. Pitch damping; forward-cg.

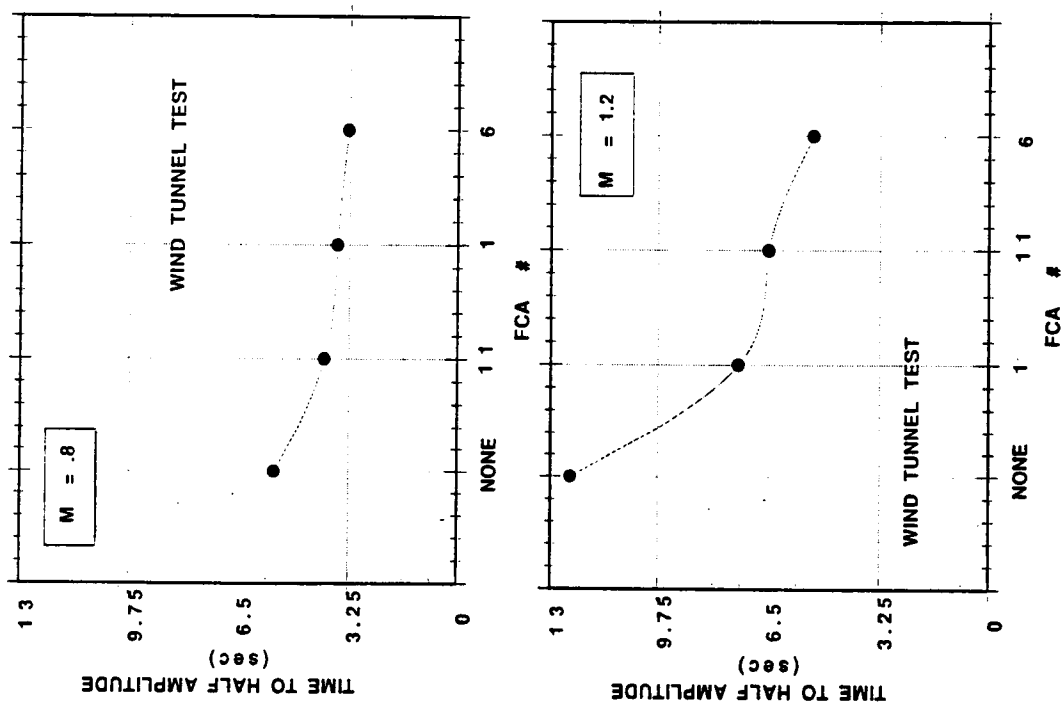
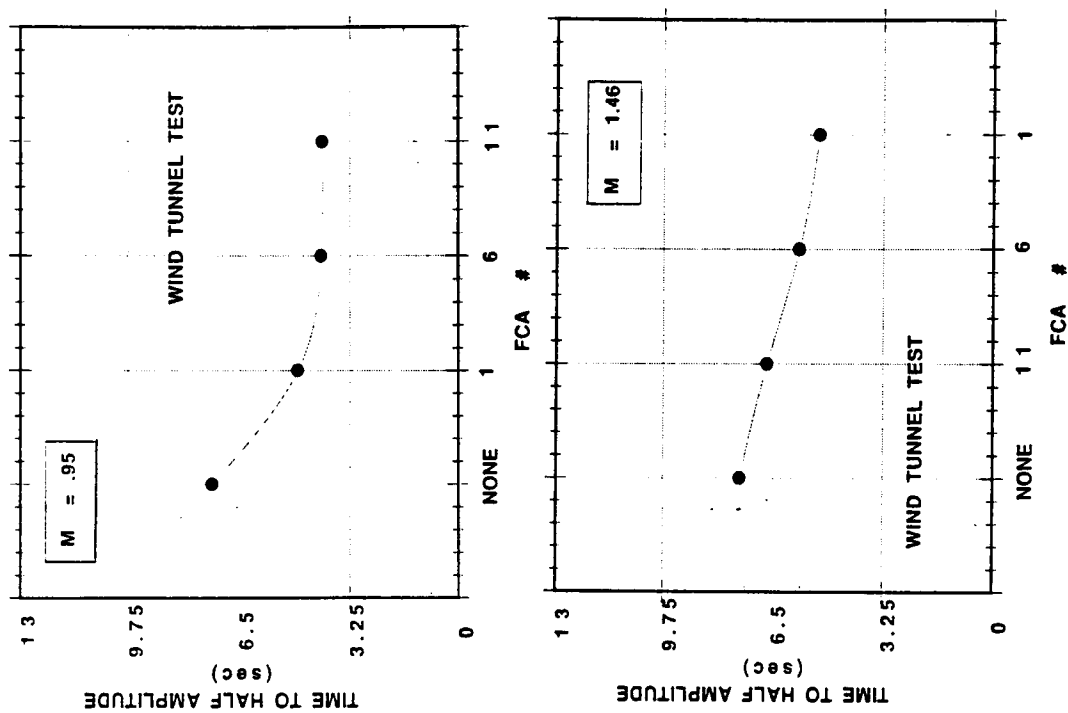


Figure 152. Short period mode;



forward-cg.

## **CHAPTER 10. STATE-OF-THE-ART ASSESSMENT OF SMART MATERIALS AND ADVANCED COMPOSITES DIRECTLY APPLICABLE TO THE DESIGNED FLIGHT CONTROL AUGMENTORS**

This research effort draws on advancements in many other fields. Launch vehicle (LV) aerodynamic surfaces have not been designed in the U.S. for nearly 40 years because of weight penalties. However, since NASA went to the Moon with the Saturn V (discussed in Chapters II and III) using 300 ft<sup>2</sup> of aerodynamic surfaces, the wealth of smart materials and advanced composites that have been developed allow for the design of very lightweight, strong, and innovative LV flight control augmentors with possible smart actuators. The state-of-the-art assessment conducted in this chapter shows that LV aerodynamic flight control augmentors of future LV's do not need to weigh as much as the aerodynamic surfaces of the Saturn LV series, and that smart materials may be incorporated into the design for possible smart actuation of the surfaces.

As part of this research, the author spent several days at the extensive production facilities for titanium matrix composite (TMC) at the Textron Specialty Materials Company in Lowell, Massachusetts, which is the world's first plant (government funded) built solely for the production of TMC (for NASA's National Aerospace Plane). While there, the author studied the TMC fabrication technology including chemical deposition plants and manufacturing plants. A Flight Control Augmentor Design Review was held with the Director of Research, Director of Fabrication, Director of Product Applications, Director of Operations, and the TMC Plant Manager. The 12 sets of designed flight control augmentors (FCA's) were reviewed, together with their prospective loads and temperature environments. Material and fabrication details were discussed at length and cost estimates were obtained for the full scale augmentors. Thus, any added weight concerns were addressed, and patent disclosures were also filed using smart materials for

smart actuation of the LV designed augmentors. The assessment in this chapter serves as precursor to these events.

New areas of engineering design are often born when previously unrelated areas of research evolve to the point where the benefits of their synergistic combination cannot be ignored. [81] Aerodynamic flight control surfaces and smart materials may well be in this class, ripe for technology fusion. A plethora of diverse classes of active materials and advanced composites is now available to the flight control engineer.

Many smart materials can function both as sensors and as actuators. A smart actuator can be a solid, a liquid, or a gas, which changes its character, and hence the character of its surrounding structure. The author has found all of the following terms used in the literature: smart structures, intelligent materials, smart materials, smart composites, smart skins, active structures, active materials, adaptive materials, shape memory effect, shape memory alloys, shape memory alloy hybrid composites, metal matrix composites, and advanced composites, with applications ranging from buildings and bridges to microelectro-mechanical biomedical devices. The purpose of this assessment is to winnow out, from this massive and burgeoning array of smart materials and advanced composites, those that are directly applicable to the designed FCA's. Nowhere in the literature has there been found any applications of smart materials or advanced composites to large Saturn class LV flight control surfaces.

In this assessment, "smart" will denote the ability to sense defined flight parameters, e.g., applied loads, temperatures, or vibration, and actuate an appropriate response in a predetermined and controllable fashion. While smart structures employ sensors, actuators, and control mechanisms integrated into the structure, smart materials can be fabricated in such a manner that sensors, actuators, and control systems will be part of the microstructure of the material itself. Microfabrication technologies, which have evolved from the integrated circuit, are being applied to produce silicon

micromechanisms such as integrated sensors and actuators on a chip. Micropumps and valves will deliver microquantities of liquids or gases. As this rapidly growing field of microelectrical-mechanical systems (MEMS) develops, new selections of sensors and actuators will be available for space applications. [82]

### 10.1 Microelectrical-Mechanical Systems

MEMS have increasing aerospace applications and can increase the sampling speed. A valuable data base of microfabrication techniques and actuator designs has emerged. Methods of microactuation include electrostatic and piezoelectric (PE) actuators, fluid expansion, and bimetals. Impact-friction actuated micromechanisms are new actuators in which lateral resonant structures are retracted by electrostatic comb drives and propelled forward toward the actuated micromechanism by elastic forces generated by folded beam flexures at frequencies ranging from 10 to 20 kHz (Fig. 153). [82]

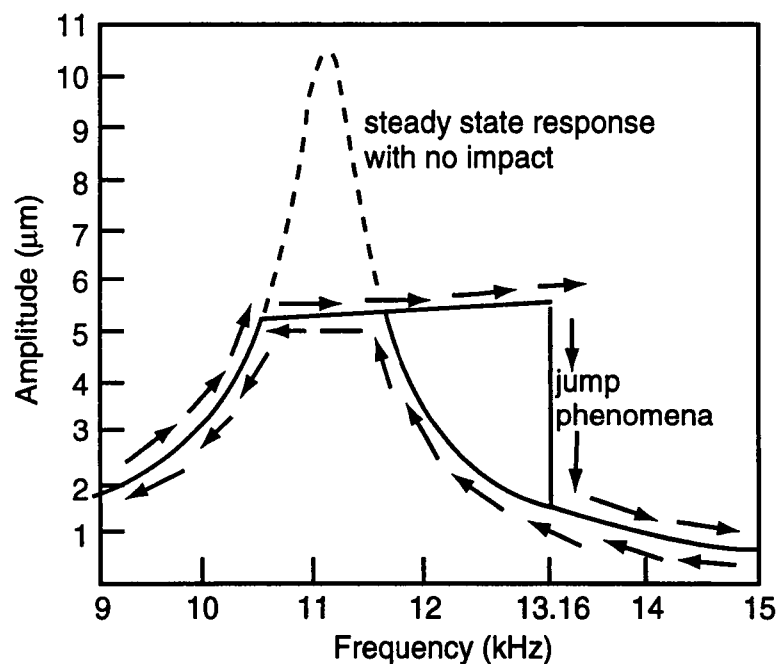


Figure 153. Impact actuated jump frequency phenomena with MEMS.

The jump frequency is a strong function of the coefficient of restitution. Two types of impact actuators are a microoscillator and a polysilicon microvibromotor. Near equiatomic Ti-Ni alloy undergoes a forceful shape recovery due to a crystalline phase change at near ambient temperature. Many active modal modification strategies for vibration control have been demonstrated with a variety of smart materials including Ni-Ti fiber-reinforced epoxy. [81] Shape memory devices are heat actuated. The heat of actuation must be removed before the next cycle can commence. The energy required for miniature actuators can be supplied by small conventional batteries. Ti-Ni actuators change shape about 3%. Currently, these devices deliver relatively low forces and small displacements and there are many competing technologies. [82]

## **10.2 Smart Materials**

Smart structures and smart materials have emerged during the past few years as 21st century technology. These materials have the inherent ability to interface with modern control systems, and their dynamic response characteristics can be actively controlled in real time. They can allow mode suppression, vibration isolation, dynamic alignment, and shape control, and have been successfully demonstrated. [83] Smart flight vehicle skins can determine the presence and amount of ice buildup on critical aerodynamic surfaces. Smart flight controls integrate the technologies of controls, aerodynamics, flight dynamics, sensors, actuators, and materials (Fig. 154).

The dual versatility of some key elements such as piezoelectrics, magnetostrictives, and shape memory alloys can be exploited by employing the same material as a sensor and an actuator. For example, a shape memory alloy could serve as a sensor to monitor ice accumulation or vibration response on a flight control surface prior to becoming an actuator and limiting the amplitude of vibration or ridding the ice. PE ceramics have proven effective both as sensors and actuators in active vibration control and active acoustic absorption. [84] Smart structures have the sensors and actuators



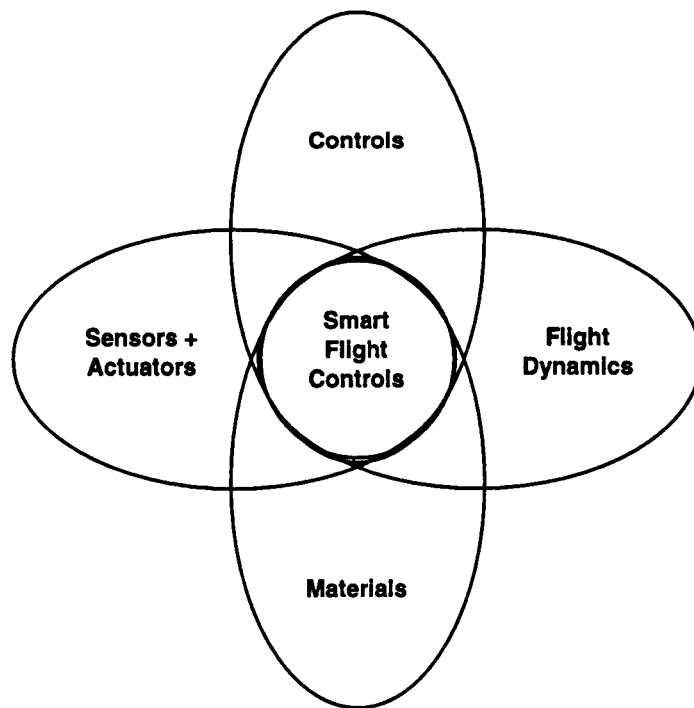


Figure 154. Key disciplines of smart flight control technology.

embedded into the structure. The ideas are simple. At the structural level, an intimately integrated sensor system provides data on the structure's environment and its loading to a processing and control system which in turn signals intimately integrated actuators to modify the structure's properties in an appropriate fashion. These systems offer immense benefits in space structures applications.

The smart material takes the concept one stage further by an important integration of engineering, material science, chemistry, and physics. The sensing and response functions are built into the material itself using a chemical or morphological structure to provide the responsivity. Photochromics could be classed as smart materials and indeed they are, but the interest is in much more far reaching ideas such as materials which change stiffness, mechanical properties, or even mass distribution under the influence of external fields. These sensing and actuating devices can be easily embedded in advanced

composites. A more sophisticated approach is to embed the actuators in the flight controls, also to respond to flight conditions by changing deflection positions. Materials which can physically deform themselves are candidates for smart flight control surfaces. Chemical synthesis methods, such as sol-gel and Langmuir-Blodgett techniques, and physical methods for film formation, such as ion beam deposition and implantation, laser melting and ablation, plasma etching, and chemical vapor depositions have been joined by new classes of techniques which do material manipulation on an atomic scale, such as the scanning tunneling microprobe. [81]

Smart materials are tailored to provide combinations of sensor, information processor, effector, and feedback functions within the materials themselves and to adapt to their environment and provide a useful response to changes. Both the sensor and the actuator functions with the appropriate feedback must be integrated. There are two main approaches to designing smart materials (Fig. 155). [81] One approach combines system sensor, processors and effector functions in a single group. The other approach follows integrated circuit fabrication concepts and uses deposition and materials modification methods to create complex multilayer structures with embedded sensor, memory, logic, and signal functions. [81]

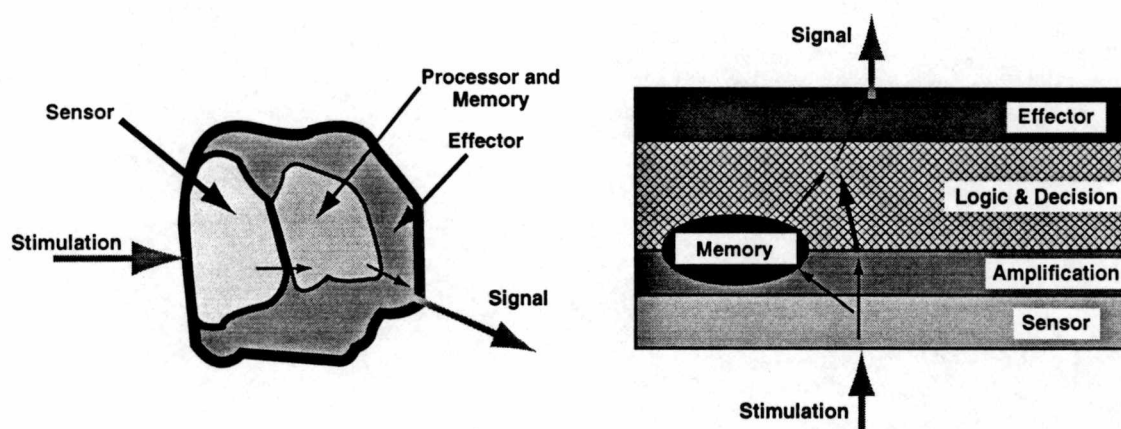


Figure 155. Two approaches to design of smart materials.

In LV FCA's, a basic smart structure could be a surface film on the flight control, which incorporates a network of sensors and acoustic wave generators which could relay flight parameter information such as stress, loading, temperature, dynamic pressure, etc., and respond with flight control actuation. Smart flight control surfaces would require in-flight measurement of rapidly changing dynamic loads on the vehicle. The requirement extends to mechanical vibrations at acoustic frequencies and spatial resolutions of centimeters or better. Active vibration control of smart fibrous composite structures featuring electrorheological (ER) fluids, PE, and fiber optic sensing systems have been demonstrated. Shape memory alloys, PE, and ER fluids have been used to increase aerodynamic efficiency.

Dynamic aeroelastic control uses induced strain actuators such as piezoceramics and electrostrictives which are regulated to apply equivalent forces and moments on the aerodynamic surfaces in order to deform and effect dynamic aeroelastic control. Active surfaces with integrated strain actuators provide a natural mechanism for effecting multiple-actuator dynamic control and for implementing modern model-based control methodologies such as H-infinity. Strain actuated active aerodynamic surfaces can be used for gust and load alleviation (disturbance rejection), flutter and vibration suppression (plant regulation), and maneuver enhancement (command following). [83]

Variable camber has been used to decrease root bending moment, drag, and wind gust response. An F-111 investigation addressed adaptive modifications of wing LE's and TE's to achieve large deformations. [83] It has been shown that significant reduction in shock-induced drag can be achieved by changing shape during flight regime transition. By judicious selection, smart materials can synthesize numerous classes of hybrid actuation systems in order to satisfy a broad range of performance specifications. Some hybrid smart materials feature embedded multifunctional actuation systems that capitalize

on the diverse strengths of ER fluids and PE materials and which operate in conjunction with fiber optic sensing systems. [81]

### **10.3 Sensors**

Optical fiber technology has substantial aerospace and hydrospace usage and provides a basis for gathering data from many points with negligible interference and wide bandwidths. Optical fibers react to temperature and pressure by distorting light. Smart wing LE's with 300 optical sensors in 3 layers have been used by Boeing. [85] McDonnell Douglas has test flown smart structures on the F-15 LE's, and smart helicopter rotors. [85] Fiber optic flight control systems have also been tested in a NASA F-18. Because optical fibers can transmit more than one signal at a time, they offer multiplexing advantages to flight control and can communicate with avionics and actuators. The same fibers that monitor conditions can be used for control. The optimum fibers could be the communication pathway between a LV's flight controls and its avionics. LV flight control systems could benefit from fiber optic smart sensors in the manufacturing stage. Fiber sensors embedded in the material could monitor structural changes and vibration, temperature, etc., in support of a vehicle shape control system to allow adaptive guidance. Fiber optic smart structures in combination with actuators have been used for flight vehicle vibration control, localized strain monitoring, and icing control. [81]

A key requirement in a fiber sensor system for smart materials application is compatibility with the composite material. The sensor must not affect the performance of the flight control surface. Glass fibers are very compatible with composite materials. Physical similarities between the optical fibers and the reinforcing fibers used in composites allow the optical sensor to be successfully embedded within the material. Both fibers have similar densities, and optical fibers are capable of withstanding strains of

the same magnitude as the composite itself. Although monomode optical fibers can be made as small as 100  $\mu\text{m}$  in diameter, this is ten times the size of a typical reinforcing fiber. For good strain transfer between composite and optical fiber, there must be sufficient bonding. This can be enhanced by the use of suitable coatings. Tensile, compressive, and shear testing of composite materials containing embedded sensors and actuators have indicated minimal effect on the mechanical preparation of the composite. By careful choice of sensors, actuators, their coatings, and host composite compatibility, it is possible to prevent degradation of the mechanical properties of the host composite. It has been shown that embedded arrays of optical fibers had no detrimental influence on the resistance of composites to delamination. It has also been shown that the compressive strength of Kevlar<sup>TM</sup>-epoxy panels is not compromised. [81]

What do embedded optical fibers really measure? Interpretation of signals produced by structurally embedded phase-based optical fiber sensors experiencing thermomechanical loading is complicated because the fiber is subjected to a strain state, which is a tensor quantity, in addition to the scalar temperature field. Therefore, the scalar signal produced by an embedded optical fiber sensor has contributions from a minimum of four quantities; three principal strains and the temperature. Interferometric optical fiber sensors, therefore, do not measure the strain or the temperature directly. They measure the optical phase shifts from which the strain and temperature states are inferred. The interferometric optical fiber sensors are the most prominent in smart materials. Fiber-optic interferometric and polarimetric strain gauges are particularly well suited for embedding into smart composite materials, which could easily be used as flight control surfaces. It has been shown that spatial modulation is applicable to optical fibers braided into a three-dimensional (3-D) composite material, and sensor sensitivity and dynamic range can be improved by special geometrical configurations of the embedded optical

fibers (bow-tie and e-core high bi-refrangement fibers), and launching light into one of the eigenmodes of the fiber. [81]

A large body of research has been done on the analysis of intermodal interference patterns to extract information from an optical fiber to determine changing environmental conditions seen by the fiber along its length. Fiber optic strain gauges compare favorably to the conventional resistive strain gauges. They also allow for additional sensing of the stress fields transverse to the fiber axis. As the applied stress increases, the result is a modal power redistribution caused by intermodal coupling. The modal power redistribution is such that there is a transfer of power to lower-order modes. [81]

Mach-Zehnder, Michelson, and Fabry-Perot are the main dual path interferometric fiber optic sensors used. MBB-Deutsche Aerospace has established teams of researchers to develop smart structures for aerospace applications including automated positioning of flight control systems, flutter suppression, vibration damping, impact detection, and modal analysis of flight controls. Shape control and active vibration damping have been investigated for flight controls in particular. It is hoped that the twist and camber can be optimized for flight conditions. Modal domain interferometric fiber optic sensors are used to detect vibration in smart helicopter blades using embedded PE crystals. Contour holography has also been used. [81]

An autonomous aircraft wing and flight control deicing system has been developed by Innovative Dynamics for NASA Lewis Research Center (LeRC). Sensors in the LE detect the ice, and then vibrations of the skin are excited to knock it off. The system includes actuator coils, strain sensors, microelectronic circuits, and a microprocessor integrated into modules mounted internally. The module performs two operations: measurement and deicing. During measurement, relatively small excitations are applied to selected coils, and the resulting small vibrations are monitored by the strain

sensors. The sensor readings are fed to the central processor, which is programmed to recognize patterns of vibrations that signify various thicknesses and distributions of ice.

When the central processor determines that the ice at a given location is thicker than allowable, the module at that location is commanded to perform the deicing operation in which a much larger excitation is applied to the coils. The resulting larger mechanical impulse knocks the ice off that spot. The system monitors the thickness of ice continually, performing the deicing operation only when and where necessary. [86]

Very high-temperature strain gauges that will operate at temperatures of 2,000 °F and adjust for the thermal components of the output are being developed at LeRC. An active gauge wire is bonded to the strain specimen via a ceramic base coat. An inactive gauge wire made of the same alloy as the active wire is laid out around the active gauge wire. A piece of ceramic cloth is cemented at its edges to the base coat and covers the inactive wire and holds it in loose contact with the base coat. Thus, the strain from the specimen is coupled into the active gauge wire, but not into the inactive wire. The active gauge wires sense the strains, while the inactive gauge wires provide the compensation for the thermal contributions to the gauge readings. [86]

In other flight control embedded sensor applications, the use of composite materials readily allows the embedding of sensors. It may be desirable to measure strain every 10 cm to fully map out the strain field. Two classes of fiber sensors could be used, one set to localize an event and one set to make detailed assessments. The fiber sensors would be very close to the diameter of the fiber, and a number of sensors can be multiplexed along a single fiber line. When existing fiber optic sensors are considered, two candidates for meeting these requirements are fiber optic grating and etalon-based fiber sensors. These sensors may be multiplexed into strings of fiber optic sensors. These strings could be multiplexed using a switch that could be used to interrogate sensors when needed. Figure 156 shows a portion of the fiber optic smart material

architecture. [81] A fiber sensor demodulator is used to extract the data from the sensors in the string being accessed. The data are then formatted and transmitted to a system signal processor, which in turn transfers the data to the vehicle assessment system. [81]

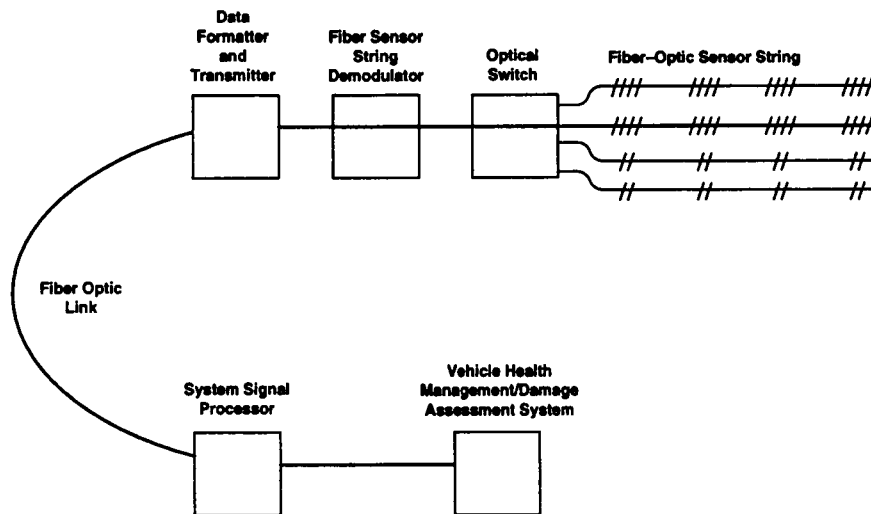


Figure 156. Fiber optic smart material architecture.

### 10.3.1 Fully Distributed Sensing of Strain

The locations and magnitudes of strain-induced mode coupling and relative changes in propagation constants have been measured using broadband optical sources in two-mode fibers. Differential mode delay occurs between the stress point and the output end of the fiber. This delay can be compensated for at the receiver to permit coherent detection. If a large fraction of the power launched into the sensing fiber can carry the sensor information, performance is improved. System architectures offering efficient use of launched power employ forward scattered light. Two-mode elliptical core fiber sensors have been widely proposed for strain and vibration sensing in smart materials and structures applications. One of the most attractive features of this approach is the single-fiber configuration which can be implemented using insensitive single-mode input and output fibers to interrogate the core sensing section. A two-mode fiber sensor can be



viewed as a form of common fiber two-beam interferometric device, where the two beams are formed by the two spatial modes in the fiber. Due to their different propagation constants, there is an effective optical path difference between the modes within the fiber. Remote demodulation of an elliptical-core fiber strain sensor using phase detection developed for two-beam fiber interferometry and multiplexing of core sensors remotely addressed by modulated-source interrogation have been demonstrated by the Naval Research Laboratory. [81]

Proximity sensors are of the transmissive type. They are based on a triangulation technique using optical fibers and graded-index (GRIN) lenses. One fiber provides the illuminating beam, while the other detects parts of the light scattered by the target. Any target displacement in the sensing region gives rise to the intensity modulation of the detected signal. The range of operation and resolution of the sensors can be varied according to the geometrical arrangement of the optical head. These sensors can operate as on/off switches as well as analog sensors. A differential two-core fiber can be used for separate measurement of strain and temperature, and a dual-wavelength technique can be used to perform absolute measurements over an extended dynamic range for aerospace applications. Interferometry has the advantage of high sensitivity with low power levels. To be able to measure strain over a wide temperature range, the system must monitor both temperature and strain simultaneously, and two-core fibers are suited. The cores should work as independent waveguides. [81]

### **10.3.2 Strain Sensors With Phase Sensitive Detection**

Bragg grating sensors are useful for quasi-static distributed sensing in smart structures and materials for the real time evaluation of load, strain, temperature, vibration, etc. Their wavelength-encoded nature of output has distinct advantages over direct intensity based sensing schemes. As the sensed information is encoded directly into the

wavelengths, which is an absolute parameter, the output does not depend on the total light levels, losses in the connecting fibers and couplers, or source power. The means to detect wavelength shifts of a grating sensor are based on the use of an unbalanced fiber interferometer wavelength discriminator. The wavelength component, reflected back from a fiber grating illuminated by a broadband source, is fed to an unbalanced Mach-Zehnder interferometer. This light becomes the source into the interferometer, and wavelength shifts induced by perturbation of the grating resemble a wavelength modulated source. [81]

This concept has been widely used in the field of interferometric fiber sensors to introduce a phase carrier signal into an unbalanced interferometer using direct laser emission-frequency modulation for the purpose of phase demodulation. The converse of this can also be used. The unbalanced interferometer can be used as a discriminator to detect the wavelength shifts in the effective source formed by the strained grating element. High resolution techniques for detection of dynamic strain-induced shifts in the reflection wavelength of an in-fiber Bragg grating sensor have been demonstrated. Dynamic strain resolution at frequencies above 100 Hz has been demonstrated by the office of Naval Research. [81]

Fiber optic grating sensors are produced by exposing an ultraviolet (UV) beam on a short length of an optical fiber by a transverse holographic method. The UV beam causes permanent changes in the index of refraction along a small part of the fiber. If broadband light is sent into the fiber, there will be notches and peaks in the transmitted and reflected spectrum, respectively, where the Bragg condition is satisfied. By using fiber optic gratings with different wavelength, one optical fiber can be used for spatially distributed sensing. [81]

Optical fibers, transducers, and PE skins have been used for structural vibration and shape control. The upper operational temperature of silica-based optical fiber sensors

is about 1,000 °C; other optical sensors, e.g., sapphire based, have demonstrated use at 2,000 °C. In addition to optical sensors, PE film can be sputtered onto a metal fiber. This active sensor develops an electric field when strained and can detect vibration as well as fracture in a manner similar to acoustic emission. These are well suited for embedding in composite materials and have been successfully tested in carbon composite beams and filament wound cylinders at vibration frequencies of 2,100 Hz. [81] Flight vehicle smart skins incorporate PE film modal sensors which supply information on individual dynamic mode shapes. A direct measurement of the natural modes of vibration could help eliminate the control system estimation techniques based on complex algorithms that decompose a vibration into natural modes. In-flight aircraft deicing systems using PE films to sense change in wing vibration signature that result from ice buildup could be used for LV FCA's.

Two new sensor-related technologies, silicon micromachined sensors and polyimide-based waveguide structures, have been developed (Honeywell) that make acoustic waveguide sensors available for smart flight control surfaces. In the silicon micromachined sensors, applied pressure exerts a force that deflects a diaphragm which delivers an applied stress to the micromachined sensor area. The sensor chip features an active partitioned sensor region; typical sizes are 10 to 100  $\mu\text{m}$ . Polyimide-based waveguide structures are optical waveguides or light pipes featuring micrometer-level sizes and surface conformal construction relevant to smart materials applications. Key benefits of the polyimide optical waveguide structures for smart surface applications include high-bandwidth throughput ( $>10$  GHz) and electromagnetic interference immunity. They could be easily embedded into composite smart flight control surfaces or conformally applied to metallic flight control surfaces. Honeywell investigations indicate that propagation losses of 0.01 dB/cm are soon achievable. Optical waveguide sensors featuring a silicon chip assembly have been designed for flight vehicle smart material

applications. Incident light is coupled onto the sensor assembly from the optical fiber connected to the flight vehicle support avionics interface. Optical power is sourced from an avionics interface. Individual sensors are interconnected by an optical bus medium as a fault-tolerant communications network. The incoming light is coupled onto the sensor via a GRIN lens to minimize optical alignment problems due to temperature coefficient differences between the sensor assembly and the smart material. Mechanical vibration causes the sensor's cantilever beam to oscillate, which interrupts the passage of light through the waveguide assembly. The optical light is either intensity-modulated or wavelength-modulated, depending on the application, and coupled back to the flight vehicle avionics interface for detailed analysis. These sensors have low-profile conformal packaging and high-temperature operation ranges. [81]

Optical fiber sensors for aircraft damage monitoring can detect cracks of 5 to 30  $\mu\text{m}$  in Al alloys. Fabry-Perot sensing systems have been designed for aerodynamic surface shape control and vibration damping. To detect force, pressure, and bending, PE is the best fiber-optic sensor. [84] Sensors embedded in the LV flight control surfaces could record load histories. Embedded pressure sensors would be the ideal. [87] Embedded fiber optic-based sensors have demonstrated the ability to detect impact damage on flight control surfaces for the purpose of in-flight reconfiguration of the flight controls. The sensor output can be interpreted and passed to the vehicle control system for actuation of embedded actuators, and smart flight control surfaces could measure their own loads. Whereas PE and strain gauges give local domain information, the fiber optic interferometer and shape memory alloy methods give integrated global information. [81]

#### **10.4 Electroheological Fluids**

ER fluids are one class of smart materials that change rheological characteristics very rapidly when electric fields are applied and can provide rapid response in controlled

mechanical devices. This change in behavior in the presence of an electric field renders them suitable to control variable stiffness or damping in a material. The ER fluids are typically suspensions of micron-sized hydrophilic particles suspended in suitable hydrophobic carrier liquids, that are Newtonian (free-flowing) with no field present, but stiffen reversibly to a non-Newtonian fluid, or a solid as field strength increases so that viscosities are adjustable (Fig. 157 and Table 15). [84] When these fluids are embedded in voids in structural materials, the imposition of an electric field permits the mechanical properties of the material to be actively controlled. The voltages required to activate the phase change are typically 1 to 4 kV/mm of fluid thickness, but since current densities are in the order of 10 mA/cm<sup>2</sup>, the total power required to change properties is quite low. [84] Both hydrous and anhydrous ER fluids have been studied. Hydrous systems were restricted to small ranges of operating temperature, but the new generation of anhydrous ER fluids have overcome this limitation. [81]

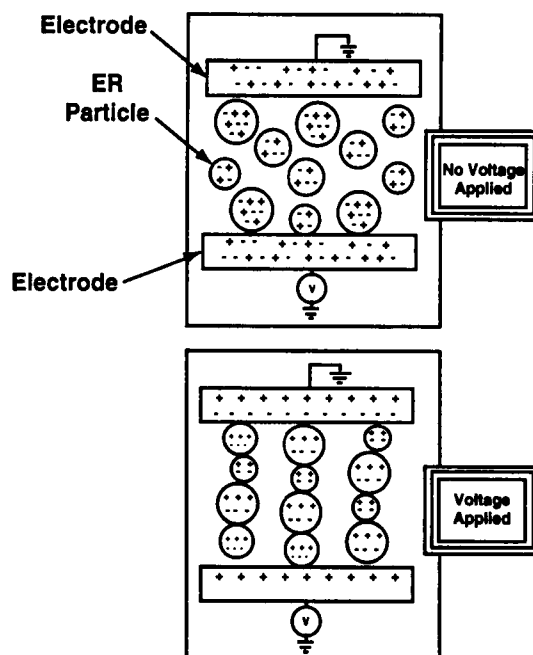


Figure 157. The ER phenomenon.

Table 15. ER hydrophilics and hydrophobics.

| Solute                    | Solvent                        | Additive                   |
|---------------------------|--------------------------------|----------------------------|
| Kerosene                  | Silica                         | Water and detergents       |
| Silicone oils             | Sodium carboxymethyl cellulose | Water                      |
| Olive oil                 | Gelatine                       | None                       |
| Mineral oil               | Aluminum dihydrogen            | Water                      |
| Transformer oil           | Carbon                         | Water                      |
| Dibutyl sebacate          | Iron oxide                     | Water and surfactant       |
| Mineral oil               | Lime                           | None                       |
| P-xylene                  | Piezoceramic                   | Water and glycerol oleates |
| Silicone oil              | Copper phthalocyanine          | None                       |
| Transformer oil           | Starch                         | None                       |
| Polychlorinated biphenyls | Sulphopropyl dextran           | Water and sorbitan         |
| Hydrocarbon oil           | Zeolite                        | None                       |

ER materials, noted for very fast response times, have many aerospace applications as both sensor and actuator where stiffness, damping, and natural frequencies of flight control surfaces can be dramatically changed in milliseconds and where variable performance is required. [81] Using Bingham plastic ER material behavior and steady-state condition, the relationship between flow rate and pressure drop for a configuration can be determined. [88] ER adaptive structures are primarily based on the controllability of the pre-yield rheology of ER suspensions. Two basic configurations of ER materials that have been incorporated in the structure and studied include the shear and the extensional configurations (Figs. 158 [81] and 159 [81]). [88] Controllable devices are mostly based on post-yield shear behavior of ER suspensions. Adaptive structures are made controllable through the incorporation of ER materials at selected locations. Controllable valves were some of the first ER devices investigated. They contain fixed electrode configurations with an ER suspension material flowing through. The pressure drop across and flow rate through the device is controlled by an electric field. Benefits of ER valves include fast response time and freedom from mechanical moving parts. Valve pressure drops as high as 1,000 lb/in<sup>2</sup> have been achieved, and modulation frequencies on the order of several hundred hertz have been realized. Studies of ER shakers have

demonstrated that high force output and wide operational bandwidth are possible. ER controllable machinery and engine mounts have been widely investigated (Fig. 160). [88] Controllable mounts in the 1 to 50 Hz range having two states, a low damping

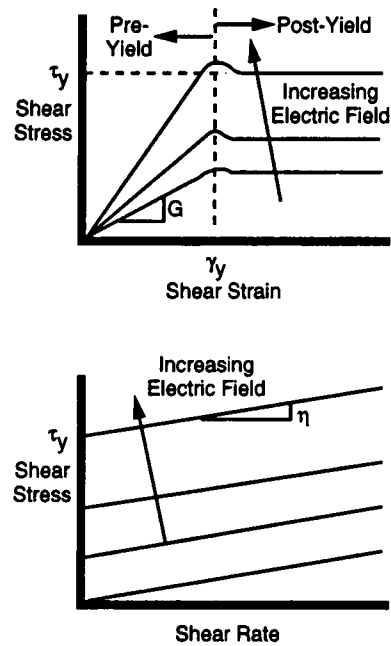


Figure 158. ER material behavior.

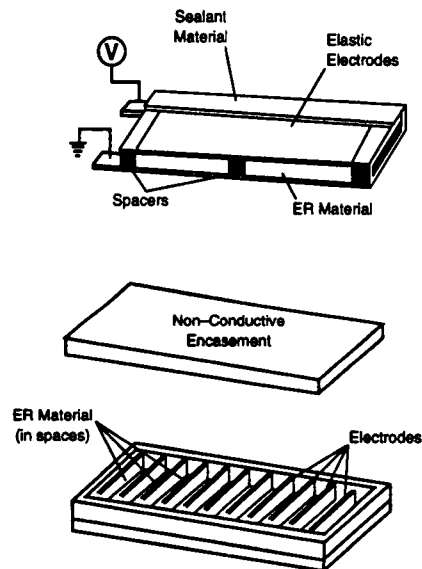


Figure 159. ER incorporation into the structure.

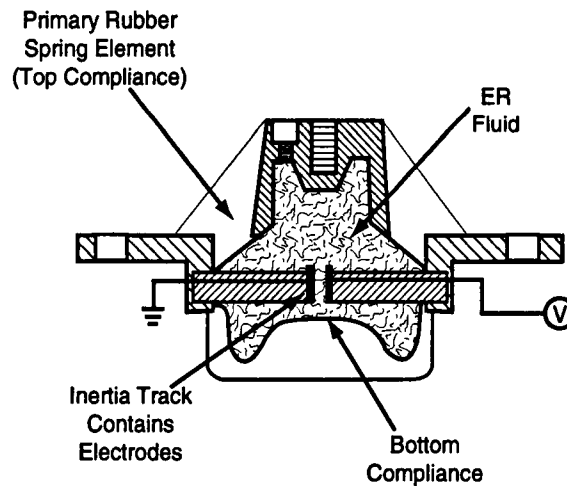


Figure 160. ER material damping application.

off-state and a very high damping on-state, have demonstrated long duration testing and positive control at temperatures of 110 °C. Response times of less than one millisecond, sinusoidal force patterns with amplitudes of 2,200 lb, and frequencies of 1,000 Hz have been demonstrated by ER shakers.

ER dampers have been investigated for aircraft landing gear components and large damping struts to reduce vibrations. ER materials have been applied to aircraft wings and helicopter rotors to control stiffness of each blade independently and are suited to flight control surfaces. These systems sense external stimuli and react in an appropriate manner so as to meet prespecified performance criteria. [81] They can modify the natural frequency of a system by sensing the dominant vibration frequency band and adjusting its viscosity. [87] The helicopter dampers handle dynamic loads of 2,000 N over a frequency range of 0 to 20 Hz. Struts have been laboratory tested up to frequencies of 150 Hz. [88]

Smart composites featuring ER fluids have been investigated for flight vehicle applications at Michigan State University. Dynamically tunable ER fluids contained in voids of the composite can be used for flight control surface vibration control. [88] Magnetorheological fluids have also been used for vibration control. [83] Some aircraft



programs are nonarticulating flight control surfaces, where the appropriate regions at the edges of the relevant airfoil sections deform in order to provide the aerodynamic control required. Many programs are in progress for smart wings to control dynamic response.

Active dampers utilizing ER fluids have been used for missile flight control fin actuation systems. The program features fins that can be electrically switched between conductive and nonconductive states for microwave shielding application. Laminates can utilize different ER fluids to design specific properties (Fig. 161). [84]

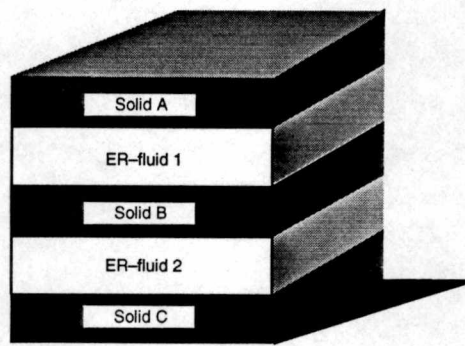


Figure 161. Laminate configuration of different ER fluids.

### 10.5 Piezoelectrics

PE materials are solids capable of exerting mechanical forces in response to an applied voltage, or vice-versa; consequently they can be used as either sensors or actuators in smart materials. PE actuators permit controlled voltages to generate axial forces or bending moments. Lead zirconate titanate,  $\text{Pb}(\text{Zr},\text{Ti})\text{O}_3$ , commonly referred to as PZT, exhibits a strong PE effect. The deformation is linear with respect to the applied field and changes sign when the electric field is reversed. PZT actuators have been used in active mirror control for many years. Tables 16 [84] and 17 [87] show typical properties. Their faults are hysteresis and aging. The Curie temperature of a PE material defines its thermal limit, some as low as 200 °F. The current generation of smart PE materials is synthesized with composite laminates (Fig. 162). [84]

Table 16. PE material properties.

| Property                 | Units                                | PVDF | PZT   | Bz-Ti-O <sub>3</sub> |
|--------------------------|--------------------------------------|------|-------|----------------------|
| Density                  | (10 <sup>3</sup> ) kg/m <sup>3</sup> | 1.78 | 7.5   | 5.7                  |
| Relative permittivity    | $\epsilon/\epsilon_0$                | 12   | 1,200 | 1,700                |
| d <sub>31</sub> Constant | (10 <sup>-12</sup> ) m/V             | 23   | 110   | 78                   |
| g <sub>31</sub> Constant | (10 <sup>-3</sup> ) V m/N            | 216  | 10    | 5                    |
| k <sub>31</sub> Constant | percent at 1 kHz                     | 12   | 30    | 21                   |

Table 17. PE film properties.

| Property                                        | Value                                                         |
|-------------------------------------------------|---------------------------------------------------------------|
| Thickness                                       | 9,28,52,110×10 <sup>-6</sup> m                                |
| Piezoelectric strain constant - d <sub>31</sub> | 23×10 <sup>-12</sup> $\frac{\text{m/m}}{\text{V/m}}$          |
| Piezoelectric strain constant - d <sub>33</sub> | -33×10 <sup>-6</sup> $\frac{\text{m/m}}{\text{V/m}}$          |
| Piezoelectric stress constant - g <sub>31</sub> | 216×10 <sup>-3</sup> $\frac{\text{V/m}}{\text{N/m}^2}$        |
| Piezoelectric stress constant - g <sub>33</sub> | -339×10 <sup>-3</sup> $\frac{\text{V/m}}{\text{N/m}^2}$       |
| Electromechanical coupling factor               | 12% (at 1 kHz)<br>19% (at 1 kHz)                              |
| Capacitance                                     | 380 pF/cm <sup>2</sup> for 28×10 <sup>-6</sup> meter film     |
| Young's modulus                                 | 2×10 <sup>9</sup> n/m <sup>2</sup>                            |
| Speed of sound                                  | 1.5 to 2.2×10 <sup>-3</sup> m/s (transverse thickness)        |
| Pyroelectric coefficient                        | -25×10 <sup>-6</sup> C/m <sup>2</sup> K                       |
| Permittivity                                    | 106 to 113×10 <sup>-12</sup> F/m                              |
| Relative permittivity                           | 12-13                                                         |
| Mass density                                    | 1.78×10 <sup>3</sup> kg/m <sup>3</sup>                        |
| Volume resistivity                              | 10 <sup>13</sup> ohm meters                                   |
| Surface metallization                           | 2.0 ohms/square for Cu-Ni                                     |
| Resistivity                                     | 0.1 ohms/square for Ag ink                                    |
| Loss tangent                                    | 0.015 to 0.02 (at 10 to 10 <sup>4</sup> Hz)                   |
| Compressive strength                            | 60×10 <sup>6</sup> N/m <sup>2</sup> (stretch axis)            |
| Tensile strength                                | 160 to 300×10 <sup>6</sup> N/m <sup>2</sup> (Transverse axis) |
| Temperature range                               | -40 to 80 °C                                                  |
| Water absorption                                | < 0.02% H <sub>2</sub> O                                      |
| Maximum operating voltage                       | 750 V/mil = 30 V/10 <sup>-6</sup> m                           |
| Breakdown voltage                               | 2,000 V/mil = 100V/10 <sup>-6</sup> m                         |

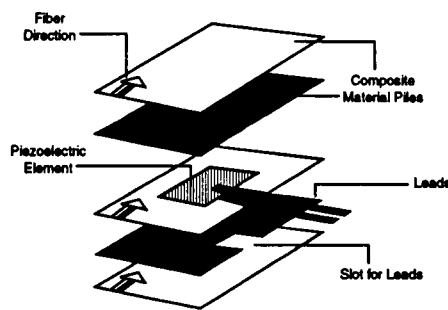


Figure 162. Embedded PE elements in composite plies.

PE materials have been used for vibration control on helicopters, and aeroelastic applications have been investigated to reduce fatigue and flutter induced by aerodynamic buffeting in flight control surfaces, fins, canards, and wings, by NASA LaRC. Aeroelastic control of aircraft wings using embedded PE materials in proportion to the wing root loads has been demonstrated by McDonnell Douglas. Flexural axis position has been controlled by the feedback gain which has controlled the lift effectiveness. Placing the flexural axis ahead of the wing aerodynamic center can cause elastic wing lift to be greater than the lift of a rigid wing of the same geometry. This effect can be used to reduce the effect of gust loads. Variable feedback gain also controls the effective bending and torsion. [83]

By incorporating PE rods into a composite material, directional actuation can be achieved, and applications include shape control of a wing or a deflection of a control surface, although shape memory alloys would allow larger deflections. The undesirable aging fault would not affect use on an expendable LV. Hysteresis and temperature limits, however, would have to be considered. These are nonlinear at high applied voltage levels and larger deformations. [87] Beam tip deflections and axial extensions have been demonstrated with piezoelectrics. [84]

Active aeroservoelastic missile fins using directionally attached piezoelectric (DAP) actuator elements have been developed. This fin uses a NACA 0012 airfoil and

has been wind tunnel tested to within 6% of predicted static deflection. [83] Smart surfaces can be used for gust load reductions and LV trim. Using adaptive controlled surfaces with smart materials for actuators, aeroelastic interaction can be tailored to produce new forces and control torques.

Most actuator systems are pneumatic, hydraulic, or electromechanical and occupy from 2 to 6% of fuselage volume. Because these actuators are mounted within the fuselage, they cannot be placed adjacent to flight control surfaces and require bearings and linkages. A fin actuator contained entirely within the aerodynamic surface will occupy no fuselage internal volume. A smart torque plate integrated into a fin could generate pitch deflections of the fin as a solid-state flight control actuator. Solid-state flight control surfaces using smart materials have been investigated. It has been shown that extremely small amounts of smart deflection could be effectively magnified by the dynamic pressure of the air flow. Performance of low AR wings have been studied using active camber and twist control. From studies of polyvinylidene film, PE fiber composites, and conventionally attached and DAP element actuator materials, it has been shown that torsional actuation favors highly orthotropic actuation elements. Of the materials studied, DAP elements generated deflections and moments that were double that of other materials. An active torque plate has been integrated into a 66- by 93-mm missile fin to produce pitch deflections in excess of  $\pm 3^\circ$  and grow with increasing airspeed. [83]

## **10.6 Electrostrictives**

Because of the high hysteresis of PE materials, electrostrictives were developed. In electrostriction, the strain is linearly proportional to the square of the applied electric field. A PE crystal generates strain. The operating principle involves the diffuse phase transition of a relaxer ferroelectric material upon the application of an electric field. The

electrostrictive phenomenon is attributed to the rotation of small electrical domains when an external electrical field is imposed. In the absence of this field, the domains are randomly oriented. The alignment of these electrical domains parallel to the electrical field results in the development of a deformation field in the electrostrictive material. [84]

The electrostrictive ceramic compound lead-magnesium-niobate (PMN) exhibits shape memory effects with an electric field. The thermoelectrostrictive effect and the quadratic electrostrictive phenomenon are dependent upon the temperature. This type of material, while exhibiting less hysteresis than piezoceramics, features a nonlinear constitutive relationship requiring some biasing in practice to achieve a nominally linear relationship over a limited range of excitation. [84]

## **10.7 Magnetostrictives**

These solid materials act similarly to the PE materials, only they respond to magnetic rather than electric fields and are the magnetic analog of electrostrictives. The magnetostriction phenomenon is attributed to the rotations of small magnetic domains in the material, which are randomly oriented when the material is not exposed to a magnetic field. The orienting of these small domains by the imposition of a magnetic field results in the development of a strain field. As the intensity of the magnetic field is increased, more magnetic domains orientate themselves so that their principle axes of anisotropy are colinear with the magnet field in each region and finally saturation is achieved. Terbium-iron alloys are typical magnetostrictive materials. [84]

These materials have large energy density and high modal damping, and they convert strain energy into a magnetic field which produces voltage and current in a transducer coil. High force, high strain, wide bandwidth, and linear actuators are available, driven by the new class of giant magnetostrictive materials such as Terfenol-D™. Terfenol-D™ is for terbium (Ter), iron (Fe), dysprosium (D) and Naval Ordinance

Laboratory (NOL) and uses the form  $Tb_x-Dy_{1-x}-Fe_2$ , a common example of which is  $Tb_{0.3}-Dy_{0.7}-Fe_2$ . [89] A solenoid is used to supply the magnetic field and a Terfenol™ rod is used to supply the strain. The Terfenol™ core is placed inside a solenoid, and a current is passed through the solenoid inducing a strain in a direction along the core axis. The Terfenol™ rod is then elongated. The displacement is proportional to the magnetic field intensity. The Terfenol-D™ drive element is sized based on the force and displacement requirements. Terfenol-D™ actuators are known for high energy density, thus a compact actuator can produce a high force. Terfenol™ actuators offer improvements over stacked PZT devices with no aging effects or thermoelastic creep and operate in temperatures over 300 °C.

Studies using Terfenol-D™ in dampers, actuators, and velocity sensors have been done by Satcon Technology Corp. for MSFC. It has been shown that Terfenol-D™ has a wide temperature operating range from above room temperature to cryogenic, and that it is a competitive alternative to PE actuators (Table 18 [89] shows Terfenol-D™ material properties). Strokes over 100  $\mu m$  and forces of 1,700 N are available from small actuators. The self-sensing capabilities of Terfenol-D™ actuators allow development of smart actuators.

Table 18. Terfenol-D™ material properties.

| Property              | Value                                | Units            |
|-----------------------|--------------------------------------|------------------|
| Density               | $9.25 \times 10^3$                   | $kg/m^3$         |
| Young's modulus       | $2.5 \text{ to } 3.5 \times 10^{10}$ | $N/m^2$          |
| Tensile strength      | 28                                   | MPa              |
| Compressive Strength  | 700                                  | MPa              |
| Thermal expansion     | $12 \times 10^{-6}$                  | $^{\circ}C^{-1}$ |
| Resistivity           | 60                                   | $\mu L/cm$       |
| Relative permeability | 5 to 10                              |                  |
| Coupling factor       | 0.7 to 0.75                          |                  |
| Sound speed           | 1,720                                | m/s              |
| Magnetostriction      | 1,520 to 2,000                       | p/m              |
| Energy density        | $1.4 \text{ to } 2.5 \times 10^4$    | $J/m^3$          |

Magnetostrictives have been used for flight control surface vibration, flutter suppression, and airfoil camber optimizations (Naval Surface Weapons Center). To control flutter in canard, fins, and wings, the critical vibration modes must be damped. A LV goes through three speed regimes during its ascent trajectory and may be subject to undesirable flight control surface vibrations. A NASA LaRC study has shown Terfenol-D™ to have superior strength, higher modulus, and wider operational bandwidth than shape memory alloys. It can generate enough force to overcome the aerodynamic loads on a flight vehicle and has a low magnetic field requirement. Magnetostriction devices are currently being evaluated for shape change of control surfaces on torpedoes and gimbaling of flight deck simulators. Also being evaluated is an integrated actuation system for individual control of helicopter main rotor blades using Terfenol-D™ actuators to provide higher harmonic control of the individual blades. [87]

Aerodynamic surface applications of twist control as well as camber control have demonstrated that significant deflections can be produced. Air loads can be used to further twist a flight control surface. Aileron deflections of 31°/m in twist, and 44°/m in bending have been demonstrated. [81] Grumman Aircraft is using Terfenol-D™ for the control of aircraft flight control surfaces. They have shown that a Boeing 707 aircraft wing can be adaptively deformed over its entire cross section to use optimized aerodynamic shape for varying flight speed regimes. Terfenol-D™ actuators meet the load requirements while being light in weight. Ribs and spars are replaced with smart members containing rods of Terfenol-D™. In order to maximize the displacement obtainable, a diamond-shaped structure was constructed which deflects the TE by 60° (Fig. 163). [87] This technology has great potential in the area of aeroservoelasticity. Higher power versions of Terfenol-D™ smart materials are being developed by Etrema Products, Inc., under a \$250,000 Navy contract. Table 19 shows magnetostrictive

material comparisons. [87] While Terfenol-D™ has 35 times larger strains than Metglas, it cannot yet be laid into composites. [87]

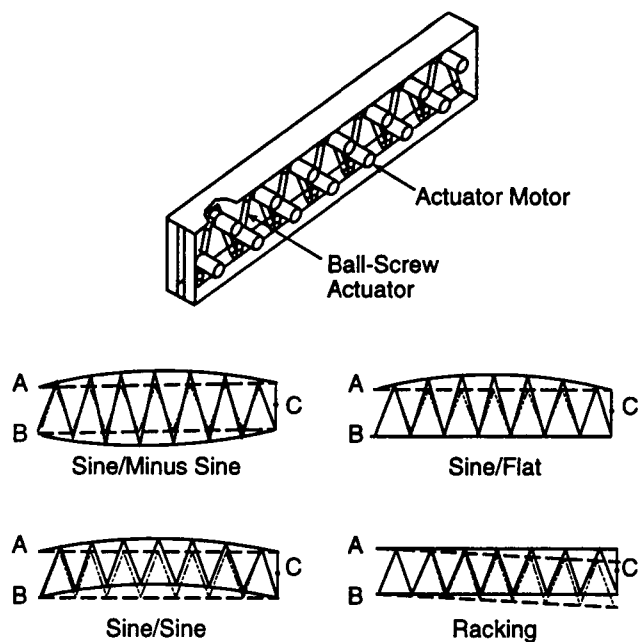


Figure 163. B-707 aileron test article and deformations achieved.

Table 19. Comparison of magnetostrictives.

|                   | Nickel                            | Metglas                           | Terfenol-D™                       |
|-------------------|-----------------------------------|-----------------------------------|-----------------------------------|
| Maximum Strain    | 50 p/m                            | 50 p/m                            | 1,800 p/m                         |
| Field Required    | 6,000 G                           | 1 G                               | 500 G                             |
| Resistivity       | $7 \times 10^{-8} \Omega\text{m}$ | $1 \times 10^{-6} \Omega\text{m}$ | $6 \times 10^{-7} \Omega\text{m}$ |
| Available in foil | yes                               | yes                               | no                                |

## 10.8 Shape Memory Films

Shape memory ferroelastic and ferroelectric thin films have dramatically dependent mechanical properties as their key feature. These are associated with a first-order martensitic transformation from a high-symmetry high-temperature austenitic phase to a low-symmetry low-temperature martensitic phase. The low-temperature phase is easily twinned into many crystallographic twin variants, with one crystallographic twin



growing at the expense of the others with an applied stress. On heating, the twins revert to their original configuration before transformation into the austenitic phase.

Macroscopically, the martensitic material is easily deformed through the growth of these twins, which can accommodate plastic true strains which are recovered on heating, through transformation into the high-temperature phase and shape, hence the term shape and memory effect (SME). Ti-Ni for example, changes 12% in resistance with an increase in stress, which makes it a useful sensor material (Fig. 164). When the material is in its high-temperature phase, large stresses cause isothermal transformation back to the lower-symmetry phase, which can be deformed several percent. On release of the stress, the material reverts back to the austenitic phase with complete recovery of the deformation. The isothermal nature of the deformation has been termed superplastic. Alloys which exhibit SME include: Au-Cd, Ag-Cd, In-Tl, Cu-Al-Zn, Cu-Al-Ni, Ti-Pt, Ti-Nb, and Ti-Ni. [82]

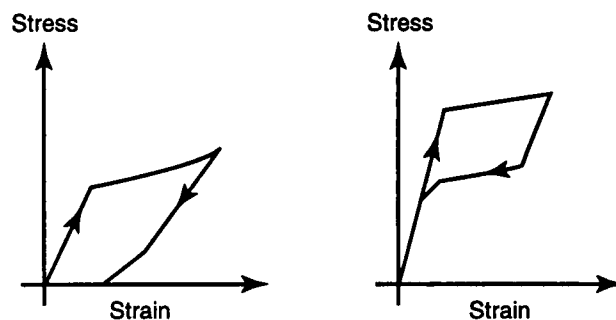


Figure 164. Superelasticity and shape memory effect.

The SME is not limited to metals. Shape-memory-plastics, such as Norsoex and Zeon Shable, have also been developed. Mechanical work has also been obtained from the 15% expansion of paraffin. High-output paraffin positioning actuators have been used to produce 150 lb forces over 0.1 in displacements when powered by 20 W at 28 V. This method can be used to address dynamic response vs. load for accurate positioning for <1 Hz. Shaft extensions result from pressures of 10,000 lb/in<sup>2</sup>. Actuation temperatures are 60

to 110 °C. On cooling, the paraffin begins to solidify after 1.5 °C of thermal hysteresis. Then the actuators retract. [84]

## **10.9 Shape Memory Alloys**

Shape memory alloys (SMA's) are a special class of smart materials which convert thermal energy into mechanical work and are ferrous and nonferrous. There are more than 10 basic SMA's. Some of the most common materials are Nitinol™ (12,000 lb/in<sup>2</sup> yield strength and 6% strain limit), a titanium and nickel alloy named for the Naval Ordnance Laboratory (see above), Cu-Zn-Al, Cu-Al-Ni, and Fe-Mn-Si. SMA's change shape as a function of temperature due to a crystalline phase change. This process occurs between the two different crystalline phases of the alloy, austenite and martensite. Martensite is the low-temperature phase, while austenite is the high-temperature phase that forms above the critical transition phase. The change in crystal structure is from an ordered cubic crystal to a monoclinic crystal phase upon the application of heat. The monoclinic phase has no equal sides and no right angles. To undergo the martensitic transformation, first observed by the German metallurgist, Adolf Martens, the material forms alternate bands or twins, an extremely fine structure of the characteristic martensitic microstructure (Fig. 165). [84] In the soft martensitic state, little force is required to align the twins and deform the material. Upon application of heat, the phase change to the face-centered cubic austenitic state restores the shape and generates significant force and deformation in the process. As shown by studies done for MSFC by E\*Sorb (1992), SMA's can be used to do impressive physical work. Martin Marietta has shown SMA's to have 30 to 40 times the mechanical transductance of PE materials.

The fundamental characteristic of the reversible SMA is that regardless of the severity of the deformation of the low-temperature martensitic phase, heating induces a reverse transformation which allows return to the original shape at the high-temperature

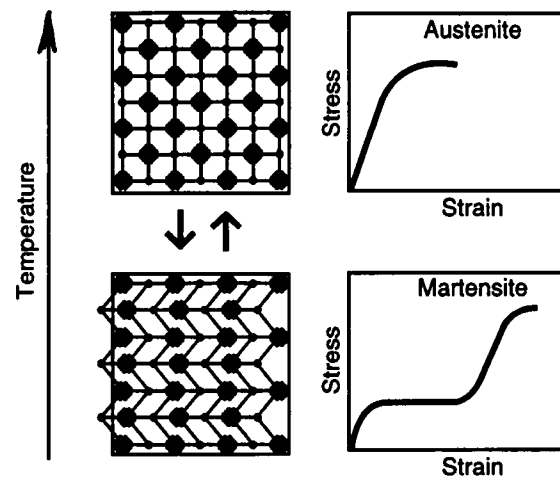


Figure 165. Martensitic transformation of shape memory alloys.

austenitic phase. The heat can be produced by fluids or electricity. If the heating and cooling is controlled by pulsed direct electric current, repeated cyclic motions with high degrees of accuracy can be achieved. [87] The entire subject of martensitic phase transformations is one of considerable technological interest. Understanding the basic physical phenomena has been aided by applying finite element techniques to solve continuum elastic models for phase transitions and by considering atomistic model Hamiltonians of a martensitic system. [84]

SMA's have been utilized both as actuators and as sensors. The Ni-Ti family exhibit good force output and comparatively low hysteresis. SMA actuators are typically actuated by electric current. The transformation temperature of these materials can be selectively tuned over a broad range, and temperature hysteresis has been improved to  $< 2\%$ . The Intelligent Materials Laboratory at Michigan State University has developed a new class of SMA actuators which deliver variable output (Fig. 166). [84] The response time of the SMA actuator is directly controlled by the cooling rate. In an expendable LV, the cryogenics may be used to assist with the cooling rate, and the number of load cycles would not be a limiting factor for the application considered in this research.

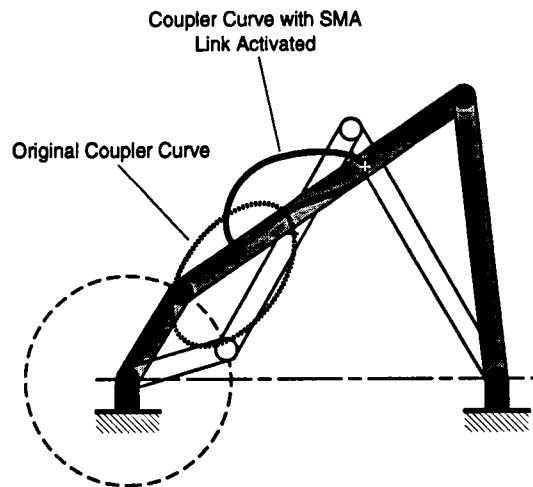


Figure 166. Using shape memory alloys for variable output actuation.

SMA devices have already proven their importance in dynamic systems by controlling vibrations in flexible beams. SMA unique characteristics include their Young's modulus of elasticity ( $E$ ), temperature relations, damping characteristics, and tunable  $E$  and recovery force. Tuning the properties or recovery stress requires heating or cooling, which may not be suitable for high-frequency vibration control but good for low-frequency control. The high damping capacity of the martensitic phase of SMA's can be used in passive vibration control. The hysteretic damping of pseudoelastic SMA's can be exploited in passive/adaptive vibration control. The  $E$  of high-temperature austenite is three to four times as large as that of low-temperature martensite. Shear modulus has the same relation. When Nitinol™ is heated to cause material transformation, stiffness is increased by a factor of 4 and the yield strength is increased by a factor of 10. Active mass dampers are already in use. They are generally tuned to the first fundamental frequency of the structure, and they are only effective when the first mode is the dominant vibration mode. Variable stiffness systems use active control to adjust structural stiffness so that resonant modes of a structure are steered away from dominant modes. [81]

SMA smart structures are typically fabricated with reinforced composite materials. SMA reinforced composites use the SMA as the reinforcing fiber, which can be easily embedded into a variety of flight vehicle composite materials and used to control structural response such as static deformation and vibration. Nitinol™ can be embedded in a graphite epoxy matrix. DuPont produces FP fibers for this application also. Active modal modification can be achieved by heating SMA fibers and changing the stiffness of all or portions of a flight control surface. Natural frequencies and mode shapes can be tailored by the fiber volume fraction and number of actuators. [84] Adaptive resonance tuning is achieved using SMA hybrid composite aerodynamic surfaces which are locally stiffened by selectively activating embedded SMA fibers, enabling the surface to attenuate vibration over several activation-dependent frequency ranges. These characteristics can be utilized in a controlled fashion to manipulate dynamic response. SMA's possess actuation, sensing, and control capabilities which can be used to provide adaptability to changing environmental conditions such as loads during ascent. It has been shown (Rogers 1990) that lower order (third through sixth) vibration modes can be successfully altered (first and second modes are more difficult to modify with global actuation but are being worked). Figure 167 [84] shows altered mode shapes. Fibers are activated at the center to change first and second mode shapes.

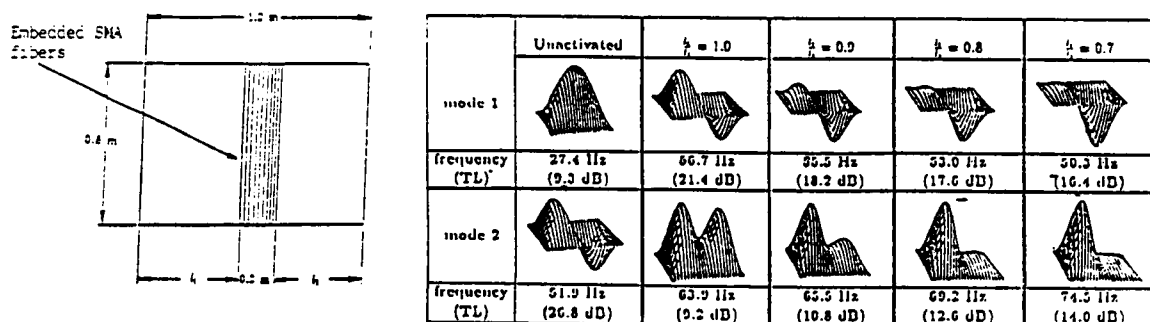


Figure 167. Shape memory alloy fibers for resonance tuning.

Aerospace applications include the Goodyear-developed space antennae. This wire hemisphere of SMA is crumpled into a tight ball less than 5 cm across. When heated above 77 °C, the ball opens up into its original shape, a fully formed antenna. This application has demonstrated the magnitude of deformation possible with SMA's. [85] A wire 30 mil in diameter can generate 75 lb of actuation force. By modulating the heating rate, the degree of control surface deflection can be controlled.

SMA springs as control elements have been used as force and displacement actuators. SMA hybrid composites are a class of materials which combine the strain recovery and stiffness transformation capabilities of SMA fibers with the structural characteristics of advanced composite materials and can produce smart flight control surfaces which can generate substantial aerodynamic forces and control torques. The origin of this concept actually originated from fish fins which have flexible structures that are actuated by contracting muscle fibers contained within a pliable skin. The Naval Warfare Center has used SMA's for adjustable camber in flight control surfaces. The concept employs SMA wires to control the flight of aerodynamic or hydrodynamic vehicles. An operational model has been fabricated and demonstrated. Lift measurements indicated 40% higher lift forces than rigid flight control surfaces at the same angles of attack ( $\alpha$ ). Shape memory alloys adjustable camber (SMAAC) control surfaces are a viable method of developing flexible control surfaces with variable camber and  $\alpha$ . The SMAAC control surface produces increased lift at a given  $\alpha$ . The SMAAC concept employs a flexible spring backbone which is activated by contracting SMA wires and is covered by a pliable elastomer skin to produce an adjustable control surface shape. Low A swept wings and fins have been studied. Self contained actuators require only electrical leads (Defense Advanced Research Projects Agency sponsored work). [81]

Some aircraft programs are nonarticulating flight control surfaces, where the appropriate regions at the edges of the relevant airfoil sections deform in order to provide

the aerodynamic control required. There are many programs in progress for smart wings to control dynamic response (Figs. 168 [83] and 169 [84]). This flight vehicle application features finite element control segments and incorporates sensors, actuators, and microprocessing capabilities. The sensors monitor chosen vehicle and environmental variables. The control strategy synthesizes the desired response by controlling the material characteristics of the actuators in the specific finite element segments. [84]

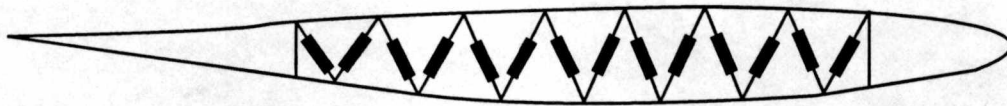


Figure 168. Use of adaptive truss for wing rib.

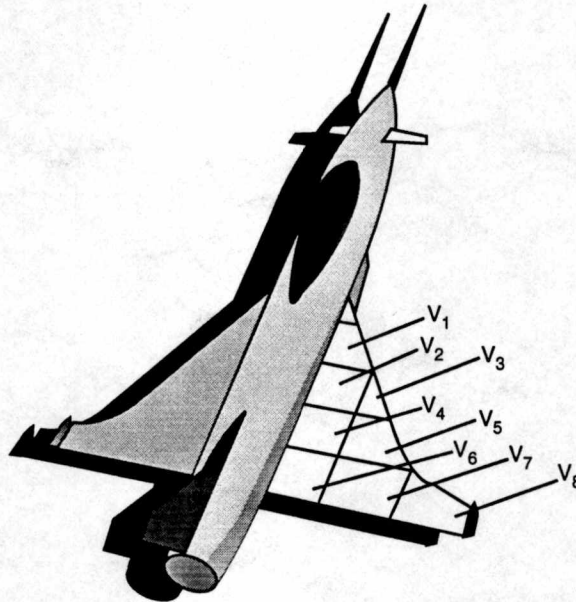


Figure 169. Active control of wing dynamic response using smart materials.

In flight vehicles, the employment of smart materials alleviates the need for the over-design situation based on the worst-case scenario of operating conditions which, of course, results in inferior performance at nominal conditions. Innovative FCA's could integrate sensing, processing, and actuator functions, and respond autonomously to acceleration flight loads, flutter, buffeting, wind, wind shear, and wind gusts. Larger wind

loads could be allowed and launch opportunities increased. Smart skins provide a dynamic interface with the external flight environment and allow all smart components to be embedded in the flight control surfaces, therefore, not affecting the aerodynamics of the surface. This allows the avionics design engineer greater freedom. Optimal control strategies could be employed to tailor the aerodynamic and aeroelastic characteristics of the flight controls to provide maximum control authority augmentation.

For example, a fiber-optic sensing system could instantaneously sense vibration and temperature, which would permit the microprocessor to optimally actuate the ER fluid, magnetostrictive, SMA, and PE domains in the flight control surfaces. The SMA actuators could be used for generating large changes, the PE actuators for changing surface geometries, and the ER and magnetostrictives to control vibration. SMA's allow large changes to geometry (Fig. 170 [87]). The flight control surface can be designed at the macromechanical level by selection of size, shape, type, number, and spatial distribution of the actuator. Table 20 shows a quantitative comparison of typical smart materials. [87]

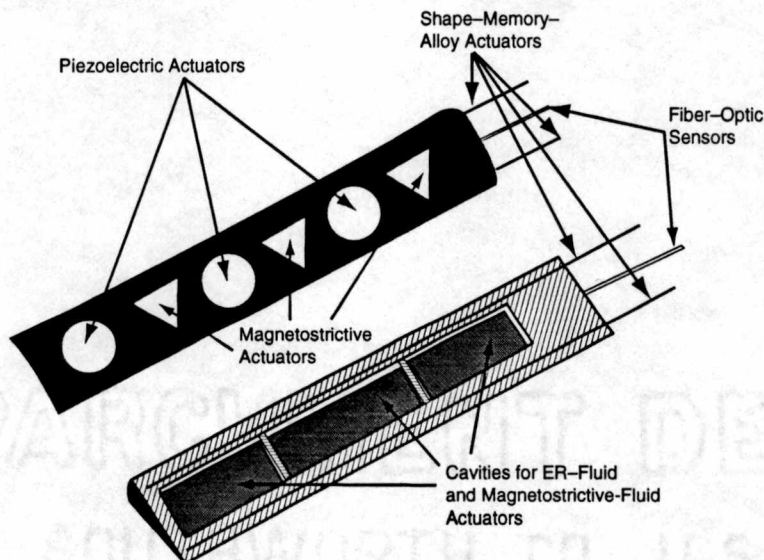


Figure 170. Smart material helicopter rotor blade.



Table 20. Comparison of smart materials.

|                                          | PZT G-1195<br>Piezoceramic | PVDF<br>Piezo Film | PMN-BA<br>Electrostr. | Nitinol™<br>Shp. Mem. | Terfenol-D™<br>Magnetostr. |
|------------------------------------------|----------------------------|--------------------|-----------------------|-----------------------|----------------------------|
| Max. strain (p/m)                        | 300                        | 300                | 600                   | 20,000 AC             | 1,800                      |
| E (lb/in <sup>2</sup> )×10 <sup>-6</sup> | 9                          | 0.3                | 17                    | 8                     | 7                          |
| T max (°C)                               | 360                        | 100                | high                  | 45                    | 380                        |
| Hysteresis (percent)                     | 10                         | >10                | <1                    | 5                     | 2                          |
| Bandwidth                                | kHz                        | kHz                | kHz                   | 1 Hz                  | 100 Hz                     |
| Temp. sen. (percent/°C)                  | 0.05                       | 0.8                | 0.9                   |                       | 0.3                        |

Cincotta et al. received a patent in May 1992 on an articulated control surface for flight control using SMA's. Beauchamp et al. received a patent in February 1993 on an articulated fin/wing control system using SMA's. [90, 91] See also References 92 and 93.

#### 10.10 Advanced Composites for Flight Controls

NASA's investment of more than \$89 million in the Advanced Composites Technology (ACT) Program, whose goal is "to make composites safe and affordable," has been instrumental in the development of many new flight control surface materials and fabrication techniques. One example of the new composite-part fabrication techniques involves near-net-shape preforms, made by applying textile weaving technologies to carbon or other high-strength fibers. Matrix material, such as low-viscosity epoxy, is then injected into the preform, and the part is cured to a shape requiring minimal machining. Lockheed will use braided textile preforms for braided composite helicopter parts. McDonnell Douglas is using through-the-thickness stitching of multilayer fabric for preforms, with subsequent injection of matrix resin, for major structural parts. By the turn of the century, they hope to certify a 200-passenger aircraft with a totally composite wing. The vehicle would be as much as 40% composites by weight if used in the fuselage also. [94] Many flight control surface composite materials which have become standard are being superseded.

Kevlar-49™, DuPont's trade name for an aramid fiber (poly(p-phenyleneterephthalamide)) made in China, has a tensile strength of 200 ksi and a strength to weight ratio of 101.8 and has been used on aircraft noses and wing LE's. Kevlar-49™ (Fig. 171 [95]) has the lowest specific gravity and highest tensile strength/weight ratio among current fibers in their class, with superior damage tolerance against impact or other dynamic loadings. Kevlar-49™ fibers do not melt or support combustion, have a very high vibration damping coefficient, are hygroscopic, and can absorb up to 6% moisture at 100% relative humidity and 23 °C, but will start to carbonize at 427 °C. The maximum long-term use temperature for Kevlar-49™ is about 160 °C. Boron fiber-reinforced epoxy skins have been used since 1969 on aircraft aerodynamic surfaces such as the F-14 horizontal stabilizer. Carbon/graphite fibers, widely used in aerospace, have tensile moduli of  $150 \times 10^6$  lb/in<sup>2</sup>, very low coefficients of thermal expansion, and high fatigue strengths. Carbon fibers, either alone or in hybridization with Kevlar-49™ fibers, have become the primary material for flight control surfaces, wings, and fuselages (Table 21). [95] A second-generation Kevlar™ fiber is Kevlar-149™, which has the highest

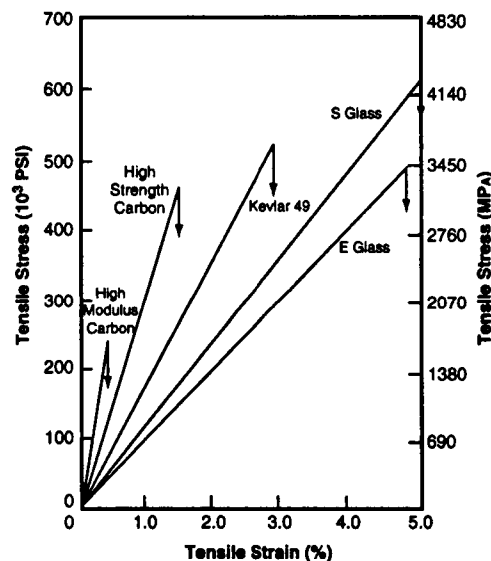
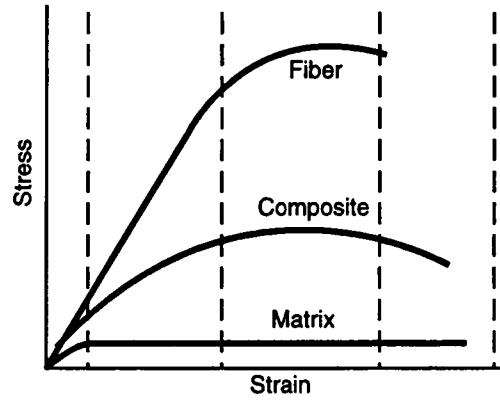


Figure 171. Tensile stress-strain for Kevlar-49™.

Table 21. Composite flight control applications.

| Military Aircraft       | Component                                                                            | Material           |                            | Overall Weight Savings Over Metal Parts (percent) |
|-------------------------|--------------------------------------------------------------------------------------|--------------------|----------------------------|---------------------------------------------------|
| F-14 (1969)             | Skin on the horizontal stabilizer box                                                | Boron fiber-epoxy  |                            | 19                                                |
| F-111                   | Underwing fairings                                                                   | Carbon fiber-epoxy |                            | 25                                                |
| F-15 (1975)             | Fin, rubber, and stabilizer skins                                                    | Boron fiber-epoxy  |                            |                                                   |
| F-16 (1977)             | Skins on vertical fin box, fin leading edge                                          | Carbon fiber-epoxy |                            | 23                                                |
| F/A-18 (1978)           | Wing skins, horizontal and vertical tail-boxes; wing and tail control surfaces, etc. | Carbon fiber-epoxy |                            | 35                                                |
| AV-8B (1982)            | Wing skins and sub-structures; forward fuselage; horizontal stabilizer; flaps        | Carbon fiber-epoxy |                            | 25                                                |
|                         |                                                                                      |                    |                            |                                                   |
| Commercial Aircraft     | Component                                                                            | Weight (lb)        | Weight Reduction (percent) | Comments                                          |
| Boeing 727              | Elevator facesheets                                                                  | 98                 | 25                         | 10 units installed in 1980                        |
| 737                     | Horizontal stabilizer                                                                | 204                | 22                         | Installed in 1973                                 |
| 737                     | Wing spoilers                                                                        | —                  | 37                         |                                                   |
| 757                     | Ailerons, rudders, elevators, fairings, etc.                                         | 3,340 (total)      | 31                         |                                                   |
| McDonnell Douglas DC-10 | Upper rudder                                                                         | 67                 | 26                         | 13 units installed in 1976                        |
| DC-10                   | Vertical stabilizer                                                                  | 834                | 17                         |                                                   |
| Lockheed L-1011         | Aileron                                                                              | 107                | 23                         | 10 units installed in 1981                        |
| L-1011                  | Vertical stabilizer                                                                  | 622                | 25                         |                                                   |

tensile modulus of all commercially available aramid fibers, and is 40% higher than Kevlar-49™. [95] The structural integrity and durability of these components have built up confidence in their performance and prompted much wider use (Fig. 172). [96]



| Material*       | Longitudinal modulus ( $E_{11}$ ),<br>$\text{In} \times 10^7$ ( $\text{m} \times 10^7$ ) | Transverse modulus ( $E_{22}$ ),<br>$\text{In} \times 10^7$ ( $\text{m} \times 10^7$ ) | Shear modulus ( $G_{12}$ ),<br>$\text{In} \times 10^7$ ( $\text{m} \times 10^7$ ) |
|-----------------|------------------------------------------------------------------------------------------|----------------------------------------------------------------------------------------|-----------------------------------------------------------------------------------|
| Borsic/aluminum | 35 (0.89)                                                                                | 20 (0.51)                                                                              | 10 (0.25)                                                                         |
| Boron/epoxy     | 43 (1.09)                                                                                | 4 (0.10)                                                                               | 1.4 (0.04)                                                                        |
| Carbon/epoxy    | 42 (1.07)                                                                                | 1.9 (0.05)                                                                             | 1.3 (0.03)                                                                        |
| Titanium        | 10 (0.25)                                                                                | 10 (0.25)                                                                              | 4 (0.10)                                                                          |

\* All composites, 50 vol percent fiber.

Figure 172. Composite moduli compared with titanium.

The damping properties of the material represent its ability to reduce the transmission of vibration and the resonance amplitudes of vibration (Table 22 [95]). Fiber-reinforced composites generally have higher damping factors than metals, and the coefficients of thermal expansion are much less, as can be seen in Table 23. [95] Flexural properties, such as flexural strength and modulus, are determined by loading a composite beam specimen of rectangular cross section in either a three-point or four-point bending mode (ASTM test method D790-81). The maximum fiber stress at failure on the tension side of a flexural specimen is considered the flexural strength of the material. Figure 173 shows flexural load deflections of typical unidirectional laminates. [95] The Kevlar-49™

has a highly nonlinear load-deflection curve due to compressive yielding. The flexural modulus is a critical function of the lamina stacking sequence, and therefore will not always correlate with the tensile modulus, which is less dependent on the stacking sequence. In angle-ply laminates, a bending moment creates both bending and twisting curvatures which influence the measure of flexural modulus. [95]

Table 22. Damping factors of representative laminates.

| Material      | Fiber Orientation           | Modulus ( $10^6$ lb/in <sup>2</sup> ) | Damping factor $\eta$ |
|---------------|-----------------------------|---------------------------------------|-----------------------|
| Mild steel    | —                           | 28                                    | 0.0017                |
| 6061 Al alloy | —                           | 10                                    | 0.0009                |
| E-glass-epoxy | 0°                          | 5.1                                   | 0.0070                |
| Boron-epoxy   | 0°                          | 26.8                                  | 0.0067                |
| Carbon-epoxy  | 0°                          | 27.4                                  | 0.0157                |
|               | 22.5°                       | 4.7                                   | 0.0164                |
|               | 90°                         | 1.0                                   | 0.0319                |
|               | [0/22.5/45/90] <sub>s</sub> | 10.0                                  | 0.0201                |

Table 23. Coefficients of thermal expansion of representative laminates.

| Material           | Coefficient of Thermal Expansion,<br>10 <sup>-6</sup> m/m per °C (10 <sup>-6</sup> in/in per °F) |                         |                              |
|--------------------|--------------------------------------------------------------------------------------------------|-------------------------|------------------------------|
|                    | Unidirectional (0°)                                                                              |                         | Quasi-isotropic              |
|                    | Longitudinal                                                                                     | Transverse              |                              |
| S-glass-epoxy      | 6.3 (3.5)                                                                                        | 19.8 (11)               | 10.8 (6)                     |
| Kevlar 49™-epoxy   | -3.6 (-2)                                                                                        | 54 (30)                 | -0.9 to 0.9<br>(-0.5 to 0.5) |
| Carbon-epoxy       |                                                                                                  |                         |                              |
| High modulus       | -0.9 (-0.5)                                                                                      | 27 (15)                 | 0 to 0.9<br>(0 to 0.5)       |
| Ultra-high modulus | -1.44 (-0.8)                                                                                     | 30.6 (17)               | -0.9 to 0.9<br>(-0.5 to 0.5) |
| Boron-epoxy        | 4.5 (2.5)                                                                                        | 14.4 (8)                | 3.6 to 5.4<br>(2 to 3)       |
| Aluminum           |                                                                                                  | 21.6 to 25.2 (12 to 14) |                              |
| Steel              |                                                                                                  | 10.8 to 18 (6 to 10)    |                              |
| Epoxy              |                                                                                                  | 54 to 90 (30 to 50)     |                              |

Note: The fiber content in all composite laminates is 60% by volume.

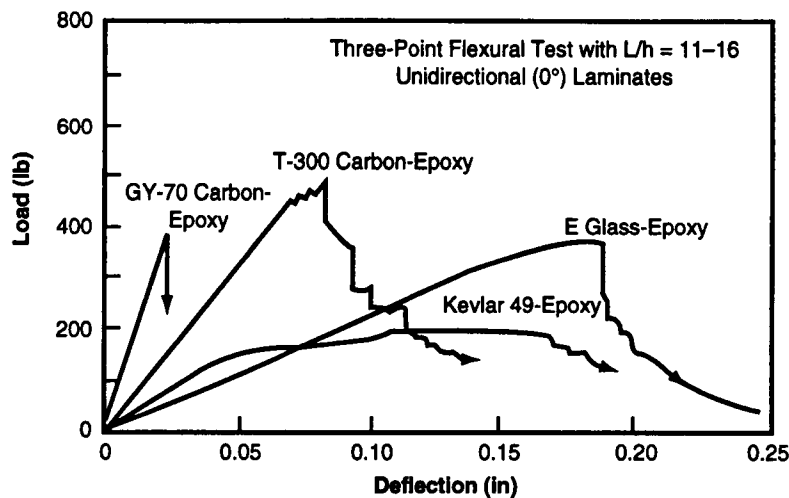


Figure 173. Load deflections for typical  $0^\circ$  unidirectional laminates.

Aerospace applications of structural composites usually use 60% fiber volume. Computer methods have been developed to optimized internal fiber arrangements in composite materials for minimum weight and maximum strength. [81] Substantial amounts of carbon and Kevlar-49<sup>TM</sup> fiber-reinforced composites have been used for some time on aircraft flight control surfaces. The Lear Fan 2100 business jet uses over 70% carbon and Kevlar-49<sup>TM</sup> for its structural weight. [95] Helicopter rotor blades have long used fiber-reinforced epoxies, not only for weight reduction, but also for their ability to tailor the dynamic frequencies. The critical flopping and twisting frequencies of the blade are controlled and tuned, not only by the classical methods of mass distribution, but also by varying the type, concentration, and distribution, as well as the orientation, of the fibers along the blade chord length. A great advantage of composites in helicopter blades is the manufacturing flexibility of the materials. The composite blades are molded into complex airfoil shapes with little or no additional manufacturing costs, while conventional metal blades are limited to shapes that can be extruded, machined, or rolled. Fabrication of large composite parts is done in autoclaves, large variable-pressure ovens for precision curing of epoxy polymers. Composite hydrofoil control flaps of Ti-clad

graphite/epoxy are used to withstand critical pressure loadings of 6,300 lb/ft<sup>2</sup>. [97] Polymeric composites such as polyimide or cyanate ester, reinforced by premium carbon fibers, have been used in aircraft flight control surfaces. When surface-treated with cold air plasma, the interlaminar shear strength is increased by 60%. [98] Hybrid composites use more than one type of fiber and/or more than one matrix.

Significant among high-temperature composites are matrix resins of polyimides and polyphenyl-quinoxaline with fibers of graphite and Kevlar™. These have been used in supersonic missiles and on fins at Mach 3 and at 800 °F (Fig. 174 [98]). Weight savings up to 67% have been achieved on missile fins. Composites are routinely selected in missile structures for weight reduction, which increases payload capability. Kevlar-49™ or S-glass reinforced epoxies have long been used in filament-wound missile motor cases, where the high strength/weight ratios for these materials allow reductions in motor case thickness resulting in additional weight savings. Kevlar-49™, acrylonitrile, allylamine, and “SB1” matrix are high in tensile strength and are thermally stable with good elastic moduli and low density.

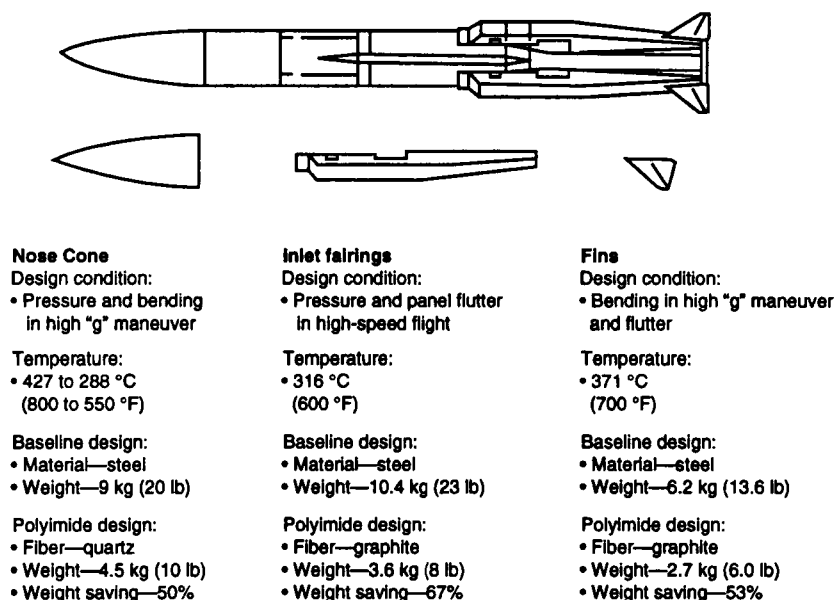


Figure 174. Supersonic missile and fins of polyimide matrix composites.

In many spacecraft applications, a major factor in selecting composites is their dimensional stability over a wide temperature range. Many carbon fiber-reinforced epoxy laminates can be designed to produce a coefficient of thermal expansion close to zero. While many aerospace alloys, such as Invar, have comparable coefficients of thermal expansion, the carbon composites have much lower specific gravities and higher strength, as well as a higher stiffness/weight ratio. This unique combination of mechanical properties has led to numerous applications in satellites where temperature ranges of  $-100$  to  $100\text{ }^{\circ}\text{C}$  must be accommodated. In the space shuttle, the composites save 3,000 lb per vehicle. Boron fiber-reinforced aluminum tubes are used for the midfuselage truss structure, Al honeycomb in combination with carbon fiber-reinforced epoxy for the payload bay doors, carbon fiber for the remote manipulator arm, and Kevlar-49™ for pressure vessels. [99]

Composite flight controls are used on many flight vehicles including the Airbus A310, Lockheed L-1011, McDonnell Douglas DC-10, F-15 (boron/epoxy), and Pegasus. Composite flight control surfaces are required to replace production metal surfaces in fit, form, function, and stiffness, and result in weight reduction. They must meet the fail-safe design criteria for limit flight loads in accordance with the Federal Aviation Administration (FAA) requirements and not degrade in their operating range of  $-54$  to  $82\text{ }^{\circ}\text{C}$  ( $-65$  to  $180\text{ }^{\circ}\text{F}$ ) or in high humidity conditions. The F-22 aircraft and the Pegasus LV use composite wing designs (Scaled Composites, Inc.). The DC-10 uses honeycomb sandwich skins, graphite/epoxy, and a 4-spar, 13-rib structure for vertical stabilizer and flight controls. Airbus was the first commercial aircraft to use composite primary structures. The vertical fin and control surfaces are carbon/epoxy. A well-known benefit of using composites is the elimination of fasteners. Example benefits can include 22% weight reduction and 95 parts vs. 2,076 parts. Boeing has saved 1,000 lb and achieved a smoother, drag-reducing skin surface by using carbon fiber/epoxy composites



on the primary structural components, the 47-ft tall vertical tailfin, and the horizontal stabilizers of the B-777 transport. This is the first American passenger airplane to have an all composite empennage.

The L-1011 uses graphite/epoxies in the vertical fin and flight controls. The L-1011 ailerons are a wedge-shaped, one-cell, box configuration consisting of a front spar, a rear spar, upper and lower covers, reinforcing ribs, LE shrouds, end fairings, TE wedge, shroud supports, and feedback, hinge, and actuator fittings. Inboard ailerons are located between the outboard and inboard flaps on each wing as shown in Figure 175 [95]. Figure 176 [25] shows a typical LV flight control surface for comparison. Each aileron is 92 inches in span, 50 inches in chord, and 10 inches in thickness.

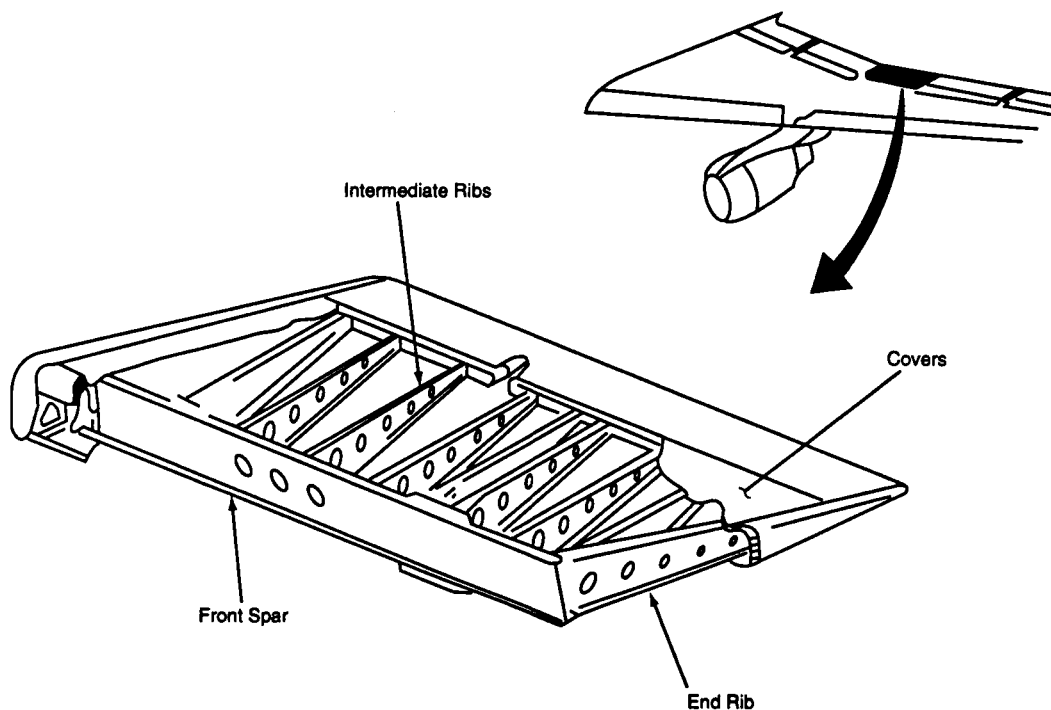


Figure 175. Construction of Lockheed L-1011 composite aileron.

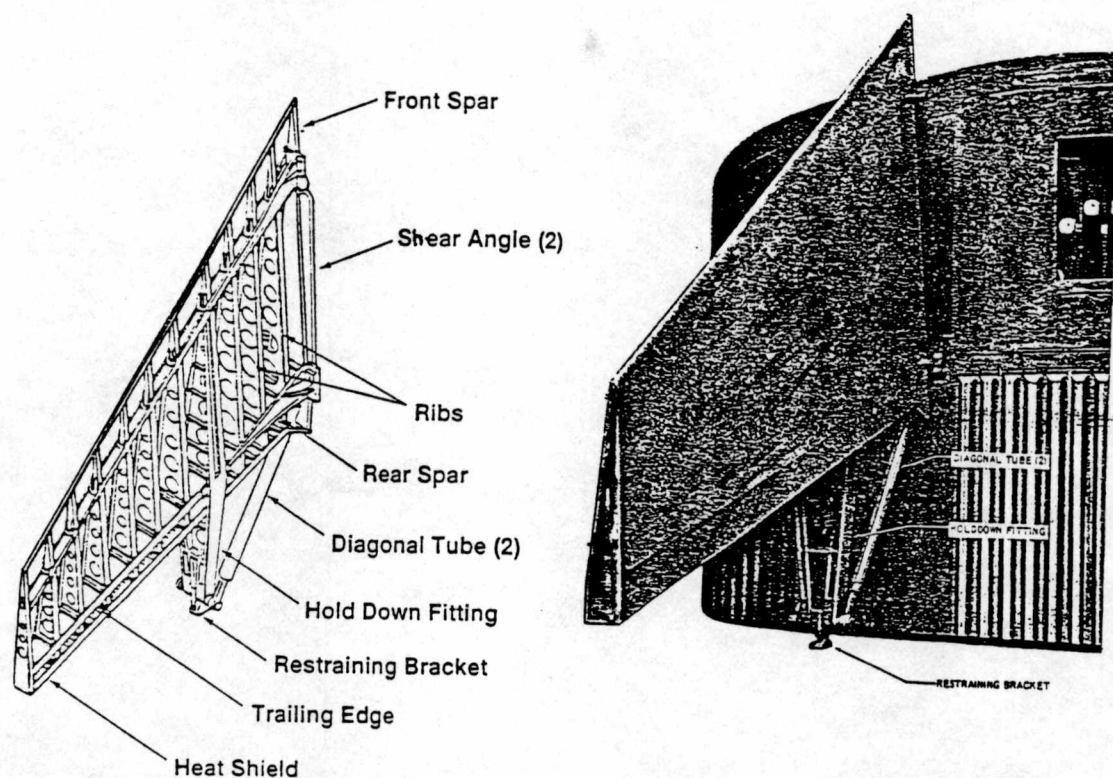


Figure 176. Typical LV aerodynamic surface (Saturn IB).

The following were selected for principal structural components of the composite ailerons. Upper and lower covers: T-300 carbon fiber epoxy sandwich construction of three layers; laminate configuration of 45/0/-45/syntactic core/-45/0/45 with the 0° plies oriented in the spanwise direction; near main rib and at ends of each cover five plies of T-300 oriented in the chordwise direction. Front spar: constant-thickness channel section of 45/0/-45/90/0 laminate; zero plies front spar in spanwise direction. Main ribs: used at three hinge or actuator locations for transferring loads from the fittings to the aileron covers and spars. In the aileron assembly, the upper cover, all ribs, and two spars are permanently fastened with titanium screws and stainless steel collars. The lower covers

are removable. All carbon-epoxy parts are painted with a urethane coat. These composite flight control surfaces are 23% lighter and contain 50% fewer parts (Table 24 [95]). Table 25 gives a summary of ground tests. [95]

Table 24. Comparison of composite and metal flight control surfaces.

|                     | Composite | Aluminum |
|---------------------|-----------|----------|
| Weight (lb)         | 100.1     | 140.4    |
| Number of ribs      | 10        | 18       |
| Number of parts     | 205       | 398      |
| Number of fasteners | 2,574     | 5,253    |

Table 25. Composite flight control surface ground tests.

|                                                                                                                    |                                                                                                                                                                                         |
|--------------------------------------------------------------------------------------------------------------------|-----------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------|
| Vibration in the flapping mode                                                                                     | Resonance frequencies comparable to those of metal ailerons                                                                                                                             |
| Vibration in the torsional mode                                                                                    | Resonance frequencies comparable to those of metal ailerons                                                                                                                             |
| Chordwise static bending stiffness                                                                                 | Composite ailerons 27% less stiff than metal ailerons                                                                                                                                   |
| Static torsional stiffness                                                                                         | Comparable to metal ailerons                                                                                                                                                            |
| Static loading                                                                                                     | 124% design ultimate load without failure at 12° down-aileron positions<br><br>139% design ultimate load at 20° up-aileron positions with postbuckling of the hinge and backup rib webs |
| Impact loading to cause visible damage at four locations followed by one lifetime flight-by-flight fatigue loading | Slight growth of damage (caused by impact loading) during the fatigue cycling                                                                                                           |
| Simulated lightning followed by static loading                                                                     | Burn-through and delamination over a small area; however, no evidence of growth of this damage during static testing                                                                    |

The B-777 composite consists of a high-strength carbon fiber containing toughened epoxy resin supplied by Toray Industries, headquartered in Tokyo. This Toray composite material was demonstrated to NASA engineers. The material's damage resistance is seven times that of conventional composites. The Toray flight control surface and empennage composite material, designated P3202, dents like metal when damaged so that visual inspection of flight control surfaces would be 100% reliable. P3202 contains Toray's own T-800 graphite carbon fiber in a matrix of 3900-2 epoxy. Epoxy is a thermosetting resin which, once cured, cannot be remelted and tends to be brittle. The P3202 gets its toughness from inherently more rubbery microparticles of thermoplastic material introduced between the thermo setting prepreg ply layers. [94]

### **10.11 Carbon-Carbon Composites**

Carbon-carbon composites (CCC's) are the most highly developed of the ceramic-ceramic composites, consist of reinforcing fibrous carbon substrate in a carbonaceous matrix, and are made in a variety of forms (Fig. 177) with tensile moduli up to  $100 \times 10^6$  lb/in<sup>2</sup>. [100] CCC's were developed to meet aerospace material needs of structural integrity at high temperature in excess of 5,000 °F and have been used extensively in missiles, exhaust nozzle throats and flaps, turbine blades, heat shields, reentry vehicles, and soon, pistons for internal combustion engines. A CCC piston for use in reciprocating engines has been successfully tested at LaRC, and a U.S. patent has been issued. 3-D CCC is an attractive material for nozzle throat and exit areas. 3-D weaving technology, as developed by Aerospatiale and Brochier, S.A. in France, has been licensed and transferred to U.S. firms because 3-D CCC components are replacing the weaker two-dimensional materials. CCC's are used on the space shuttle because of their high strength levels at temperatures over 1,650 °C and stiffness to resist flight loads and large thermal gradients. Ceramic matrix composites, slated for use on the high speed civil transport

(HSCT) engines, consist of inorganics such as silicon nitride or silicon carbide reinforced with high-strength ceramic fibers.

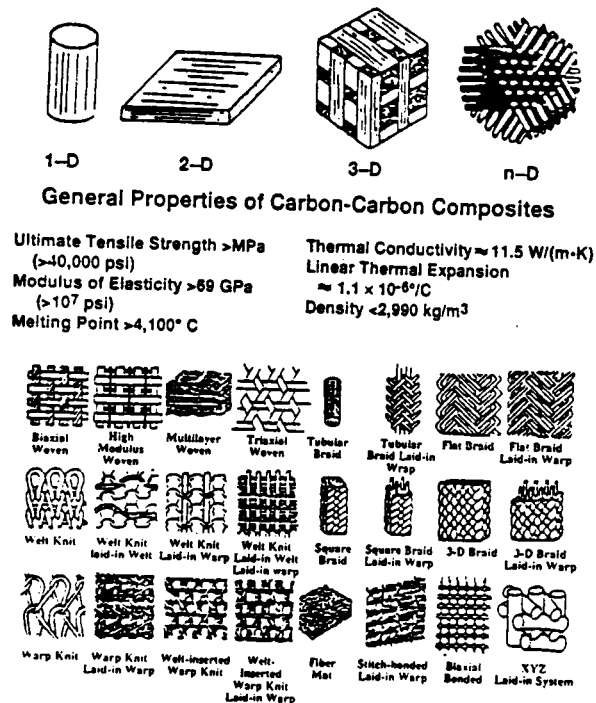


Figure 177. General properties and weaves of carbon-carbon composites.

Carbon fibers are 93 to 95% carbon, whereas the higher modulus graphite fibers are 99+% carbon. Graphite filaments have very high strengths and moduli which remain constant at high temperatures. Graphite fibers are the stiffest fibers known and approximately five times more rigid than steel. The fibers themselves are composites; only part of the carbon present has been converted to graphite in tiny crystalline platelets specially oriented with respect to the fiber axis. These are used extensively in the missile and aircraft industry (Fig. 178 [100]), hence are directly applicable to the LV FCA's.

CCC's have matured considerably since first used on spacecraft. The key to these advances is improved oxidation-resistant materials for atmospheric use at high temperatures. Although CCC's can withstand temperatures  $> 3,000^{\circ}\text{C}$  in a vacuum or

inert atmosphere, they oxidize when in an oxygen atmosphere at 600 °C. To use CCC parts in an oxidizing atmosphere, they must be compounded with materials that produce oxidation-protective coatings, or must be coated and sealed. On the space shuttle, the CCC LE's and nose cap are converted to silicon carbide in a high temperature diffusion-coating process. Improvements are being developed for oxidation protection of CCC components on the space shuttle. Figure 179 [100] shows three levels of CCC strength efficiency. The first, labeled space shuttle material, is the strength level of reusable CC (RCC).

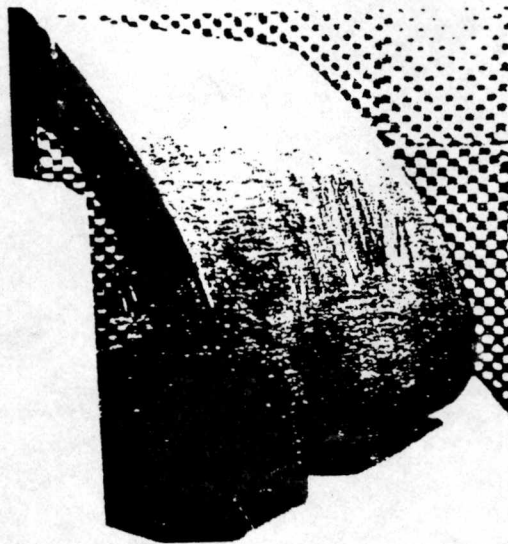


Figure 178. Space shuttle orbiter CCC wing LE

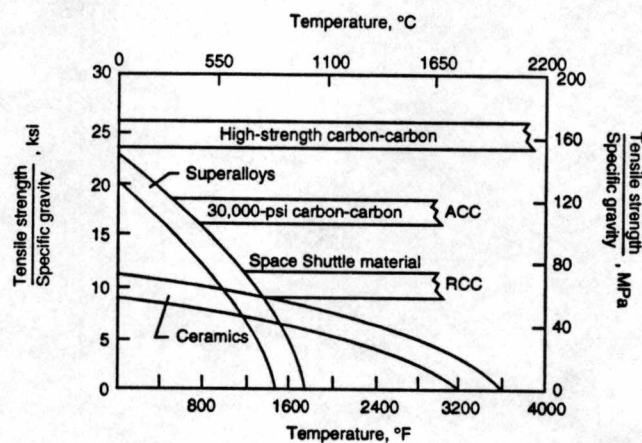


Figure 179. Carbon-carbon strength-to-density ratios.

Reusable CCC's are being used for fin LE's on missiles, wind tunnel test articles, and many flight vehicles including the National Aerospace Plane (NASP). Shuttle wing LE's and the nose withstand 1,540 °C. The LE's contain 22 CCC airfoil panels and 22 sealing strips of CCC. Reusable CCC on the space shuttle is protected from oxidation by a two-layer coating of a porous silicon carbide (Si-C) inner layer sealed with Si-O<sub>2</sub> and an outer coating of alkali silicate glass filled with Si-C particulate. Si-C fiber has been used as a reinforcement in ceramics for LE's of wings and control surfaces of hypersonic flight vehicles. Reusable CCC's have good strength up to temperatures of 2,760 °C. [100]

CCC has been chosen for flight control surfaces on the NASP. Continuous graphite fibers are combined in a 3-D geometry with a graphite matrix to create the CCC for the NASP wing LE's and for missile nose cones. [99] LaRC has sponsored this development effort. LTV Corp. is developing the CCC elevon control surfaces, and CCC will accommodate the 1,930 °C temperatures on the NASP wing LE's. The European spaceplane, Hermes, is also using CCC for the LE's. [100] Carbon-fiber reinforced plastic (CFR), which is used in the fin box for the Airbus A300/A310, has lowered total manufacturing costs by 85%. Shell assemblies, spars, and ribs of the fin box are stiffened by a modular grid (Fig. 180). [97] Each module is an Al core, around which the prepreg carbon fiber-reinforced plastic is wrapped. The skin is also laid up mechanically with reinforcements in the spar and rib areas.

Advanced CCC's (ACC's), which are twice as strong (Fig. 179), have been selected as the material to be used for the flight control surfaces, body flaps, chines, and fin LE's of the NASA Personnel Launch System, the HL-20. This ACC will withstand up to 3,300 °F. Carbon/boron nitride composite made from carbon fibers in a boron nitride matrix which can withstand temperatures twice as high as CCC's has been developed by the University of Illinois under Advanced Projects Research Agency and Naval Research support.

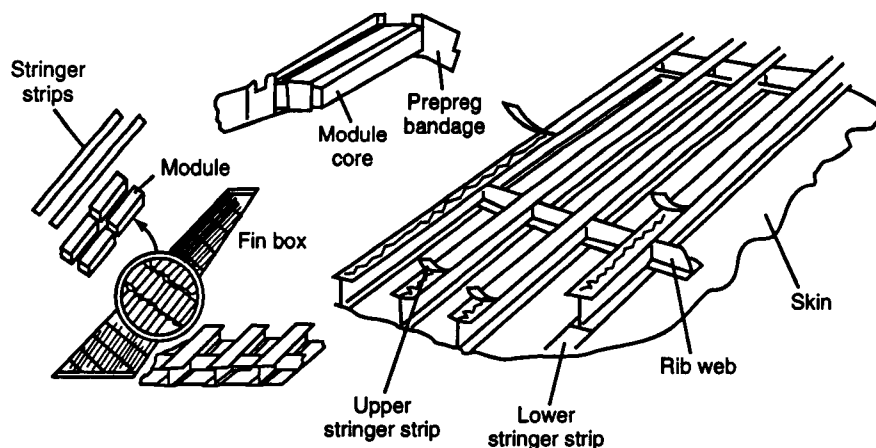


Figure 180. Carbon-fiber reinforced Airbus fin box.

### 10.12 Intermetallics

Intermetallics have a characteristic ordered crystal structure leading to high resistance to deformation and melting points  $> 1,650^{\circ}\text{C}$ . Ni-Al alloys have been used in 1,380 K temperature ranges on flight vehicle engines. Several intermetallics possess strength up to three times greater than that found in  $\text{Ni}_3\text{-Al}$  at comparable temperatures (Table 26). [89]

The Al-Li alloy has much aerospace usage, including the Centaur and Titan payload adapter, due to the 10% decrease in density and 10% increase in elastic modulus. Alcoa has recently developed a special new Al-Li, the C-155, for flight vehicle stabilizers and flight controls (*Aviation Week*, January 24, 1994). Lockheed's large-scale tests of Al-Li alloys have shown service life to be three to five times greater than conventional 2124-T851 Al plate. Vought Aircraft Company has explored various forms of laminated intermetallic. The fault of the intermetallic at ambient temperature is low ductility. Nb-Ti-Al is good at  $1,700^{\circ}\text{C}$  but brittle at room temperature. Ordered intermetallics (Fig. 181), [89] based on aluminides and silicides, have superior oxidation and corrosion resistance, good high-temperature strength and stiffness, and low density, but plagued by brittle fracture and poor ductility at ambient temperature. Examples are:  $\text{Ti-Al}_3$



(3.4g/cm<sup>3</sup>), Ti<sub>3</sub>-Al (4.2g/cm<sup>3</sup>), and Ti-Al-C. Titanium alloys such as  $\alpha$ ,  $\beta$  alloys, e.g., Ti-6Al-9V and metastable  $\beta$  alloys, e.g., Ti-10V-2Fe-3Al are used at temperatures over 500 °C. [95]

Table 26. Characteristics of selected intermetallics.

| Intermetallic Compound            | Crystal Structure (Ordered) | Melting Point °C (°F) | Density g/cm <sup>3</sup> (lb/in <sup>3</sup> ) | Young's Modulus GPa (10 <sup>3</sup> lb/in <sup>2</sup> ) |
|-----------------------------------|-----------------------------|-----------------------|-------------------------------------------------|-----------------------------------------------------------|
| Ni <sub>3</sub> -Al               | face centered cubic         | 1,390 (2,530)         | 7.50 (0.274)                                    | 178.5 (25.9)                                              |
| Ni-Al                             | body centered cubic         | 1,640 (2,980)         | 5.86 (0.214)                                    | 294.2 (42.7)                                              |
| Fe <sub>3</sub> -Al               | body centered cubic         | 1,540 (2,800)         | 6.72 (0.245)                                    | 140.0 (20.4)                                              |
| Fe-Al                             | body centered cubic         | 1,250 (2,280)         | 5.56 (0.203)                                    | 260.4 (37.8)                                              |
| Ti <sub>3</sub> -Al               | hexagonal close packaged    | 1,600 (2,910)         | 4.20 (0.153)                                    | 144.7 (21.0)                                              |
| Ti-Al                             | tetragonal                  | 1,460 (2,660)         | 3.91 (0.143)                                    | 175.6 (25.5)                                              |
| Al <sub>3</sub> -Ti               | tetragonal                  | 1,350 (2,460)         | 3.40 (0.124)                                    |                                                           |
| Nb <sub>2</sub> -Be <sub>17</sub> | rhombohedral                | 1,705 (3,100)         | 3.28 (0.120)                                    | 296.3 (43.0)                                              |
| Al <sub>3</sub> -Nb               | tetragonal                  | 1,600 (2,910)         | 4.54 (0.166)                                    |                                                           |

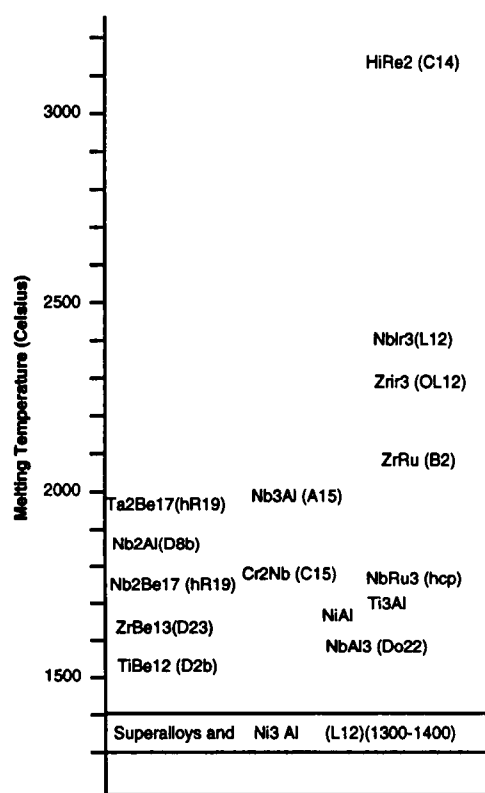


Figure 181. Melting points of example intermetallics compared to superalloys.

Intermetallics have been improved in ductility by incorporating ductile particles of other metals such as Nb, e.g., Ti-14Al-21Nb ( $\alpha$ -2) which can be used up to 980 °C, and  $\beta$  stabilizers such as Nb, V, and Mo have improved the Ti<sub>3</sub>-Al alloys with 10 to 25% Nb addition to promote ductility, thus allowing the fabrication of complex shapes, Ti-24Al-9Nb-2V, Ti-14Al-21Nb ( $\alpha$ -2), Ti-14Al-19Nb-3V-2Mo(super  $\alpha$ -2), and Ti-33Al-6Nb-1.4Ta( $\gamma$ ) being among the best. Ti-6Al-4V has UTS of  $200 \times 10^3$  lb/in<sup>2</sup>. Tensile strength and ductility of Ti-6Al-TiC is greatly affected by microstructure. Fast cooling rates in formation generate higher strength. Adding third elements, such as 2% by weight of Mn or V (Ti-30Al-20V) to Ti-Al-based alloys improves strength and ductility. This improvement is recognized as the result of refinement of the microstructure of grains, which reduces the slip length of dislocations that correspond to increased strength and ductility. A  $\gamma$ -Ti-Al (49% Ti, 47% al, 27% Cr, 2% N<sub>o</sub>) has good strength to 1,400 °F (developed and tested by General Electric in aircraft engines in 1994).

Recent advances in material science have shown that Ti-Al-base intermetallics can be used as high temperature aerospace airframe structural material. [98] They exhibit good stiffness combined with adequate creep and oxidation resistance at temperatures of 1,000 °C. An advantage with Ti intermetallics, e.g., Ti-6Al-4V, is the relatively simple application of superplastic forming and diffusion bonding (Fig. 182 [89]) which allows forming and welding in one step. Aluminum alloys do not exhibit superplastic properties following a conventional production technique. Many titanium aluminides have been slated to be used in the NASP, and Figure 183 [89] shows their comparison with composites.

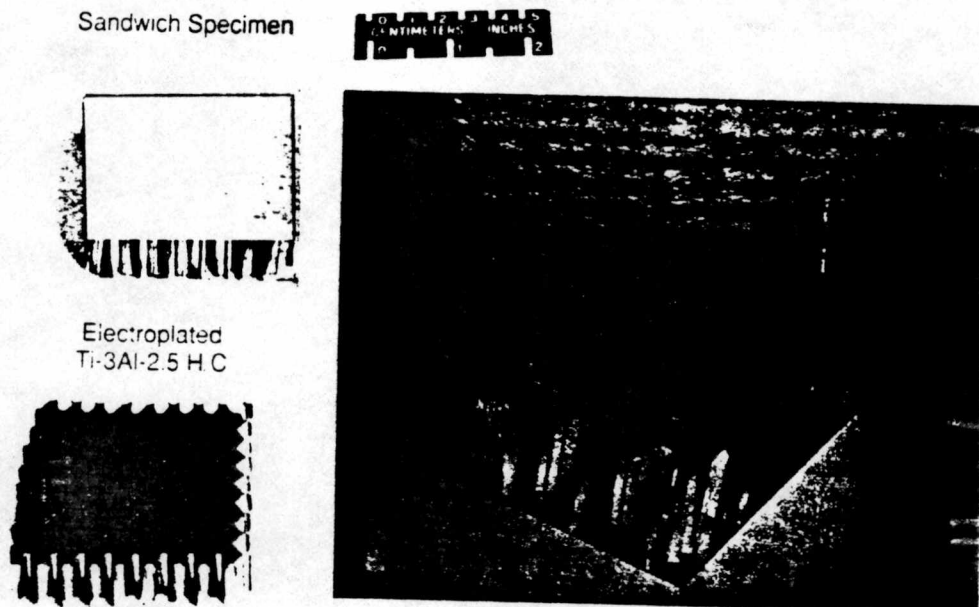


Figure 182.  $\text{Ti}_3\text{-Al}$  honeycomb structure using diffusion bonding.

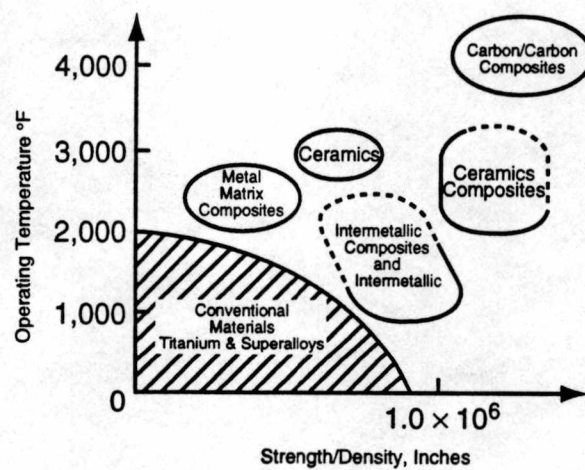


Figure 183. Comparison of intermetallics and composites.

As a result of extensive development programs in both the U.S. and Russia, the intermetallic titanium aluminides, both in composite and monolithic configurations, have significant flight vehicle usage. [89] Since materials are the main driver for the HSCT airframe, NASA is developing a new generation of intermetallic matrix composites for the HSCT. LaRC is researching the prevention of oxidation and embrittlement in Ti-Al intermetallics by the use of a 10- $\mu$ m thick Al-Si-B<sub>x</sub> coating. Intermetallics are known for excellent high-temperature properties, and their composites are excellent candidate materials for flight control surfaces of hypersonic/supersonic vehicles, including the LV application considered in this work.

### **10.13 Metal Matrix Composites**

While conventional composites have many proven applications in flight vehicle control surfaces, they can suffer from fatigue loads at very high temperatures. Typical upper temperature limits on conventional composites are 800 °F. Boron fibers offer excellent resistance to buckling and high compressive strength in addition to extremely high tensile moduli, but they are temperature limited. The metal matrix composite (MMC) has been created to solve this problem. MMC's offer high electrical and thermal conductivity in combination with mechanical strength.

MMC materials benefit from property synergisms which offer greater strength and stiffness, higher temperature capability, lower weight, and better fatigue behavior and wear resistance than the polymer matrices. MMC's can also have strength/weight ratios higher than those of intermetallics and can offer 60% savings in weight while still retaining key properties. In the last decade, there has been accelerated research in MMC development, because materials with high strength/weight ratios and high temperature capability are needed for hypersonic flight vehicles. MMC's, based on the principle that the sum is greater than the parts, offer revolutionary new choices in material selection for innovative flight control surfaces. Tailoring the properties of MMC's to specific

applications has motivated extended research over the past two decades. [96] New technology has allowed the manufacture of a plethora of new MMC's that have superior properties to their monolithic counterparts and are being used with increasing frequency in aerospace applications. [89]

The advantages of reinforcing metals with ceramic particles to produce MMC's are well known. New generations of MMC's have been produced with enhanced elevated temperature properties and casting, extruding, and forging capability. Aluminum oxide has excellent strength retention up to 1,370 °C. The modulus of metals is an order of magnitude higher than that of polymers (Tables 27 and 28). [95]

Table 27. Typical reinforcements used in metal matrix composites.

| Reinforcement   | Manufacture (Designation) | Form Mono/Tow | Diameter $\mu\text{m}$ (Mils) | Strength GPa (ksi)   | Modulus GPa (Msi) | Elong. Ult % | Density g/cc (lb/cu in) | CTE ppm/K | Thermal Cond. W/mK | Cost-\$/lb  | Production Status- lb/yr                     |
|-----------------|---------------------------|---------------|-------------------------------|----------------------|-------------------|--------------|-------------------------|-----------|--------------------|-------------|----------------------------------------------|
| Tungsten        |                           | Mono          | variable                      | 2.41 (350) variable  | 406 (59)          | High var     | 19.30                   | 5         |                    | 200         | Special order                                |
| Molybdenum      |                           | Mono          | variable                      | 1.03 (150)           | 365 (53)          |              | 10.20                   | 5.8       |                    | 200         | Special order                                |
| Carbon-Fib      | Amoco-Thor-P35            | Tow           | 10, 11 $\mu\text{m}$          | 1.9/1.72(275/250)    | 380 (415 (55/60)) | 0.50         | 2.00                    | -1.3      | 120/1101           |             | Commercially available in limited quantities |
|                 | Thermal-P75               | Tow           | 10 $\mu\text{m}$              | 2.1-1.9 (300-275)    | 520 (75)          | 0.40         | 2.00                    | -1.4      | 185                |             | -                                            |
|                 | Thermal-P100              | Tow           | 10 $\mu\text{m}$              | 2.2 (325)            | 724 (105)         | 0.31         | 2.15                    | -1.45     | 520                |             | -                                            |
|                 | Thermal-P130              | Tow           | 10 $\mu\text{m}$              | 2.2 (325)            | 827 (120)         | 0.27         | 2.18                    | -1.45     | 640                |             | -                                            |
|                 | DuPont-E-55               | Tow           | 9.5 $\mu\text{m}$             | 3.2 (465)            | 378 (55)          | 0.74         | 2.14                    |           |                    |             | -                                            |
|                 | DuPont-E-75               | Tow           | 9.4 $\mu\text{m}$             | 3.1 (450)            | 516 (75)          | 0.56         | 2.16                    |           |                    |             | -                                            |
|                 | DuPont-E-105              | Tow           | 9.3 $\mu\text{m}$             | 3.3 (480)            | 724 (105)         | 0.55         | 2.17                    |           |                    |             | -                                            |
|                 | DuPont-E-120              | Tow           | 9.2 $\mu\text{m}$             | 3.4 (500)            | 894 (120)         | 0.55         | 2.19                    |           |                    |             | -                                            |
|                 | DuPont-E-130              | Tow           | 9.2 $\mu\text{m}$             | 3.9 (570)            | 894 (130)         | 0.55         | 2.19                    |           |                    |             | -                                            |
|                 | Tonen-J1M                 | Tow           | 9.7 $\mu\text{m}$             | 2.96 (430)           | 489 (70)          | 0.61         | 2.14                    |           |                    |             | -                                            |
|                 | Tonen-U11M                | Tow           | 9.7 $\mu\text{m}$             | 2.96 (430)           | 699 (100)         | 0.43         | 2.17                    |           |                    |             | -                                            |
|                 | Mitsubishi-K133           | Tow           | 10 $\mu\text{m}$              | 2.19 (314)           | 447 (64)          | 0.53         | 2.08                    |           |                    |             | -                                            |
|                 | Dialed-K135               | -             | 10 $\mu\text{m}$              | 2.58 (371)           | 553 (79)          | 0.47         | 2.10                    |           |                    |             | -                                            |
|                 | Dialed-K137               | -             | 10 $\mu\text{m}$              | 2.70 (386)           | 650 (93)          | 0.42         | 2.12                    |           |                    |             | -                                            |
|                 | Dialed-K139               | -             | 10 $\mu\text{m}$              | 2.79 (400)           | 747 (107)         | 0.37         | 2.14                    |           |                    |             | -                                            |
| Carbon-Fan      | Hercules IM7              | -             | 5 $\mu\text{m}$               | 5.31 (770)           | 301 (43.6)        | 1.81         | 1.78                    |           |                    | 20          | Commercial                                   |
| Boron           | Testron                   | Mono          | 100-140 $\mu\text{m}$         | 3.52 (510)           | 400 (58)          | 0.89         | 2.60                    | 4.5       |                    | 600         | 30,000                                       |
| Silicon Carbide | Testron                   | Mono          | 75 & 140 $\mu\text{m}$        | 3.79 (543)           | 400 (59)          | 0.94         | 3.00                    | 4.2       | 200                | 2,500-5,000 | 1100-5 T F O                                 |
| SiC             | B.P. Sigma                | Mono          | 100 $\mu\text{m}$             | 3.45 (500)           | 414 (60)          | 0.83         | 3.10                    | 4.2       | 200                |             | (Experimental)                               |
|                 | DuPont                    | -             | 30 $\mu\text{m}$              | 1.17 (170)           | 379 (55)          | 0.31         | 3.20                    | 4.2       | -200               |             | Experimental                                 |
| Alumina         | DuPont-FF                 | Tow           | 20 $\mu\text{m}$              | 1.59 (230)           | 379 (55)          | 0.42         | 4.00                    | 5.6       |                    | 200         | 500                                          |
|                 | PRD-166                   | Tow           | 20 $\mu\text{m}$              | 2.28 (330)           | 379 (55)          | 0.60         | 4.20                    | 6.8       |                    | 300         | ~500                                         |
|                 | Sunlomo 85% Alumina       | Tow           | 17 $\mu\text{m}$              | 1.79 (280)           | 207 (30)          | 0.93         | 4.00                    | 4.5       |                    | 300         | 90,000                                       |
|                 | 3M-Alpha Alumina          | Tow           | 10-20 $\mu\text{m}$           | -2.06-2.76 (300-400) | 379 (55)          | 0.55-0.73    | 0.40                    |           |                    |             | Experimental                                 |
|                 | Saphikon                  | Mono          | 75-150 $\mu\text{m}$          | 2.07 (300)           | 448 (65)          | 0.46         | 4.00                    |           |                    | 100,000     | Experimental                                 |

Table 28. Mechanical properties of some metal matrix composites.

| Material                                 | Specific Gravity | Tensile Modulus, GPa (Gsi) | Yield Strength, MPa (ksi) | UTS, MPa (ksi) | Elongation percent | CTE $10^{-6}/^{\circ}\text{C}$ | Melting point, $^{\circ}\text{C}$ |
|------------------------------------------|------------------|----------------------------|---------------------------|----------------|--------------------|--------------------------------|-----------------------------------|
| Aluminum alloys:                         |                  |                            |                           |                |                    |                                |                                   |
| 2024-T6                                  | 2.78             | 70<br>(10.1)               | 468.9<br>(68)             | 579.3<br>(84)  | 11                 | 23.2                           |                                   |
| 6061-T6                                  | 2.70             | 70<br>(10.1)               | 275.9<br>(40)             | 310.3<br>(45)  | 17                 | 23.6                           |                                   |
| 7075-T6                                  | 2.80             | 70<br>(10.1)               | 503.5<br>(73)             | 572.4<br>(83)  | 11                 | 23.6                           |                                   |
| 8009                                     | 2.92             | 88<br>(12.7)               | 407<br>(59)               | 448<br>(64.9)  | 17                 | 23.5                           |                                   |
| 380<br>(as cast)                         | 2.71             | 70<br>(10.1)               | 165.5<br>(24)             | 331<br>(48)    | 4                  | —                              | 540                               |
| Titanium alloys:                         |                  |                            |                           |                |                    |                                |                                   |
| Ti-6Al-4V<br>(solution treated and aged) | 4.43             | 110<br>(16)                | 1,068<br>(155)            | 1,171<br>(170) | 8                  | 9.5                            | 1,650                             |
| Magnesium alloys:                        |                  |                            |                           |                |                    |                                |                                   |
| AZ91A                                    | 1.81             | 45<br>(6.5)                | 158.6<br>(23)             | 234.5<br>(34)  | 3                  | 26                             | 650                               |
| Zinc-aluminum alloys:                    |                  |                            |                           |                |                    |                                |                                   |
| ZA-27<br>(pressure die-cast)             | 5                | 78<br>(11.3)               | 370<br>(43.6)             | 425<br>(61.6)  | 3                  | 26                             | 375                               |

In the MMC, the matrix is a key element which is chosen to transfer stresses between the fibers, providing a barrier against an adverse environment, and protecting the surface of the fibers from mechanical abrasion (Fig. 184 [101]). The matrix plays a minor role in the tensile load-carrying capacity of a composite structure. However, selection of a matrix has a major influence on the interlaminar shear as well as the in-plane shear properties. Properties parallel to the fiber direction are dominated by the fiber, while those perpendicular to the fiber are dominated by the matrix. Therefore, the plies, orientation, and location are carefully designed. The interlaminar shear strength is an important design consideration for bending loads, whereas the in-plane shear strength is important under torsional loads. The matrix provides lateral support against the

possibility of fiber buckling under compression loading, thus influencing to some extent the compressive strength. The interaction between fibers and matrix is also important in designing damage-tolerant structures. The metallic matrices have been selected for high temperature applications, with Al, Ti, and Mg being the most common. These metallic matrix materials require high modulus fibers because their own moduli are high,  $10 \times 10^6$ ,  $16 \times 10^6$ , and  $6.5 \times 10^6$  lb/in<sup>2</sup>, respectively. These are combined with boron, silicon, and graphite. Intermetallic aluminides and silicates are attractive matrix materials due to their resistance to high temperature oxidation. The service temperature of a composite depends on the melting point of the matrix material. MMC's can be used in temperature ranges from cryogenic to 5,000 °F. [101]

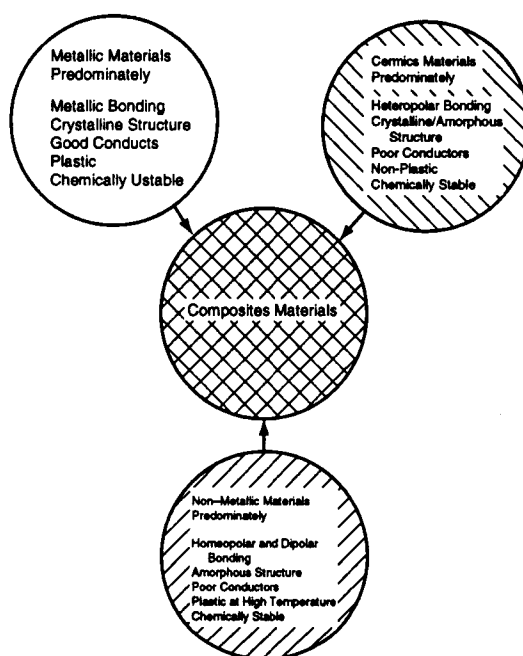


Figure 184. Composite groups.

The reinforcing materials supply the basic strength of the MMC. They can, however, also conduct heat or electricity. If a reinforcement is to improve the strength of a given matrix, it must be both stronger and stiffer than the matrix. Developments of new

filaments and whiskers give the engineer freedom to design the material to suit the needs. Chemical interactions between fibers and matrix components and interdiffusional phenomena must be considered to prevent fiber weakening, coarsening, or dissolution. MMC reinforcements include continuous or discontinuous filaments, whiskers or particulates, randomly oriented or aligned, and dispersion-strengthened, continuous, or eutectic phases. The dimensions of the reinforcements can vary from the submicrometer whiskers in situ composites to fibers of 100  $\mu\text{m}$  diameter. [102] Ceramic fibers are notable for their high-temperature applications in metal and ceramic matrix composites. Their melting points are close to 3,000 °C. Much of the recent work on MMC's is based on silicon carbide whiskers or particulates. Fibers are the most important class of reinforcements due to their ability to transfer strength to matrix materials. Table 29 [89] shows the desirable characteristics of a reinforcing fiber. Among fibers which have been used in the aircraft industry because of high temperature capability are carbon and graphite used to reinforce Al. Figure 185 shows how the Si-C MMC's compare to the intermetallics. [89]

In comparison to the intermetallics, the high cyclic fatigue strength of the MMC is superior, and they possess the advantage of high strength/weight ratio and designability beneficial to aerospace vehicles. Boron-aluminide composite is a MMC which has high strength/weight ratio, stiffness-to-density ratio, excellent compression strength, and is used on the space shuttle. Composite laminated plates and stiffened panels are typically used on flight vehicles. The materials can be aeroelastically tailored for vibration

Table 29. Desirable characteristics of reinforcing fiber.

- |    |                                                                                           |
|----|-------------------------------------------------------------------------------------------|
| 1. | Compatible coefficient of thermal expansion (CTE) with the matrix                         |
| 2. | Chemical compatibility with the matrix even after extended elevated temperature exposures |
| 3. | Low density                                                                               |
| 4. | High modulus of elasticity                                                                |
| 5. | High strength at elevated temperature                                                     |
| 6. | Good oxidation resistance                                                                 |
| 7. | 50 to 250 $\mu\text{m}$ diameter continuous monofilament                                  |
| 8. | Mass production feasibility (including affordable cost).                                  |



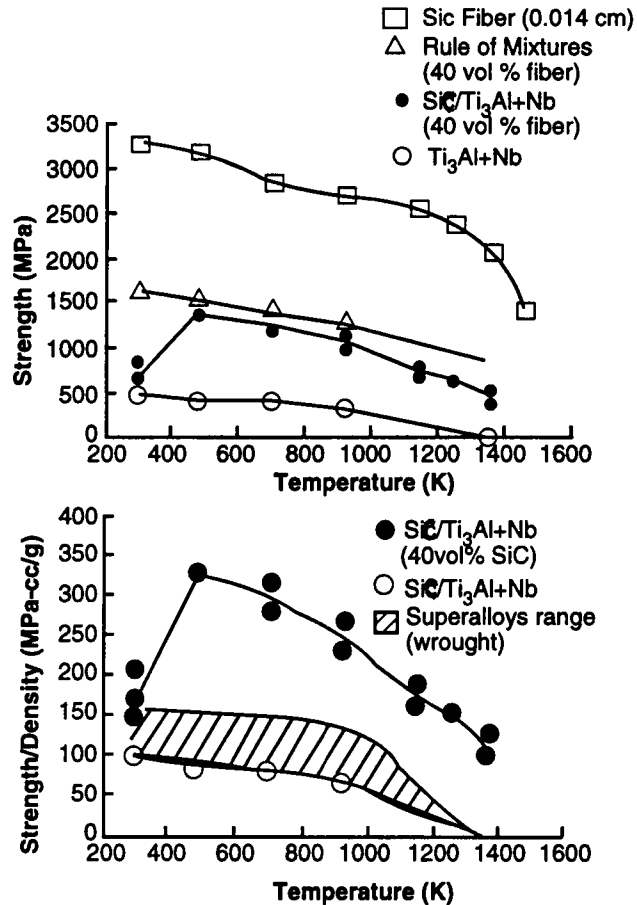


Figure 185. Comparison of metal matrix composite and corresponding intermetallic.

control. [98] The dynamic behaviors, including natural frequencies, mode shapes, and specific damping can be designed by means of changing the orientation of the layers (e.g., 0°, 90°, +45°, -45°, etc.) and the order and numbers (e.g., 1 to 30) of the layers. Strain energy is a convenient parameter to use in defining the damping of a composite. It has been used in damping analysis of fiber reinforced composites. Approaches for damping analysis include modal strain energy, complex eigenvalue method, and direct frequency response. [89] Figure 186 shows 238 nodes, 714 degrees of freedom, 312 composite bars, 152 metallic bars, 99 metallic half web elements, 209 composite half web elements, 8 metallic triangular elements, and 18 composite triangular elements. [98]

Thickness of a Lamina:  $t_0 = 0.125$  mm  
 Laminae in  $0^\circ, 90^\circ, 2 \times (\pm 45^\circ)$

1#, 2#, 3#, ..., 15# – Displacement  
 Constraint Points

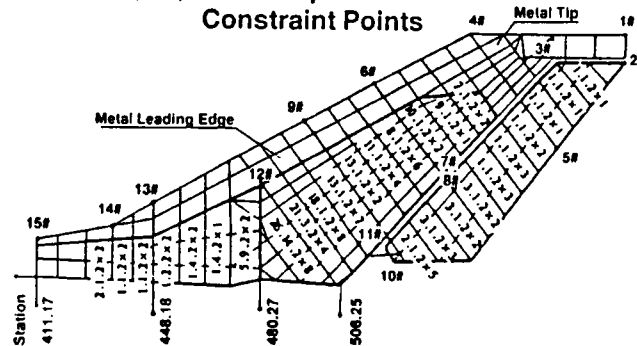


Figure 186. Composite laminae of aerodynamic surface and flight control.

MMC's are increasingly used in flight vehicle primary structures which are subjected to severe loading and environment, and would be well suited for LV FCA's. Lockheed has used Textron's MMC's for vertical stabilizer aerodynamic surfaces. MMC missile fins have been successfully tested. [99] For the application considered in this research, the best type of continuous reinforcement fibers for intermetallic titanium aluminides are Si-C and  $Al_2O_3$ . Of all the composite materials surveyed in this research effort, the MMC is best suited for flight control applications on LV's and hypersonic aircraft.

#### 10.14 Titanium Matrix Composites

The rapid development of fiber reinforced MMC's during recent years is closely related to the successful development of many protective coatings suitable for different operating conditions and reactive environments. MMC's have been developed for high-temperature applications with the titanium matrix composite (TMC) in the lead. Continuous Si-C fibers reinforce the Ti matrix. The filaments sustain the primary load, while the metal handles the shear load. Although CCC's can withstand temperatures up to

4,000 °F, they lack the strength of the TMC. TMC has the strength of steel at half the weight, can withstand temperatures up to 2,000 °F, and can produce synergistic advantages over their base elements. TMC's can also use metal fabrication processes such as brazing. Ti<sub>3</sub>-Al-based matrices reinforced with Si-C are a class of high service temperature TMC with attractive moduli, and demonstrate clear advantages over monolithic Ti aluminides and excellent suitability for LV flight control surface applications.

As part of this research, the author spent several days at Textron Specialty Materials Company in Lowell, MA, to study the manufacture of TMC and conduct a design review on the designed FCA's. Textron produces a range of high-temperature materials for aerospace applications, and has the world's first plant which became operational in 1993 (Government funded) built solely for the production of TMC for the NASA NASP. The NASP wing LE's, which must withstand over 2,500 °F, will use their CCC material. The  $\alpha$ -2 Ti aluminides slated for NASP usage save up to 75% in weight and operate at temperatures of 1,700 °F. One Textron TMC, the SCS-6/Ti-15V-3Cr-3Al-3Sn, has successfully passed a series of NASP flight test simulations on the one-third scale NASP fuselage that has been built and cycled to skin temperatures of 1,300 °F under loads. Textron has fabricated 4- by 8-ft TMC skin panels and associated TMC stiffeners for the NASP. (Ti-14Al-21Nb has been the best  $\alpha$ -2 previously available.)

A newly developed  $\beta$ -21S surpasses this in oxidation resistance. The  $\beta$ -21S has been selected for the NASP fuselage, wings, and flight control surfaces. The  $\beta$ -21S TMC has very good ductility and rolls easily into a 4- to 5- mil thick foil for compositing. While oxidation weakens many  $\beta$  alloys, the 21S proves to be an exception. Textron's MMC's have been used for flight control surfaces by Lockheed, and missile fins of Textron MMC have been successfully tested. [99]

Processes such as spray deposition and vapor deposition involve less severe or prolonged high temperature metal/ceramic contact. The present situation for Ti is that no ideal ceramic reinforcement has yet emerged. Si-C is apparently far from ideal, as the reaction layer both forms relatively quickly and has a markedly deleterious effect on composite properties. However, Si-C is the most readily available of the candidate long fiber reinforcements, which are needed for good creep resistance. This has resulted in production of coated Si-C fibers that delay or prevent the deleterious interfacial reaction. Several techniques are available for deposition of thin coatings on long fibers, including chemical vapor and physical vapor deposition. [103]

At the heart of the TMC is a continuous Si-C fiber that reinforces the Ti matrix. Textron, the only U.S. company producing these fibers, has developed a family of continuous Si-C filaments, the SCS-6 fiber family, with a coating developed to reinforce the Ti matrix. These fibers are fabricated by chemical vapor deposition (CVD) utilizing a heated carbon monofilament core spun from coal-tar pitch that acts as a nucleation site for deposition of the  $\beta$  Si-C material. Each core filament is drawn through a glass-tube reactor, where electrodes contact the fiber and heat it by electrical resistance. Silane and hydrocarbon vapors enter the reactor and decompose, depositing Si-C on the fiber as it passes through. Because of the extreme reactivity of the Ti matrix with Si-C at temperatures greater than 1,600 °F, a coating must be used on the Si-C fiber. The fiber itself is a complicated composite, consisting of many different layers of  $\beta$  Si-C deposited on the carbon core with different carbonaceous coatings covering the Si-C layers. A Si-C fiber cross-section showing the various deposition zones including the unique surface zone that provides superior tailored matrix bonding is shown in Figure 187. [99] For the detailed structural and chemical aspects of each layer of this fiber, the reader is referred to X.J. Ning and P. Pirouz, *Journal of Materials Research*, vol. 6, No. 10, October 1991. For fiber processing details, the reader is referred to P.R. Hoffman, Textron President,

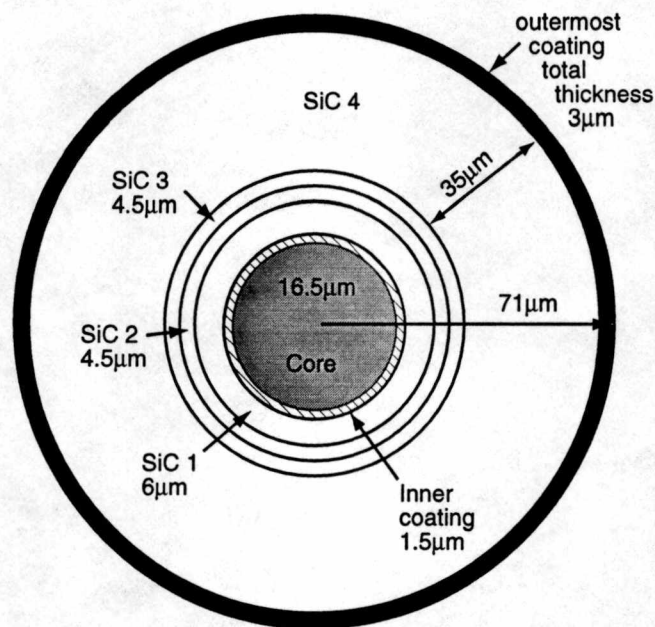


Figure 187. Si-C fiber microstructure.

"Processing of Fiber Reinforced Titanium Composites," Second ASM Paris Conference, September 1991.

The outer layer of the resulting strand is made carbon rich so that the Ti matrix cannot penetrate the fiber and reduce its strength. The  $\beta$  Si-C is present as such across all of the fiber cross-section except the last few micrometers at the surface where, by altering the gas flows in the bottom of the tubular reactor, the surface composition and structure are modified by an addition of amorphous carbon that heals the crystalline surface for improved surface strength, followed by a modification of the Si to C ratio to provide improved bonding with the metal. The surface of the Si-C fiber must be tailored to the matrix. One type fiber has been combined with Al to form the MMC which is used for tactical missile fuselages and for carrying primary structural loads in the space shuttle orbiter. The fiber used is 35% 6Al-Ti-4V and 48% 6061 Al. Peak deposition temperature is 2,370 °F. Above this grain growth results, and below this internal property degradation stresses occur. These Si-C fibers handle the primary loads and have a tensile strength of

over 550 ksi, a modulus of over 58 m/in<sup>2</sup>, and a density of 0.11 lb/in<sup>3</sup>. They are currently produced in two diameters, 5.6 and 3 mil. Figure 188 shows the tensile strength vs. time at temperature (1,650°F) of the SCS-9/ $\beta$  21S composite. [99]

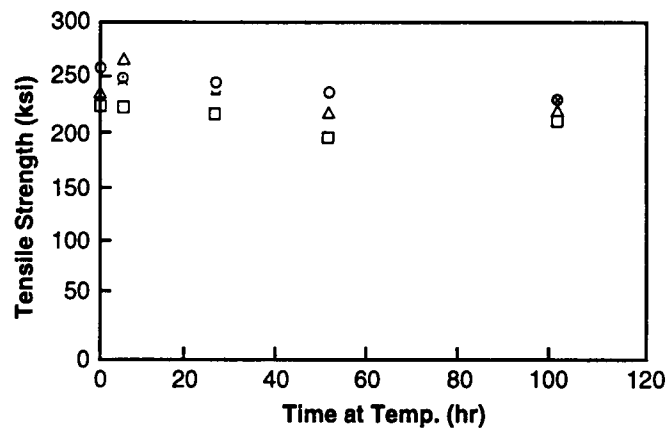


Figure 188. Tensile exposure result of SCS-9/ $\beta$  21S composite.

A number of techniques are used for composite fabrication using either foil or powder matrix material in combination with a continuous fiber. These include hot isostatic pressing, vacuum hot pressing, arc spraying, plasma spraying, powder cloth technique, electron beam vapor deposition, and a woven fiber mat. Thermal expansion and chemical compatibility between the fiber and the matrix are the major challenges to the fabrication of these high performance composites. Studies of TMC's have shown that reaction between the Ti<sub>3</sub>-Al matrix and Si-C-based fiber is less than that of Ti-6Al-4V/Si-C composites, and that with increasing Nb addition to Ti<sub>3</sub>-Al the reaction zone width decreases. Textron has developed the SCS-6/(Ti-24Al-11Nb) with excellent properties. The new  $\beta$  will have improved fiber and matrix CTE compatibility over the  $\alpha$ -2.

In the processing of a TMC, the principle objective is to firmly embed the fiber within the matrix while achieving uniform spacing of the fibers without voids or delamination and providing a tailored interfacial bond between fiber and matrix. In general, and as is done for Si-C/Al, this is done at or near molten metal temperature.

However, Ti is extremely reactive at elevated temperature. To alleviate reactions at the fiber/matrix interface which reduce strength, the processing temperature is reduced. This requires a solid-state high-pressure consolidation process that demands careful control of metal placement. [99]

Textron uses two techniques for turning the Si-C fibers into preform sheets with Ti. In the first, Si-C fiber is wound around a large circular drum with the desired spacing. After the drum is placed in an argon-filled chamber, a hot Ti plasma is sprayed over and around the strands, bonding to them and forming the metal matrix of a monotape up to 10-ft long and 2-ft wide. The other method is weaving. Single-arm rapier looms and shuttle looms are used to produce mats of 100 to 140 reinforcing fibers per inch which are held together by a cross-weave of Ti ribbons. Next, the fiber weaves are sandwiched between layers of Ti foil about 0.005-in thick. Varying the orientation of Si-C fibers from layer to layer increases the final strength of the TMC. Fiber content of TMC preform sheets is typically 35% by volume (Fig. 189). [99]

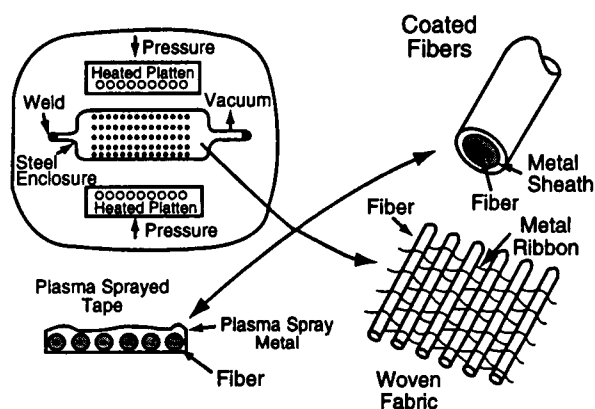


Figure 189. Fiber preforms.

The foil and fiber-weave sandwiches, or the plasma-sprayed preforms, are then laid up in clean rooms into a mold, which is then placed in a steel "bag." The bag is a

rectangular, flat steel frame that contains the mold. The upper and lower skins of the 0.024-in thick sheet steel are placed over the frame and welded to it, forming the complete sealed bag. A vacuum is then imposed by evacuating the air from the steel bag. The skins are thick enough to prevent tearing as they compress at high temperature, but do not restrict movement. Next is the hot isostatic press (HIP) (Fig. 190), where the material is heated. [99]

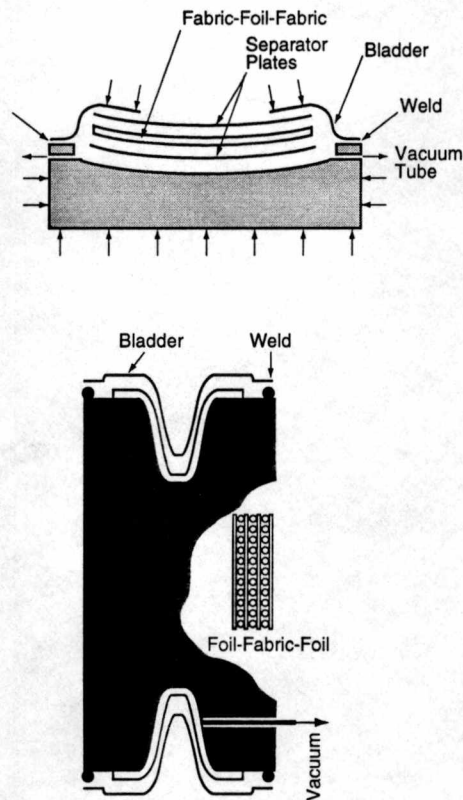


Figure 190. HIP processing of skin panels and stiffeners.

The gas pressure deforms the bladder which consolidates the composite against the mold. HIP conditions are 15 ksi pressure at temperatures up to 2,350 °F for a consolidation time of 2 to 4 hr. After the HIP process, the fixture is flame-cut open, and final trimming is done with a water-jet cutter that uses garnet in suspension. Tubular TMC parts are HIP processed between concentric cylinders. Large parts (such as LV



FCA's) and complicated parts use a HIP, similar to an autoclave but which operates at higher temperatures and pressures. Diffusion-bonding TMC parts to one another can be done during HIP processing or in a separate operation such as an autoclave. [99]

TMC materials are subjected to extensive testing to provide mechanical information about the interfacial bond strength between matrix and reinforcement. Figure 191 shows the top and underside of a Ti-6Al-4V/35% volume Si-C monofilament composite wedge after testing. [104] In Textron's inspection facilities, many methods are used on TMC parts which include x ray, ultrasound, and scanning electron microscopes. Textron maintains a comprehensive network of laboratories that provide full material property assessment and ballistic evaluation. [99] Interfacial chemical reactions are a concern for Ti composites, because Ti and its alloys tend to react with most candidate reinforcements. The diffusion bonding process commonly used in fabricating TMC's can cause significant interfacial reaction when the reinforcement is Si-C monofilament. Table 30 shows interfacial shear strength. [104]

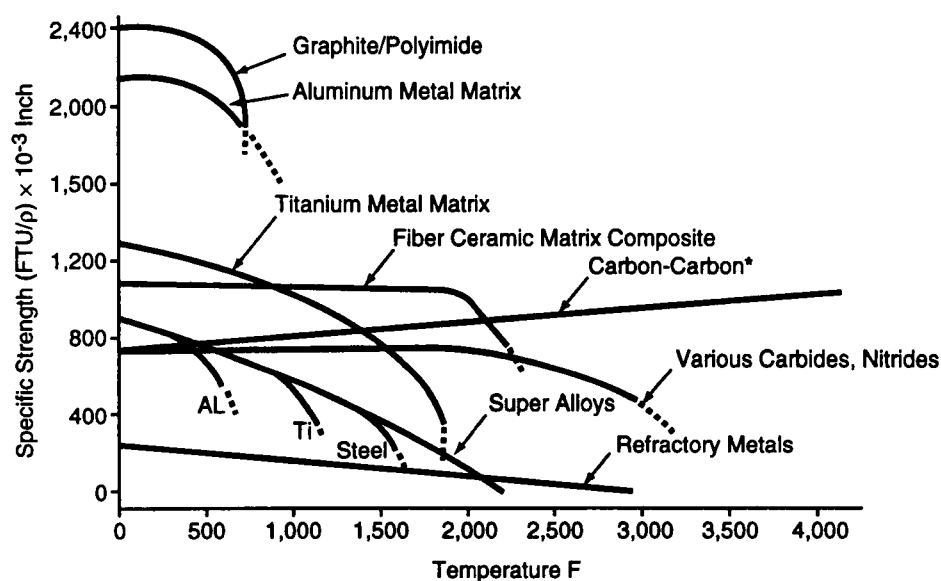


Figure 191. TMC bond strength after testing as depicted by electron micrograph.

Table 30. Interfacial shear strength of TMC.

| Matrix             | Si-C fiber | Processing                        | Test method   | $\tau^*$<br>(MPa) | $\tau_f$<br>(MPa) |
|--------------------|------------|-----------------------------------|---------------|-------------------|-------------------|
| Ti                 | Sigma-1    | spray deposit                     | pull-out      | 50                | 12                |
| Ti-6Al-4V          | Sigma-1    | diffusion bond                    | fragmentation | 345               | —                 |
| Ti-6Al-4V          | Sigma-2    | diffusion bond                    | push-out      | —                 | 80                |
| Ti-6Al-4V          | Sigma-2    | diffusion bond, 59 h at 815 °C    | push-out      | —                 | 211               |
| Ti-25Al-10Nb-3V    | SCS-6      | powder hot press                  | push-out      | 110               | 57                |
| Ti-25Al-10Nb-3V    | SCS-6      | powder hot press, 100 h at 800 °C | push-out      | 99                | 59                |
| Ti-6Al-4V          | SCS-6      | diffusion bond                    | push-out      | 150               | 90                |
| Ti-15V-3Al-3Cr-3Sn | SCS-6      | diffusion bond                    | push-out      | 120               | 80                |
| Ti-25Al-10Nb-3V    | SCS-6      | diffusion bond                    | push-out      | 120               | 50                |
| Ti                 | SCS-6      | spray deposit                     | pull-out      | 5                 | 1                 |
| Ti-6Al-4V          | SCS-6      | diffusion bond                    | fragmentation | 180               | —                 |
| Ti-6Al-4V          | B or Si-C  | diffusion bond                    | fragmentation | 240               | —                 |

Figure 192 [99] compares the high-temperature strengths of many of the advanced materials discussed in this assessment, and Table 31 compares tensile properties of TMC's and other MMC's. [103] Additionally, use of TMC material requires less thermal protection which is a considerable cost savings and a further reduction in weight.



\*Coating State-of-the-Art Temperature Limit

Figure 192. Strength comparisons of advanced materials.

Table 31. Tensile properties of TMC's and other MMC's.

| Material                                                     | Tensile strength,<br>MPa (ksi)     | Tensile modulus,<br>GPa (Msi)  |
|--------------------------------------------------------------|------------------------------------|--------------------------------|
| 6061-T6 aluminum alloy                                       | 306<br>(44.4)                      | 70<br>(10)                     |
| T-300 carbon-6061 Al alloy<br>( $v_f = 35$ to 40%)           | 1,034 to 1,276 (L)<br>(150 to 185) | 110 to 138 (L)<br>(15.9 to 20) |
| Boron-6061 Al alloy<br>( $v_f = 60\%$ )                      | 1,490 (L)<br>(216)                 | 214 (L)<br>(31)                |
|                                                              | 138 (T)<br>(20)                    | 138 (T)<br>(20)                |
| Particulate Si-C-6061-T6 Al alloy<br>( $v_f = 20\%$ )        | 552<br>(80)                        | 119.3<br>(17)                  |
| GY-70 carbon-201 Al alloy<br>( $v_f = 37.5\%$ )              | 793 (L)<br>(115)                   | 207 (L)<br>(30)                |
| Al <sub>2</sub> O <sub>3</sub> -Al alloy<br>( $v_f = 60\%$ ) | 690 (L)<br>(100)                   | 262 (L)<br>(38)                |
|                                                              | 172 to 207 (T)<br>(25 to 30)       | 152 (T)<br>(22)                |
| Ti-6Al-4V titanium alloy                                     | 890<br>(129)                       | 120<br>(17.4)                  |
| Si-C-Ti alloy<br>( $v_f = 35$ to 40%)                        | 820 (L)<br>(119)                   | 225 (L)<br>(32.6)              |
|                                                              | 380 (T)<br>(55)                    |                                |
| SCS-6-Ti alloy<br>( $v_f = 35$ to 40%)                       | 1,455 (L)<br>(211)                 | 240 (L)<br>(34.8)              |
|                                                              | 340 (T)<br>(49)                    |                                |

Since NASA's Saturn V LV took us to the Moon using 300 ft<sup>2</sup> of aerodynamic surfaces, a plethora of advanced composites and smart materials has been developed that enable the design of very lightweight, strong, and innovative LV aerodynamic FCA's. The new MMC's can not only provide substantial weight savings and increased high-temperature strength, which allow increased payload capability and flight performance, but can also reduce the thermal protection requirements, thus providing even more indirect weight and cost savings. TMC's have already been selected by NASA for the NASP, and appear to be among the candidates in the advanced composites for use on the LV FCA's of this research. The author has conducted a FCA design review with a

leading TMC producer and ascertained the feasibility of using TMC for the fabrication of full scale LV FCA's.

Many of the smart materials can be used as both sensors and actuators, and by physically deforming themselves can produce significant actuation forces. Patents have recently been granted on articulated flight control surfaces using shape memory alloys (SMA's). The replacement of conventional flight control actuation systems with SMA actuators could result in enormous savings in cost and weight. In the case of a LV, the added weight of large heavy hydraulic actuators could perhaps be replaced with the use of on board substances. The response time of the SMA actuator is directly controlled by the heating and cooling rate. Because a LV has onboard an excess of both extremely high-temperature exhaust gases from the rocket engines, and extremely low-temperature cryogenic propellants, the SMA actuator may be worth investigation for LV application.

In order to effect a maximum FCA deflection with a minimum of effort, the SMA actuator could be used to activate a small tab on the TE of the all-movable FCA. During the time of maximum dynamic pressure, extremely small amounts of SMA actuator deflection of the control tabs could be effectively magnified by the dynamic pressure of the air flow to move the entire FCA. The control tab could thus supply the hinge moments required to move the FCA. The SMA actuator could drive the control tab, which in turn could drive the FCA, which would produce the required control authority augmentation for the vehicle. Figure 193 shows H as the hinge point of the control tab and AB as the SMA actuator. When the SMA actuator (e.g., rod/spring) elongates and contracts it deflects the small control tab. This causes an automatic deflection in the opposite direction of the FCA by the force of the dynamic pressure of the airstream.

Smart materials with embedded sensors and actuators could be used to damp undesirable vibration modes as a LV goes through the subsonic, transonic, and supersonic

speed regimes in the ascent trajectory. As more LV's are designed with aft cg's, and as hypersonic flight research proceeds, smart materials and advanced composites can lend themselves to very attractive aerodynamic flight control solutions to flight stability and control challenges. This brief overview has hopefully winnowed out from the burgeoning array of literature and provided insight to those which are directly applicable to flight control augmentors of a LV. Nowhere in the literature has there been found any application of smart materials or advanced composites to the flight controls of an expendable, large Saturn class LV.

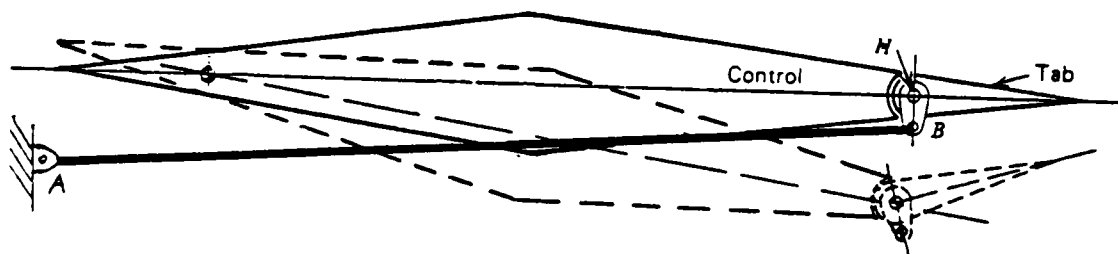


Figure 193. Author proposed launch vehicle shape memory alloy actuation of control tabs and control augmentation devices.

## CHAPTER XI. ELASTIC VEHICLE

### 11.1 Elastic Considerations

The experimental LV of this research has a very far aft cg. It was desired to know how the elastic properties of this vehicle had changed from the original forward cg configuration, and if there would be any coupling of the elastic body frequencies with the frequencies of the control system. In this chapter, the significant modes of vibration are identified and the response spectra compared for the far aft cg experimental LV and the original configuration.

The subject of flight dynamics includes three major areas: Stability and Control, Performance, [105–107] and Aeroelasticity. While performance deals with long-term vehicle behavior, and stability and control deals with short-term vehicle behavior, aeroelasticity deals with high frequencies and very short-term behavior of structural motions. When a launch vehicle (LV) flies through the atmosphere during its ascent trajectory, aerodynamic forces act over its fuselage. All LV's, in varying degrees, are subject to the same static and dynamic instabilities that can be encountered by aircraft, plus some additional ones. In reality, a LV is not a rigid body, but an elastic body.

After liftoff and during ascent, the elastic LV is subjected to a myriad of randomly varying forces including a time-dependent crosswind velocity. Some of these fluctuating loads are rather localized in extent and produce vibrations of only a small portion of the total structure. Other random loadings may be spread over the entire fuselage and produce vibrations of the complete LV (Figures 194 and 195 [108]). There are also cases where rather localized forces can produce vibratory response of the complete vehicle. Vibrations resulting from random forces can produce dynamic aeroelastic instabilities which can and have resulted in catastrophic failure. In addition, internal acoustic fields generated by the random vibrations of structural panels can have adverse effects on the crew or result in

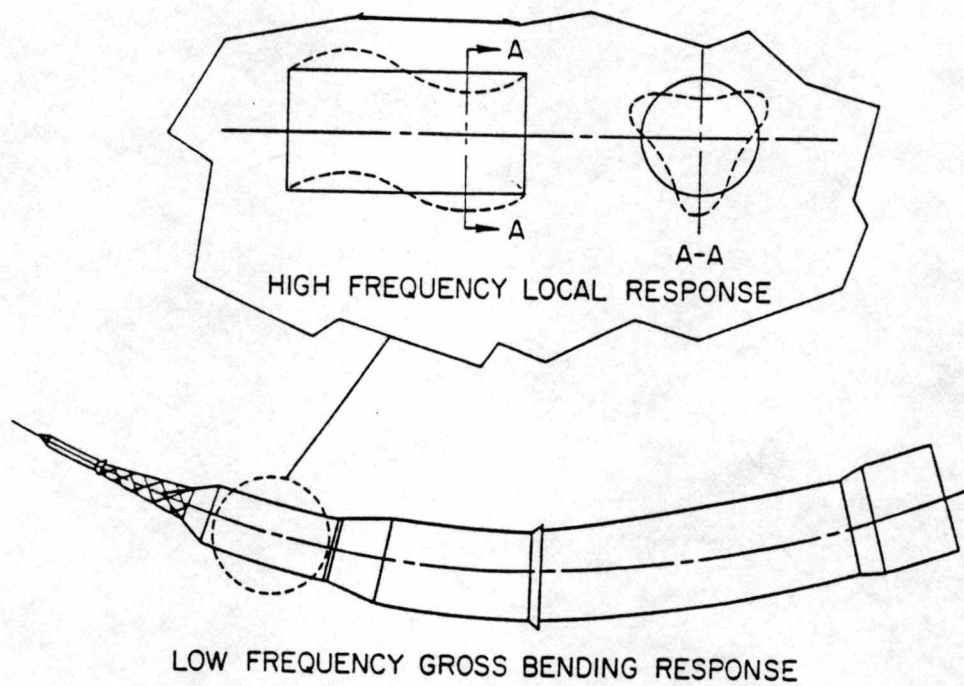


Figure 194. Types of LV elastic response.

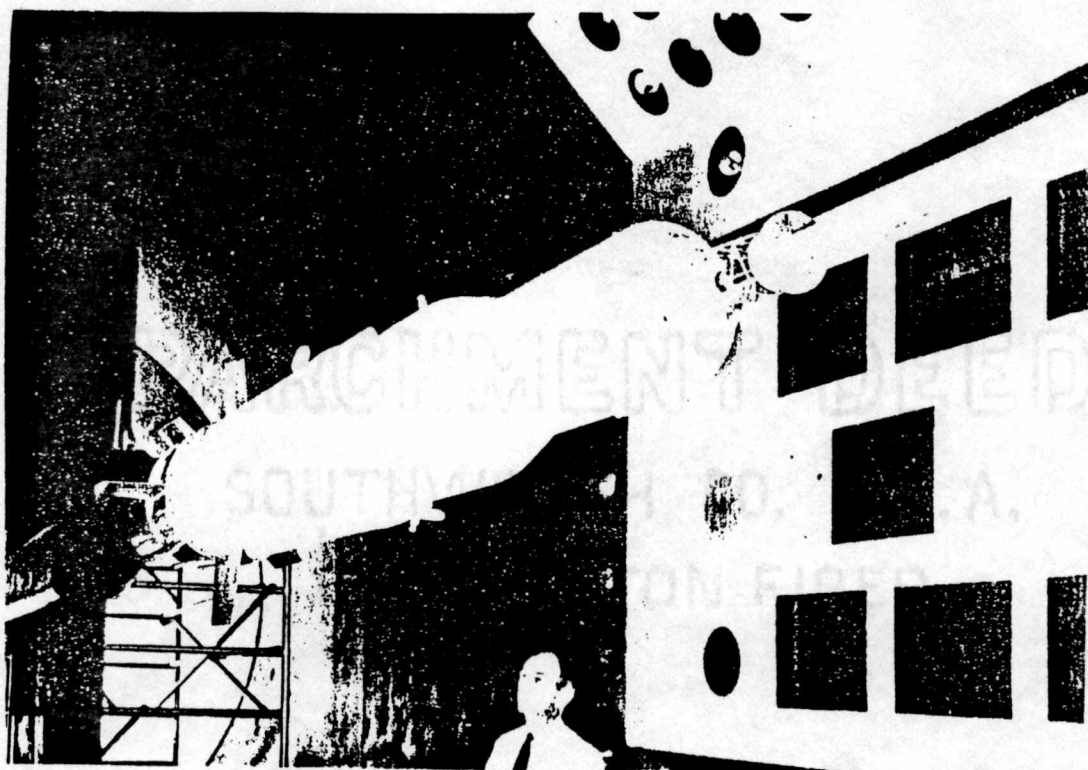


Figure 195. Saturn I 8% elastic model in wind tunnel test.

damage to sensitive payloads. Some sources of significant random forces on a LV include atmospheric turbulence (wind gusts and wind shear), ground winds (including vortex shedding from the LV), engine noise, and engine gimballing. Ground-wind loads have been a problem in smokestacks and suspension bridges as well as for LV's on the launch pad. The steady-state wind imposes a steady load due to aerodynamic drag and an oscillatory perpendicular lift load caused by vortex shedding around the LV (Figs. 196 [109] and 197 [108]). Bending moments are caused by the steady-state winds, and oscillatory loads are due to the vortex shedding. Additionally, unsteady loads are imposed by turbulence and gusts in the atmospheric winds. Also, the turbulence wake from adjacent launch pad service structures can affect the LV response. The response of LV's to ground wind loads is usually confined to the first bending mode, and the amplitude varies with time. Figure 198 [108] shows typical aeroelastic ground wind load models that have been wind tunnel tested. Table 32 shows typical source frequency ranges. [108]

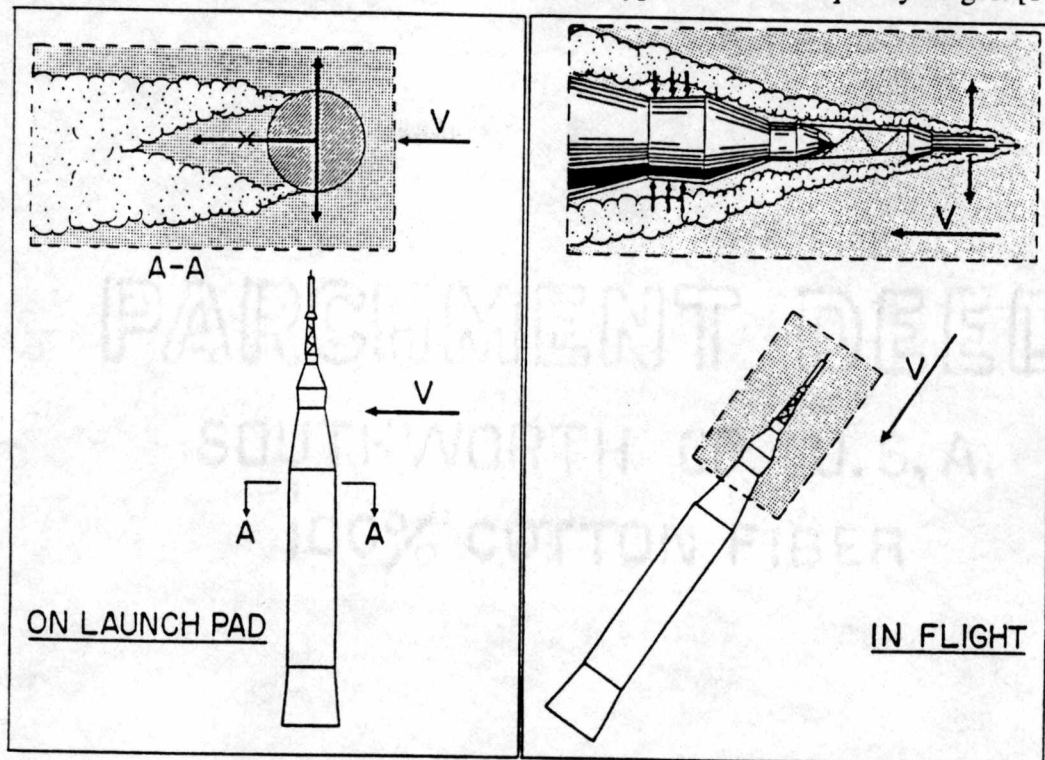


Figure 196. Loads caused by separated flow over a LV.



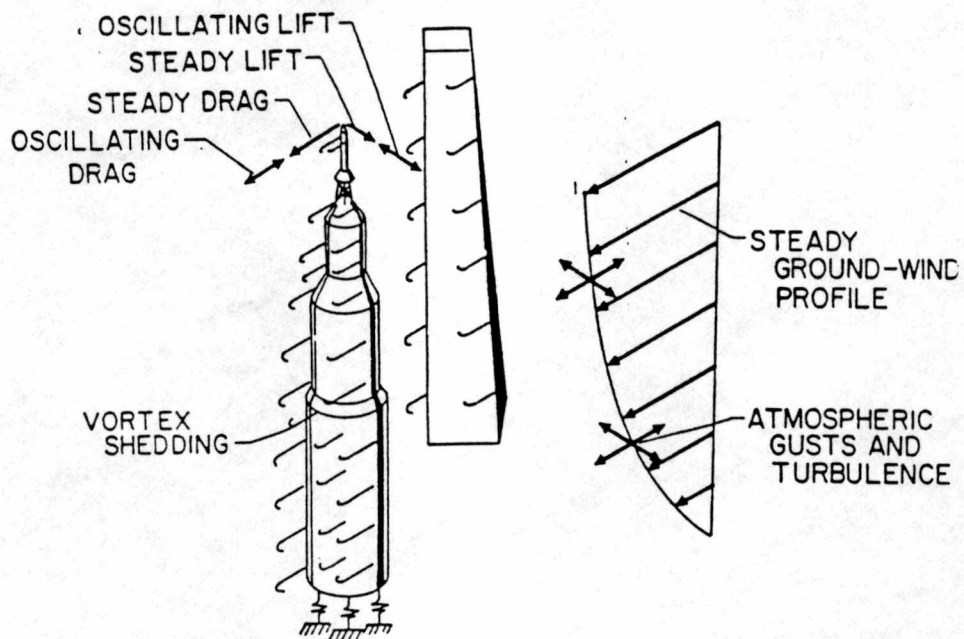


Figure 197. Ground-wind loads factors.



Figure 198. Aeroelastic Saturn and Scout wind tunnel models.

Table 32. Disturbance frequency ranges.

|                           |                 |
|---------------------------|-----------------|
| Ground Winds              | 0 – 10 Hz       |
| Engine Noise              | 10 – 10,000 Hz  |
| Atmospheric Turbulence    | 0 – 20 Hz       |
| Boundary Layer Turbulence | 100 – 10,000 Hz |
| Buffeting                 | 0.5 – 2,000 Hz  |

The importance of a LV elasticity assessment has been detailed in Reference 41. Aeroelasticity, the interaction of aerodynamic forces with an elastic body, is an important LV consideration, since a deformed fuselage can result in a change in magnitude and distribution of surface forces. The effect of aeroelasticity is to change the local angle-of-attack ( $\alpha$ ) and control deflection through redistribution of airloads. The  $\alpha$  can be changed due to body bending and body torsion. The  $\alpha$  can increase due to the elastic angular deflection of the fuselage in pitch.

$$\alpha_{\text{actual}} = \alpha_{\text{rigid-body}} + \alpha_{\text{elastic}} \quad (80)$$

Aeroelasticity can thus significantly affect the stability and control, hence, the performance of a LV. For example, the pitching moment about the cg can increase more than 16% at maximum dynamic pressure, max Q, and the flexible body aerodynamic forces can increase the required engine gimbal angle by 30%, as was the case on the Saturn V LV discussed in Chapters II and III. Flexible body aerodynamics can cause a forward shift of the cp, which was approximately 10 ft (0.3 calibers) on the Saturn V.

As the slenderness ratio of the vehicle increases, so do the aeroelastic effects especially at max Q in the ascent trajectory. The large slenderness ratio results in low bending frequency. This can adversely affect the vehicle flight stability. Example slenderness ratios for LV's discussed in this work were given in Chapter V, Table 2.

Since the slenderness ratio of the experimental LV being considered here is three times that of the space shuttle, the aeroelastic effects are more significant. As the slenderness ratio increases, the lowest bending frequency decreases; in the experimental LV considered here it is 2.8 Hz as can be seen in section 11.4 below. Considerable aeroelastic wind tunnel testing was conducted on the Saturn series of LV's. Some typical bending mode frequencies were 1.91, 3.56, 8.88, 14.44, and 26 Hz. [110] Elastic body frequencies are higher than those of the rigid body, and after liftoff the rigid-body mode frequencies are zero. When a LV is launched vertically from a launch pad, there is a large change in the structural boundary conditions which accompanies the liftoff. If the LV is idealized as a uniform beam of length  $\ell$ , the fundamental natural frequency can be expressed as

$$\omega_1 = \frac{N \pi^2}{\ell^2} \sqrt{\frac{EI}{m}} \quad (81)$$

where  $m$  is the mass/length and  $EI$  is the flexural rigidity. On the launch pad, the constant  $N$  may have values from 0.356 (cantilever support) to 1.562 (free simple support). After liftoff it rises to 2.28. Thus there could conceivably occur a sudden, six-fold jump in the most significant structural frequencies. [111]

Panel flutter is a dynamic oscillatory aeroelastic instability that occurs at certain speeds at which the energy extracted from the airstream during a period of oscillation is exactly dissipated by the hysteretic damping of the surface and has been observed in the V-2, the X-15, and many LV's since. At speeds above flutter speed, divergence of the oscillation occurs. Methods of aeroelastic analysis differ according to the time dependence of the inertial and aerodynamic loads. Panel flutter can occur in a LV during supersonic flight and can be analyzed using a cylindrical shell model, such as Donnell's

formulation [111]. The key to flutter prevention is to break up any coupling between the degrees-of-freedom (DOF).

Aeroservoelasticity is the branch of aeroelasticity which deals with the flight control surfaces. Aeroelastic wind tunnel tests conducted at LaRC with and without the rigid Saturn I aerodynamic surfaces showed the same basic first bending mode shape. There was a small decrease in the first bending mode frequency and an increase in total damping with the aerodynamic surfaces especially at subsonic speeds. The thick wedge surfaces cause these changes to a greater degree than did the thin surfaces. [110] Table 2 in Chapter V has shown that the ratio of fin area to the total surface wetted area for the Saturn I LV was 8%. The fuselage bending and torsion have generally been shown to be more significant in altering the control forces on the vehicle than the local flight control surface deformations. In addition, it was shown in Chapter V eq. (13) that the total FCA surface area is  $\approx 1.6\%$  of the LV wetted surface area, hence the FCA's have been neglected in this elastic body analysis. The analysis in this chapter looks at how the fuselage basic vibration modes change with the cg located far aft.

Since the bending modes can couple with the control system when the bending frequencies are low, it is desirable to have the first bending mode frequency much higher than the control frequency. When bending mode frequencies and control frequencies are of the same order of magnitude, potential problems exist. In an unstable Atlas missile flight, a 17-Hz limit cycle resulted from the coupling of the autopilot and the third lateral bending mode. Preflight simulations including the first three lateral bending modes did not uncover this problem because of the use of a linear third-order simulation for the hydraulic actuator, which did not incorporate the effect of the missile vibration on the hydraulic servo system. Changes to the autopilot to attenuate the third lateral bending mode then eliminated this coupling. [111] It is desired to have the control frequency approximately one-tenth of the first bending mode frequency. Flight control effectiveness

is affected by structural deformation. The forces and moments applied by the engines are governed both in magnitude and direction by the information derived from the sensors located along the fuselage. Flexible body analysis determines filter requirements and sensor locations.

Active elements should be located on the LV where their authority over the modes to control is largest (Fig. 199).[41] Vehicle angular rates and displacements, the rigid-body six DOF together with those of the elastic deflections, and the slosh are sensed by combinations of attitude gyros and accelerometers strategically located along the

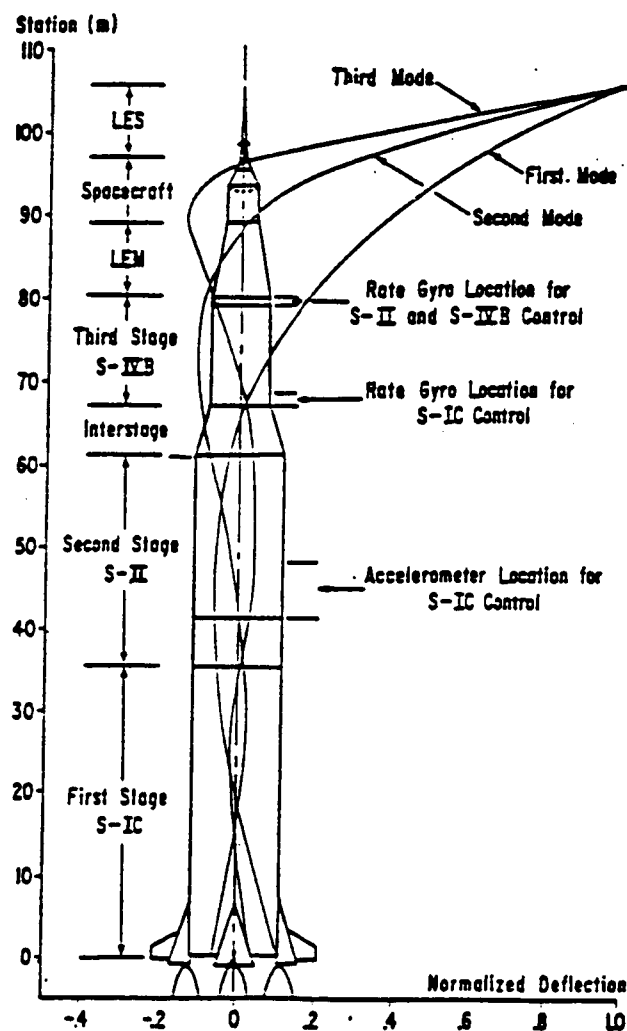


Figure 199. Saturn V mode shapes and sensor locations.

fuselage, ideally at the antinodes for gyros and at nodes for accelerometers, on the lowest vibration mode. If sensors are not properly placed, the aeroelastic effects can cause erroneous signals from the position and rate gyros on a deflected vehicle. Notch filters can be used to suppress sensor signals in selected small frequency ranges. Since multi-modes are involved, elastic DOF are best included in the control system design. Under adverse conditions, the autopilot can reinforce the amplitude of oscillation, leading ultimately to structural failure of the vehicle. The most natural way to actively damp an elastic mode is by local rate feedback. Uncertainty in modal parameters can require extended Kalman filters or adaptive control.

A prerequisite for control system design is, therefore, an adequate representation of the elastic body motion under prescribed environmental and internal disturbance forces. This representation can be conveniently expressed in terms of the vibration frequencies and mode shapes of the vehicle. The vibrations consist of longitudinal, lateral, and torsional. Of primary importance for stability analysis are the lateral vibration modes.

Ericsson et al. [112] have developed and successfully used a lumped load analysis of the Saturn LV's to assess aeroelastic characteristics of the vehicle using static load distributions as input to the elastic equations of motion. The general force distributions along the vehicle were reduced to lumped loads. The lumped load representations used on Saturn LV's can be seen in Figure 200. [112]

Since  $\Delta \alpha$  due to the aerodynamics is the major concern, a 1-D approximation could illuminate the aeroelastic effects on the stability derivatives. Basic equations are

$$\text{Bending: } \frac{\partial^2}{\partial x^2} \left[ EI \frac{\partial^2 w}{\partial x^2} \right] m(x) \ddot{w}(x, t) = F_z(x, t) \quad (82)$$

where  $F(x, t)$  is the running external load.

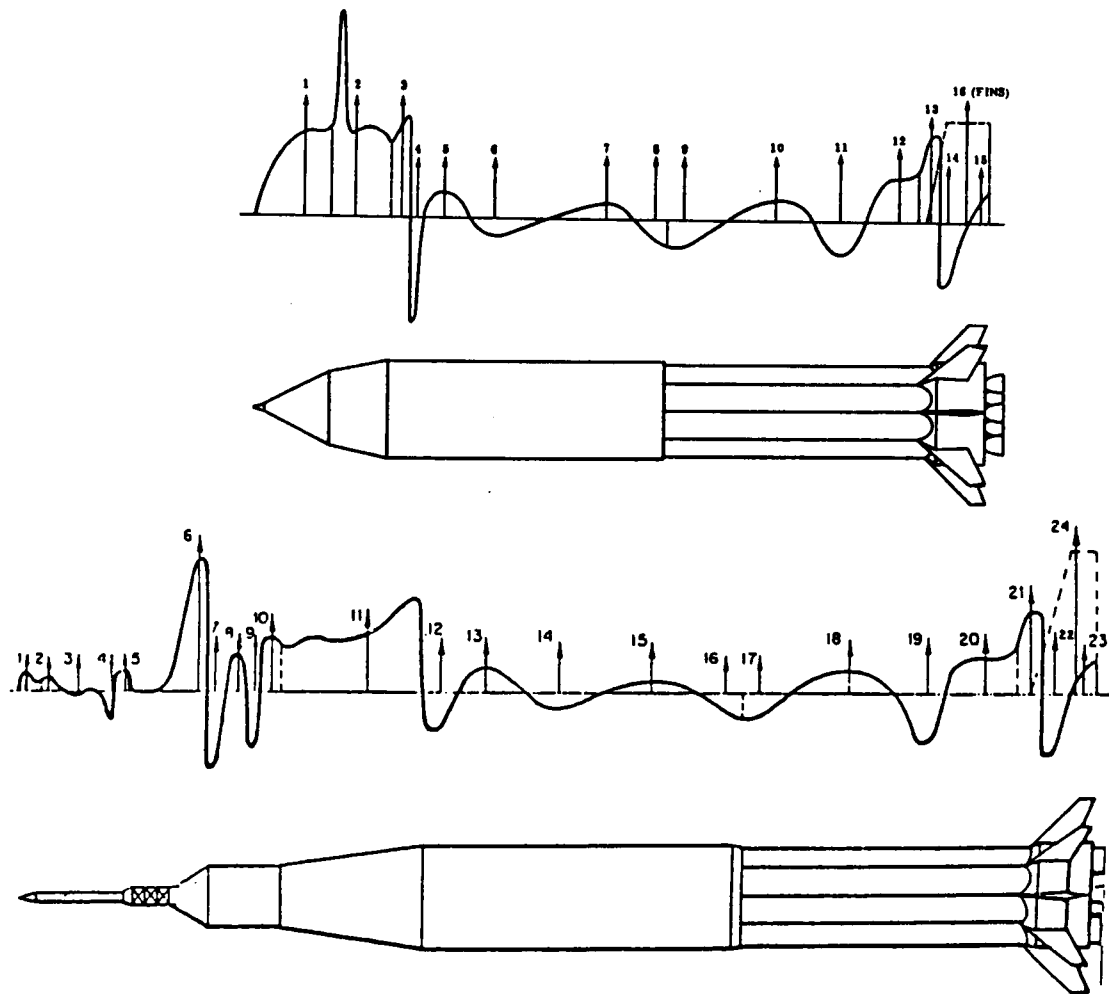


Figure 200. Saturn IB lumped normal force vectors.

$$\text{Torsion: } \frac{\partial}{\partial x} \left[ G J_x \frac{\partial \theta}{\partial x} \right] - I_x(x) \ddot{\theta}(x, t) = -T_x(x, t) \quad (83)$$

The torsion problem is of less importance than the bending as found in section 11.4 below. In fact there are longitudinal accelerations which may cause axial deformations (slosh of propellants) and vibrations of more significance than torsion.

$$\text{Longitudinal: } \frac{\partial}{\partial x} \left[ E A \frac{\partial u}{\partial x} \right] - m(x) \ddot{u}(x, t) = -F_x(x, t) \quad (84)$$

where  $u(x, t)$  = axial deformation, and  $A(x)$  = cross-sectional area.

For computational expediency and physical appeal in higher DOF, orthogonal elastic modes are used. However, lacking a set of truly orthogonal modes, completely arbitrary modes, assumed modes, may be used. (See Ref. 113) The absolute displacement of  $w(x, t)$  of the LV centerline can be conveniently represented by the summation of two rigid-body modes plus the normal modes  $\phi_i(x)$  of vibration of the elastic LV,

$$w(x, t) = w_{\text{Rigid modes}}(t) \theta_R(t) x + \sum_{i=1}^m \phi_i(x) \xi_i(t) \quad (85)$$

It can be shown that the equations of motion are

$$\begin{aligned} \text{Force: } M \ddot{w}_R &= T \delta_R + \int_0^x L(x, t) dx \\ \text{Moment: } I \ddot{\theta}_R &= -x_R T \delta_R + \int_0^x x L(x, t) dx \end{aligned} \quad (86)$$

$$\text{Elasticity: } M_i \ddot{\xi}_i + \omega_i^2 M_i \xi_i = \Xi_i(t) \equiv \phi_i(x) T \delta_R + \int_0^x L(x, t) \phi_i(x) dx$$

where  $T$  = engine thrust,  $\delta$  = gimbal deflection,  $L$  = aerodynamic force/unit length;  $M$  = generalized mass;  $\omega_i$  = frequency; and  $i$  = elastic DOF. [111]

In modal synthesis commonly used on space vehicles, the modes of individual parts are calculated, then the overall vehicle modes are obtained by combining the effects of aerodynamic loads, engine deflections, and propellant sloshing. In practice, different groups perform the analysis of different components of the vehicle. A mode synthesis approach was used by Kiefling [114] for the calculation of the Saturn I vibrational modes. The behavior of the individual element and, therefore, the entire LV is dependent on the shape functions used. The modes are ideally assumed such that the first mode contributes  $\approx 80\%$ , the second mode  $\approx 15\%$ , and the third mode  $\approx 3\%$ . It is usually not known a



priori how many modes to assume. The mode shapes of the components are used in a Rayleigh-Ritz analysis to calculate approximate mode shapes and frequencies of the complete vehicle. One such technique is the Craig-Bampton method where each component is modeled using a set of assumed modes. [113] In general, different elastic modes may have different damping factors. Nonlinear analysis with time-varying stability derivatives can be used for a more in-depth determination of flight stability and vehicle response to in-flight winds.

In dealing with the dynamic and aeroelastic behavior of complete LV's in flight, all external constraints must be released and the vehicle must be treated as free in space. A completely general formulation of the aeroelastic equations of motion of a free vehicle in flight involves six DOF. For the six DOF unrestrained vehicle, the total surface force on the vehicle is

$$\bar{F} = \bar{F}_{\text{disturbance}} + \bar{F}_{\text{aerodynamic}}$$

The integral equations of linear and angular momentum are

$$\frac{d}{dt} \iiint_V \frac{d\bar{r}'}{dt} \rho dV = M \frac{d^2 \bar{r}'_o}{dt^2} = \iint_S (\bar{F}_{\text{dist}} + \bar{F}_{\text{aero}}) dS \quad (87)$$

$$\frac{d}{dt} \iiint_V \bar{r} \times \frac{d\bar{r}}{dt} \rho dV = \iint_S \bar{r} \times (\bar{F}_{\text{dist}} + \bar{F}_{\text{aero}}) dS$$

where  $\rho$  = mass / unit volume;  $V$  and  $S$  = volume and surface integrations, respectively;  $M$  = total mass;  $\bar{r}$  = position vector. The elastic equation can be expressed as

$$\tilde{\zeta}(\mathbf{q}) + \rho \bar{a} = 0 \quad (88)$$

where  $\tilde{\zeta}(\ )$  is a tensor differential operator with the form

$$\tilde{\zeta}(\ ) = - [ (\lambda + G) \text{grad div} + G \nabla^2 ] (\ ) \quad \text{and}$$

$$\bar{a} = \frac{d}{dt} \left( \frac{d\bar{r}'}{dt} \right) = \frac{d}{dt} \left( \frac{d\bar{r}'_0}{dt} + \frac{d\bar{r}}{dt} + \frac{d\bar{q}}{dt} \right) \quad (89)$$

where  $\lambda$  = Lamé elastic constant;  $\bar{a}$  = acceleration;  $G$  = shear modulus;  $\bar{r}$  = position vector from the cg; and  $\bar{q}$  = an elastic deformation vector such that  $\bar{r} = \bar{r}' + \bar{q}$ . [111]

Initial conditions must be adjoined to eq. (88). The Boundary Conditions (BC) for eq. (88) are,

$$\begin{aligned} F_x &= \lambda \Delta \bar{n} \cdot \bar{i} + G \left[ \frac{\partial u}{\partial n} + \frac{\partial u}{\partial x} \bar{n} \cdot \bar{i} + \frac{\partial v}{\partial x} \bar{n} \cdot \bar{j} + \frac{\partial w}{\partial x} \bar{n} \cdot \bar{k} \right] \\ F_y &= \lambda \Delta \bar{n} \cdot \bar{j} + G \left[ \frac{\partial v}{\partial n} + \frac{\partial u}{\partial y} \bar{n} \cdot \bar{i} + \frac{\partial v}{\partial y} \bar{n} \cdot \bar{j} + \frac{\partial w}{\partial y} \bar{n} \cdot \bar{k} \right] \\ F_z &= \lambda \Delta \bar{n} \cdot \bar{k} + G \left[ \frac{\partial w}{\partial n} + \frac{\partial u}{\partial z} \bar{n} \cdot \bar{i} + \frac{\partial v}{\partial z} \bar{n} \cdot \bar{j} + \frac{\partial w}{\partial z} \bar{n} \cdot \bar{k} \right] \quad (90) \end{aligned}$$

where  $u, v, w$  are prescribed surface displacements and  $\bar{n}$  is a unit normal to the surface in the outward direction.

In order to carry out the analyses and numerical computation, it is necessary to reduce the equations to their component form. [111]

$$\text{Linear velocity:} \quad \frac{d\bar{r}'_0}{dt} = U_x \bar{i} + U_y \bar{j} + U_z \bar{k} \quad (91)$$

where  $\bar{U} = (U_x, U_y, U_z)$

$$\text{Angular velocity:} \quad \bar{\omega} = p \bar{i} + q \bar{j} = r \bar{k} \quad (92)$$

Using these and

$$\dot{\bar{i}} = r \bar{j} - q \bar{k}; \quad \dot{\bar{j}} = -r \bar{i} + p \bar{k}; \quad \dot{\bar{k}} = q \bar{i} - p \bar{j} \quad (93)$$

the following are obtained:

$$\begin{aligned}
M(\dot{U}_x - U_y r + U_z q) &= P_x ; & M(\dot{U}_y + U_x r - U_z p) &= P_y ; \\
M(\dot{U}_z - U_x q + U_y p) &= P_z
\end{aligned} \tag{94}$$

$$\text{where} \quad (P_x, P_y, P_z) = \bar{P} = \iint_s (F_{\text{dist}} + F_{\text{aero}}) dS \tag{95}$$

$$\text{Using} \quad \frac{d\tilde{r}}{dt} = \bar{\omega} \times \tilde{r} ; \quad \text{and} \quad \frac{d\bar{q}}{dt} = \frac{\delta \bar{q}}{\delta t} + \bar{\omega} \times \bar{q} ;$$

$$\begin{aligned}
\frac{d}{dt} \iiint_V [ \tilde{r} \times (\bar{\omega} \times \tilde{r}) + \tilde{r} \times (\bar{\omega} \times \bar{q}) + \bar{q} \times (\bar{\omega} \times \tilde{r}) + \tilde{r} \times \frac{\delta \bar{q}}{\delta t} + \bar{q} \times \frac{\delta \bar{q}}{\delta t} \\
+ \bar{q} \times \bar{\omega} \times \bar{q} ] \rho dV = \iint_s \tilde{r} \times (\bar{F}_{\text{dist}} + \bar{F}_{\text{aero}}) dS
\end{aligned} \tag{95}$$

If small elastic deformations are assumed, this becomes

$$\frac{d}{dt} \iiint_V \left[ \tilde{r} \times (\bar{\omega} \times \tilde{r}) + \tilde{r} \times \frac{\delta \bar{q}}{\delta t} \right] \rho dV = \iint_s \tilde{r} \times (F_{\text{dist}} + F_{\text{aero}}) dS \tag{96}$$

In the basic formulation, three force equations, three moment equations, and three elastic equations are needed. If the elastic deformations are expressed in terms of the normal modes of the unrestrained vehicle,

$$\begin{aligned}
u &= \sum_{i=1}^{\infty} \phi_{x_i}(x, y, z) \xi_i(t) ; & v &= \sum_{i=1}^{\infty} \phi_{y_i}(x, y, z) \xi_i(t) ; \\
w &= \sum_{i=1}^{\infty} \phi_{z_i}(x, y, z) \xi_i(t) ; & \text{and} \quad \Phi_i(x, y, z) &= \phi_{x_i} \bar{i} + \phi_{y_i} \bar{j} + \phi_{z_i} \bar{k}
\end{aligned} \tag{97}$$

then

$$M_j \ddot{\xi}_j + M_j \omega_j^2 \xi_j = \Xi_{j \text{ dist}} + \Xi_{j \text{ aero}} \quad (j = 1, 2, \dots, n) \tag{98}$$

where  $M$  is the generalized mass

$$M_j = \iiint_v (\phi_{x_j}^2 + \phi_{y_j}^2 + \phi_{z_j}^2) \rho \, dV = \iiint_v \rho |\bar{\phi}_j|^2 \, dV \quad (99)$$

$$\Xi_{j_{\text{dist}}} = \iint_s (\bar{F}_{\text{dist}} \cdot \bar{\phi}_j) \, dS \quad ; \quad \Xi_{j_{\text{aero}}} = \iint_s (\bar{F}_{\text{aero}} \cdot \bar{\phi}_j) \, dS$$

## 11.2 Sloshing Propellants

During ascent, the mass of the propellant represents the major part of the LV mass, hence, sloshing of the propellants must be considered. Sloshing is the free-surface oscillation of propellants in a partially filled tank. These oscillations result from wind disturbances, engine gimbaling, attitude-stabilization control pulses, separation impulses, and elastic deformations of the vehicle. The magnitude and frequency of slosh are a function of tank slenderness; the inherent damping varies as the reciprocal of the tank diameter. Sloshing of propellant can adversely affect the flight stability of a LV, since the sloshing motion shifts the cg and can produce deforming forces on the vehicle. If the magnitude and frequencies of the sloshing motions are not properly controlled relative to the frequencies of the vehicle, the complete system may become unstable. An early example of undesirable coupling occurred on a Jupiter LV which broke up in flight when the sloshing propellant coupled with the control system, driving the vehicle unstable during max Q. After this occurrence, slosh baffles were implemented.

Today, slosh baffles are a standard feature of liquid propellant tanks and are added to provide the required damping. In a cylindrical tank, annular-ring baffles are commonly used. Figure 201 shows a MSFC baffle diagram and Figure 202 is a photograph of the ET LOX slosh baffle used in the experimental LV of this research. The magnitude of propellant sloshing and, hence, forces and moments on the LV depend on the tank

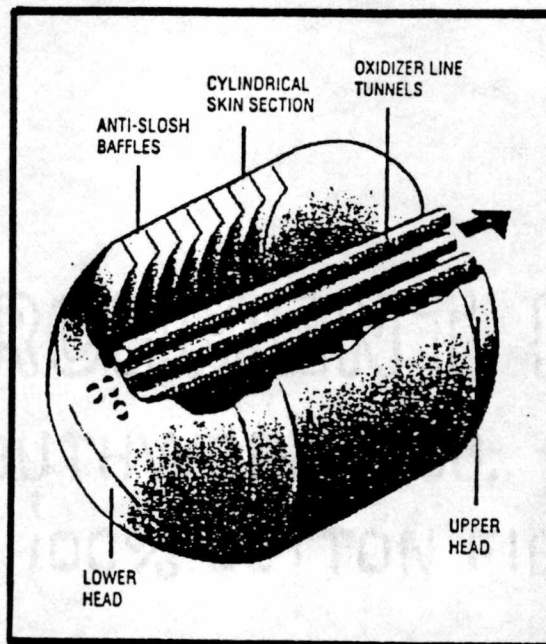


Figure 201. Ring baffles for sloshing control.

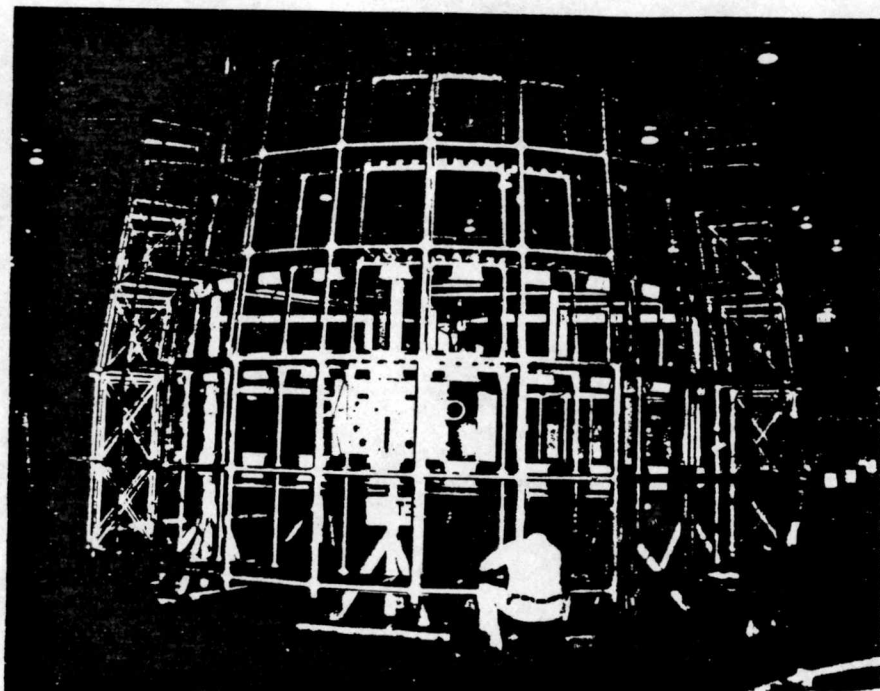


Figure 202. ET LOX slosh baffles used in the experimental L/V.

geometry, type of propellant, effective damping, and height of propellant in the tank. Baffle damping is only effective when the baffle is close to the propellant's free surface. A baffle is "added at a certain time of flight" by locating it in the propellant tank such that the free surface arrives at the baffle station just before the damping is needed.

The fundamental factors in slosh analysis are the natural frequency and the damping of the propellant in the tanks. Since the flight normally begins with the tanks full and ends with the tanks empty, the LV must be designed to be stable over a sizable range of frequencies. The design of an active damping system is a challenging task because of the large numbers of modes involved. For cylindrical tanks with a large slenderness ratio, the slosh mass for the first mode remains fairly constant as the propellant is depleted from the tank, until the free surface is about one diameter in height from the bottom of the tank, then the slosh mass decreases rapidly. On the Saturn V, several of the propellant tanks had significant sloshing modes with frequencies of about 0.5 Hz. [9] The magnitude of the second slosh mode is less than 3% of the first mode and the magnitude of successively higher slosh modes is even less. Therefore, in most LV considerations with large slenderness ratio tanks, all slosh modes higher than the first are usually neglected. Tanks with small slenderness ratios have the pronounced undesirable characteristic of increased slosh mass and decreased natural frequency. The increased slosh mass magnifies instability depending on its location along the vehicle longitudinal axis.

The longitudinal vibration can couple with the sloshing propellant and the propulsion system. Pogo (NASA SP-8055; 1970) was an early example of adverse coupling between the structure, propulsion system, and propellant (see Lehtinen and Lorenzo, Reference 115). In pogo, the longitudinal structural vibrations, excited by combustion in the propulsion system, set up a pressure wave in the propellant. The propellant lines amplify this pressure wave and transfer it to the propulsion system. This

results in a thrust oscillation which further drives the longitudinal vibration of the structure. The Saturn V developed a 5-Hz longitudinal pogo oscillation. The pogo suppression system for early Saturn LV's added accumulators in the fuel system to detune and desensitize it. As design goals, the ratio of slosh frequency to control frequency should be  $>1$ , and the ratio of slosh frequency to fundamental bending frequency should be  $<1$ . Desirable control frequency is 10% of the first bending mode frequency; however, this is not always possible because of low frequencies of this mode. Compartmentation of tanks allows a shifting of fluid frequencies to avoid proximity to other resonant frequencies, and at the same time reduces the magnitude of the sloshing mass.

Sloshing can be modeled as a series of harmonic oscillators, one for each slosh mode, and repeated for each tank. Figure 203 [116] shows a typical modeling scheme. The spring-mass mechanical analogy requires the introduction of a couple at the base of the tanks; its magnitude is a linear combination of the displacement and acceleration of the spring-mass. Thus, the influence coefficient matrix and the inertia matrix are modified accordingly. [116, 117]

In a rigid vehicle with ideal control and one tank, a danger zone for instability occurs when the slosh mass is located between the cg of the vehicle and the instantaneous center (IC) of rotation with respect to the engine gimbal point, where the vehicle has only an angular velocity. For a typical LV, the IC is on order of 10 to 15 meters forward of the cg. For increasing slosh mass in this region, more damping is required to guarantee stability, and the maximum amount of damping is required when located at the midpoint between the IC and cg. This is generally the region where maximum bending displacements occur also. [118]

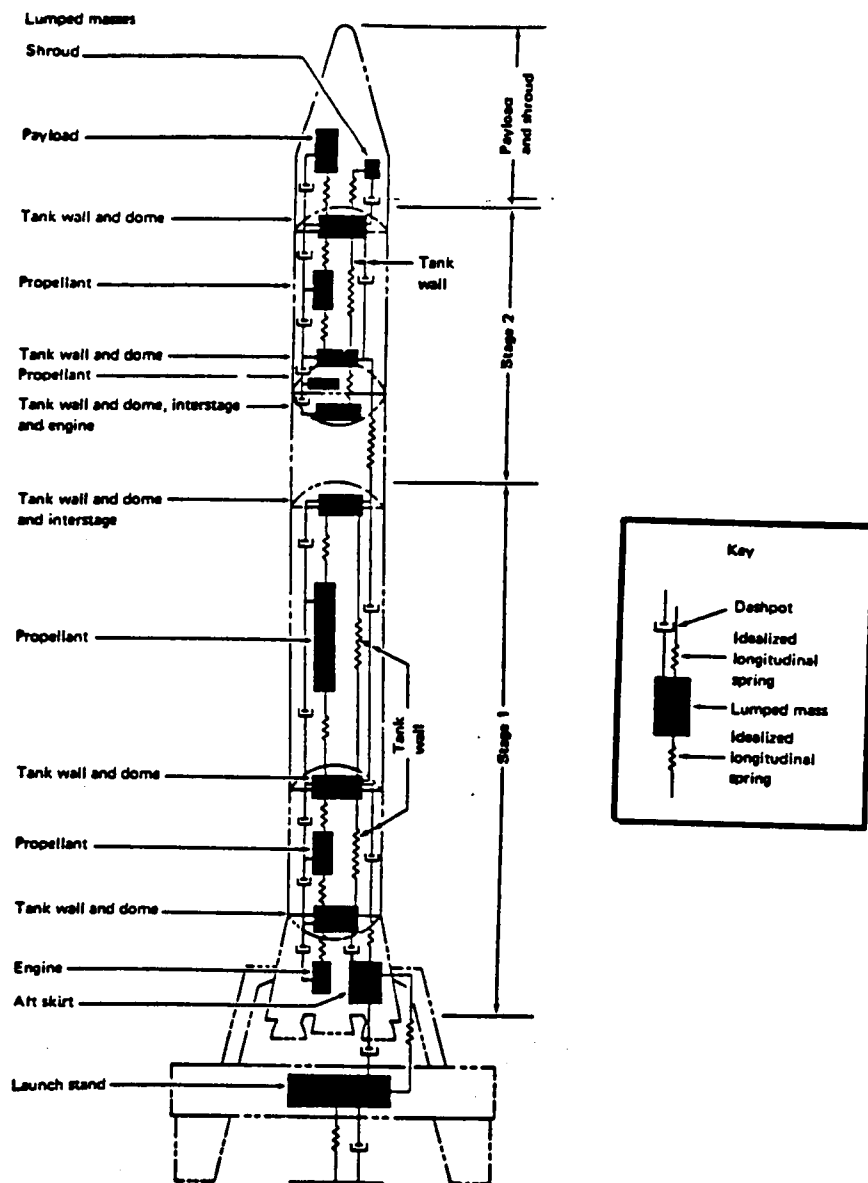


Figure 203. LV longitudinal dynamic model.

In the elastic vehicle, the danger zone for instability increases both forward of the IC and aft of the cg, and also, more damping is required as compared to the rigid vehicle. Ideally, the stability can be improved by locating the slosh mass in the region aft of the cg if there is a large separation between the slosh frequency and the control frequency.

In the author's previous slosh studies (Shuttle-C launch vehicle), it was found that the LOX slosh may be significant when the slosh mass / total mass  $> 0.1$ . This occurs in



second stage flight, after the solid rocket booster separation; the damping factor was determined to be  $> 3\%$  which was adequate for this LV. It was found that this critical mass ratio was reached at approximately 80 seconds after liftoff.

### 11.3 Elastic Model

The basis of the formulation used here is the reduction of the continuous LV system to a finite DOF system by utilization of a mechanical analogy of propellant sloshing and a normal mode superposition analysis. The NASA NASTRAN (Nasa STRuctural ANalysis) FEM normal modes analysis and frequency response has been used here to generate the modal data for this vehicle, using a free-free, branched beam model, incorporating the NASTRAN subroutines CBAR, CBEND, MASS normalization, and no damping. [119] The NASTRAN element formulations used for the dynamic analysis are based on the infinitesimal strain-displacement relations, and the FEM based on variational principles. The FEM is the representation of the complex LV structure by an assemblage of small but finite elements that are connected by a finite number of discrete points called nodes. The behavior of the LV is characterized by the value of the generalized displacements at these discrete points. The accuracy of the solution is dependent on the number of node points used as well as the distribution of the node points. The NASTRAN FEM simulates the structural properties with matrix equations. The structural matrices are generated internally based on user input of grid point locations, element connectivities, element and material properties. The NASTRAN language is Direct Matrix Abstraction Programming (DMAP). [119]

The basic types of dynamic analysis capabilities available in MSC/NASTRAN (an advanced version of NASTRAN) include normal mode, transient response, frequency response, aeroelastic and flutter analyses, and enforced motion. The dynamic load governs the solution approach. In many cases, the natural frequencies and mode shapes of

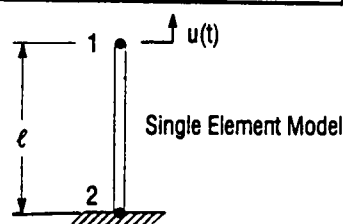
a LV provide enough information to make design decisions. Forced response is the next step in the dynamic evaluation process. MSC/NASTRAN performs real eigenvalue ( $\lambda$ ) analysis (undamped free vibration), linear frequency response analysis (steady-state response of linear structures to loads that vary with frequency), and linear transient response analysis (response to loads that vary with time), and aeroelasticity. Although the results of a  $\lambda$  analysis are not based on a specific loading, they can be used to predict the effects of applying various dynamic loads. The determination of the principal stresses and directions form the eigenvalue problem. [119]

Seven methods of  $\lambda$  extraction are provided in MSC/NASTRAN which belong to either the transformation or tracking methods. In the transformation method, the  $\lambda$  equation is first transformed into a special form from which the  $\lambda$ 's may be easily extracted. In the tracking method, the  $\lambda$ 's are extracted one at a time using an iterative procedure. Table 33 compares methods. [119] The eigenvalues have been extracted here using Lanczos and Rayleigh-Ritz iterations. Each node is associated with a geometric point of the model. The behavior of the element is then expressed as the generalized displacement DOF and generalized forces are defined only at the node points of the element. NASTRAN mass can be formatted as lumped mass or coupled mass. The difference in axial mass formulation of a fixed-free rod are demonstrated in Figure 204. [119] As the mesh of the model becomes finer, the difference in mass formulations becomes negligible.

The finite elements that have been used to model both the forward cg and the aft cg configurations of the experimental LV include those that exhibit bending and torsion. For this reason, each node point has six DOF, three translational and three rotational. The node point can thus be considered to be a spherical ball joint with six rigid-body DOF. The DOF at the nodes are then connected by an element. A set of generalized forces is

Table 33. Comparison of eigenvalue methods.

|                             | Method                                                                                          |                                                                                              |                                                      |                                                  |                                           |
|-----------------------------|-------------------------------------------------------------------------------------------------|----------------------------------------------------------------------------------------------|------------------------------------------------------|--------------------------------------------------|-------------------------------------------|
|                             | Givens,<br>Householder                                                                          | Modified<br>Givens,<br>Householder                                                           | Inverse<br>Power                                     | Sturm<br>Modified<br>Inverse Power               | Lanczos                                   |
| Reliability                 | High                                                                                            | High                                                                                         | Poor (can miss<br>modes)                             | High                                             | High                                      |
| Relative Cost:<br>Few Modes | Medium                                                                                          | Medium                                                                                       | Low                                                  | Low                                              | Medium                                    |
| Many Modes                  | High                                                                                            | High                                                                                         | High                                                 | High                                             | Medium                                    |
| Limitations                 | Cannot analyze<br>singular [M]<br><br>Expensive for<br>problems that<br>do not fit in<br>memory | Expensive for<br>many modes<br><br>Expensive for<br>problems that<br>do not fit in<br>memory | Can miss<br>modes<br><br>Expensive for<br>many modes | Expensive for<br>many modes                      | Difficulty with<br>massless<br>mechanisms |
| Best<br>Application         | Small, dense<br>matrices that fit<br>in memory<br><br>Use with<br>dynamic<br>reduction          | Small, dense<br>matrices that fit<br>in memory<br><br>Use with<br>dynamic<br>reduction       | To determine a<br>few modes                          | To determine a<br>few modes<br><br>Backup method | Medium to<br>large models                 |



Theoretical Natural Frequency:  $\omega_n = \frac{\pi}{2} \frac{\sqrt{E/\rho}}{l} = 1.5708 \frac{\sqrt{E/\rho}}{l}$

MSC/NASTRAN Lumped Mass:  $\omega_n = \sqrt{2} \frac{\sqrt{E/\rho}}{l} = 1.414 \frac{\sqrt{E/\rho}}{l}$

Classical Consistent Mass:  $\omega_n = \sqrt{3} \frac{\sqrt{E/\rho}}{l} = 1.732 \frac{\sqrt{E/\rho}}{l}$

MSC/NASTRAN Coupled Mass:  $\omega_n = \sqrt{12/5} \frac{\sqrt{E/\rho}}{l} = 1.549 \frac{\sqrt{E/\rho}}{l}$

Figure 204. Comparison of mass formulations for a ROD.

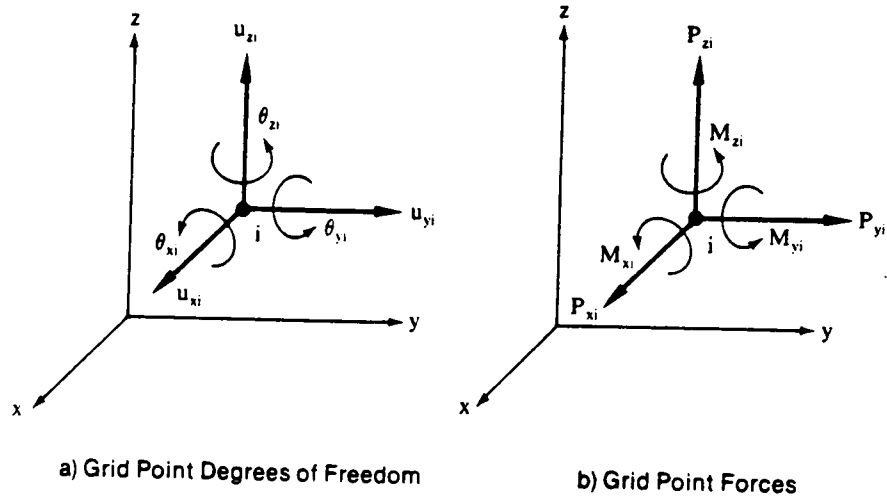


Figure 205. Node point displacements and forces.

associated with the modal displacements as shown in Figure 205. [119] The displacement set notation is

$$u_i = \begin{bmatrix} u_1 \\ u_2 \\ u_3 \\ u_4 \\ u_5 \\ u_6 \end{bmatrix} = \begin{bmatrix} u_x \\ u_y \\ u_z \\ \phi_x \\ \phi_y \\ \phi_z \end{bmatrix}$$

Thus, node point DOF are referred to by the numbers 1 through 6 at a geometric grid point. The stiffness formulation of the element behavior is  $k_{ee} u_e = P_e$  where  $k_{ee}$  is the elemental stiffness matrix and is defined with respect to a set of local elemental coordinates. The stiffness coefficient  $k_{ij}$  is the force at and in the direction of DOF  $\bar{i}$  due to a unit displacement of DOF  $\bar{j}$ , with all other displacement DOF equal to zero.

While stiffness coefficients can be obtained by direct methods for simple 1-D and 2-D elements, the variational approach is used for higher DOF models.

Stiffness coefficients for simple beam elements can be obtained by direct solution of the beam bending differential equation

$$E I \frac{d^4 v}{dx^4} = p(x) \quad (100)$$

where  $I$  is the moment of inertia of the section and  $v(x)$  is the lateral displacement of the neutral axis, and  $p(x)$  is the distributed load. [113, 117] The stiffness coefficients for the beam element are the forces associated with the sets of deflection shapes. This set of node point displacements can also be interpreted as boundary conditions (BC). The stiffness coefficients can then be obtained by solving with each of four sets of BC with  $p(x) = 0$  to obtain an appropriate deflection function, determining the moment and shear as

$$M = E I \frac{d^2 v}{dx^2} \quad \text{and} \quad V = - \frac{dM}{dx} \quad (101)$$

evaluating  $M$  and  $V$  at the ends of the beam element, and relating the beam moment and shear to the generalized node point forces.

The variational formulation based on assumed displacements is

$$- \int_V \sigma_e^T \delta \bar{\epsilon}_e dV + \int_V \bar{\kappa}_e^T \delta \bar{v}_e dV + \int_A \bar{T}_e^T \bar{v}_e \delta dA + \bar{P}_e^T \delta \bar{\mu}_e = 0 \quad (102)$$

where

$\bar{P}_e$  = generalized external forces acting at the node points

$\bar{\sigma}_e$  = set of stresses in the element

$\bar{\epsilon}_e$  = set of strains in the element

$\bar{X}_e$  = set of body forces

$\bar{T}_e$  = set of surface tractions

Approximation functions are chosen to represent the element displacement in terms of generalized DOF. The evaluation of the stiffness matrix for an element is an integration over the volume once the shape functions have been defined. (MSC/NASTRAN aeroelastic analysis utilizes a structural model, an aerodynamic model, and interpolation of deflections, which is an automated procedure to relate mathematically the aerodynamics to the structural DOF using splining techniques.) The structural load distribution on an elastic vehicle in trimmed flight is determined by solving the equations of static equilibrium which leads to aerodynamic stability derivatives and control surface deflections. [116]

On the Saturn V, a 30 DOF beam-rod model was used to determine full-scale testing requirements. The greater the numbers of DOF, the greater the accuracy of the analysis. In actuality of course, a flexible LV has an infinite number of DOF. NASTRAN gives relative displacements so that individual actual displacements can be calculated as a function of initial conditions. For branched-beam attachments, when two beam elements are connected to the same node, coupling with respect to both translation and rotation occurs. The branched-beam representation can be seen in Figure 206. [118] In these instances, the node feels the effect of both beams, and the corresponding elements in this portion of the stiffness matrix are obtained by adding the corresponding elements of the two individual stiffness matrices where they overlap at the common node. Eigenvalues can be determined by: energy methods; solving the differential equations; or solving the integral equations.

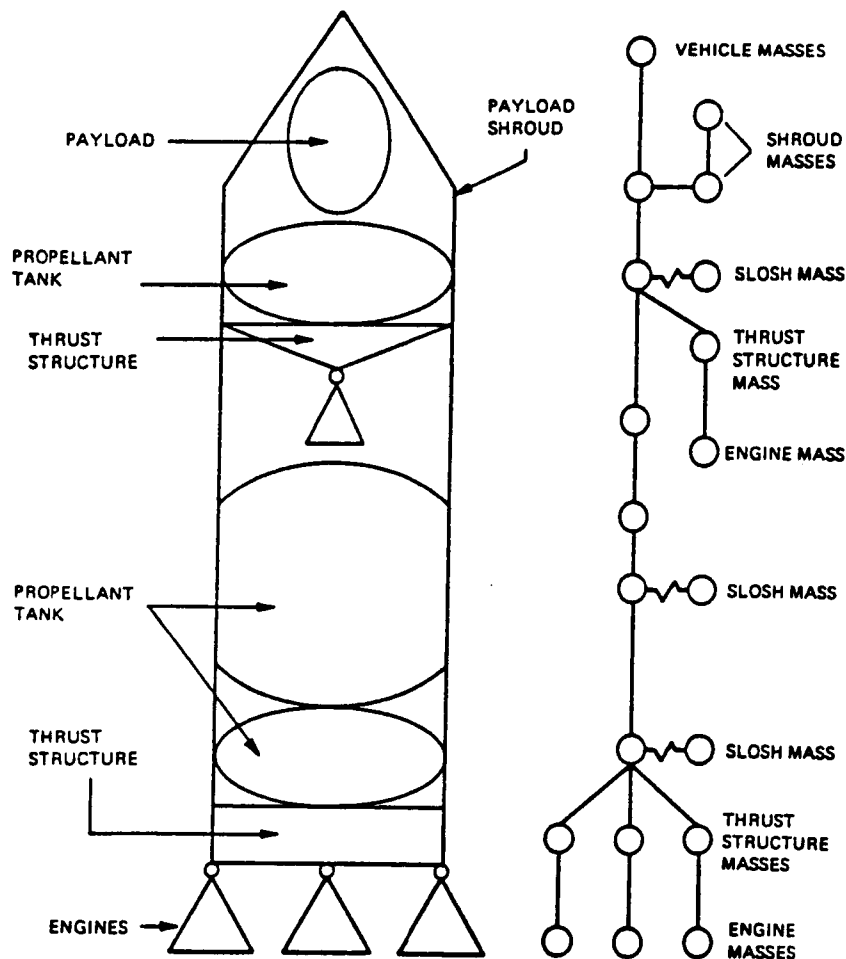


Figure 206. Branched system example.

In the analysis of the experimental LV, it is assumed that, since the aerodynamic FCA's are not primary controls and are small enough and stiff enough, additional DOF are not required.  $E = 10 \times 10^6$  for Al 2219 was used. The gross elastic effects of the LV can reduce the FCA effectiveness and reduce the longitudinal static stability as the neutral point moves forward. Normal modes and 1388 DOF have been used here for the experimental LV at two trajectory critical points: liftoff and max Q conditions. Two

configurations of the experimental LV have been studied and compared, one with the original forward cg and one with the aft cg.

#### **11.4 Resulting Frequencies and Mode Shapes**

The natural frequencies ( $\omega_n$ 's) of the LV, those at which it naturally tends to vibrate if subjected to a disturbance, along with the vehicle mode shapes are functions of the structural properties and boundary conditions (BC). If the structural properties change, the  $\omega_n$ 's change but the mode shapes may not. If the BC change, then the  $\omega_n$  and the mode shapes both change. This can be seen below along with the most significant modes. Three sets of each mode have been produced in translation, and three sets of each mode have been produced in rotation. (Reference 113 suggests that the important modes can be evaluated and used to select the appropriate time or frequency step for integrating the equations of motion. The dynamic analysis can be compared to wind tunnel test results. A normal modes analysis can be used to guide the aeroelastic wind tunnel tests on a flight vehicle. In the pretest planning stages, a normal modes analysis can be used to indicate the best location for accelerometers and gyros. After the test, a normal modes analysis can be used to correlate the wind tunnel test and simulation results.) [113]

This investigation compares the two experimental LV design configurations, one with the original forward cg and the other with the aft cg. The vehicle design changes of this research (detailed in Chapter V) which produced the aft cg include: LOX tank aft vs. LOX tank forward; added payload; deleted propellants, propellant feedlines, and intertank structural members, which are reflected in the different  $\omega_n$ 's. The orthogonality property of normal modes ensures that each mode is unique. In the results below, zero natural frequency indicates one of the rigid-body modes. Due to computer roundoff, the zero frequency  $\omega_n$ 's are on the order of  $1.0 \text{ E-04 Hz}$ .



An important characteristic of these normal modes is that the scaling or magnitude of the eigenvectors is arbitrary. These mode shapes are fundamental characteristic shapes of the LV configurations, hence the relative quantities where each eigenvector has been scaled to a unit value of generalized mass are depicted below. This normalization approach yields very small eigenvalues for a large heavy LV. Repeated root  $\omega_n$ 's occur for plane-of-symmetry vehicles and the eigenvalues are not unique since many orthogonal sets can be found. The modal methods or modal superposition can be used to determine forced response using the modal results. In this investigation, attention has been focused on parameters important for stability and control analyses, for which the LV frequencies of interest are generally below 20 Hz and quite often below 10 Hz. The test LV has been represented by a series of lumped masses connected by elastic beams. Branched-beams have been included to represent the multiple load paths, and concentrated masses have been attached by translational and rotational springs.

Eigenvalues and eigenvectors have been calculated, and mode shapes produced for the longitudinal (x-direction), lateral (y-direction and z-direction), and three rotational directions. Only the lateral deflections (z-direction) are shown in the figures below.

Table 34 lists the natural frequencies that are below 20 Hz for both the forward and aft cg configurations during trajectory critical points of liftoff and max Q. It can be seen that for the aft cg vehicle, the number of natural frequencies below 20 Hz has increased by 12% during liftoff, and has increased by 64% during max Q in the ascent trajectory. The lowest bending mode frequency has increased by 1.4% during liftoff, but has decreased by 2.8% during max Q for the aft cg configuration.

Table 34. Experimental LV natural frequencies (Hz)

| Forward cg |         | Aft cg  |         |
|------------|---------|---------|---------|
| Liftoff    | Max Q   | Liftoff | Max Q   |
| 2.5252     | 2.9227  | 2.5609  | 2.8420  |
| 2.6797     | 3.1381  | 2.6437  | 2.9202  |
| 4.9504     | 5.3889  | 5.0778  | 5.1306  |
| 5.1078     | 5.731   | 5.1852  | 6.4921  |
| 7.1450     | 8.7203  | 6.4479  | 7.7311  |
| 7.8363     | 9.3890  | 6.8917  | 8.3856  |
| 8.0998     | 9.4334  | 10.4543 | 11.5663 |
| 11.3736    | 14.4957 | 11.5939 | 11.6223 |
| 11.8005    | 15.2295 | 11.8976 | 12.6215 |
| 12.5657    | 15.9771 | 12.6223 | 13.2064 |
| 13.0057    | 18.7243 | 13.1882 | 13.7049 |
| 13.4119    |         | 13.6006 | 13.9265 |
| 15.0153    |         | 14.3102 | 15.1122 |
| 15.4947    |         | 15.1381 | 15.5340 |
| 15.6297    |         | 15.2189 | 16.0404 |
| 16.7443    |         | 16.1105 | 16.8734 |
| 19.8676    |         | 16.7120 | 17.4705 |
|            |         | 19.4097 | 19.8484 |
|            |         | 19.8079 |         |

Figure 207 lists the significant natural frequencies and shows the corresponding significant mode shapes for the forward cg configuration of the experimental LV at liftoff.

Figure 208 lists the significant natural frequencies and shows the corresponding significant mode shapes for the forward cg configuration of the experimental LV at max Q.

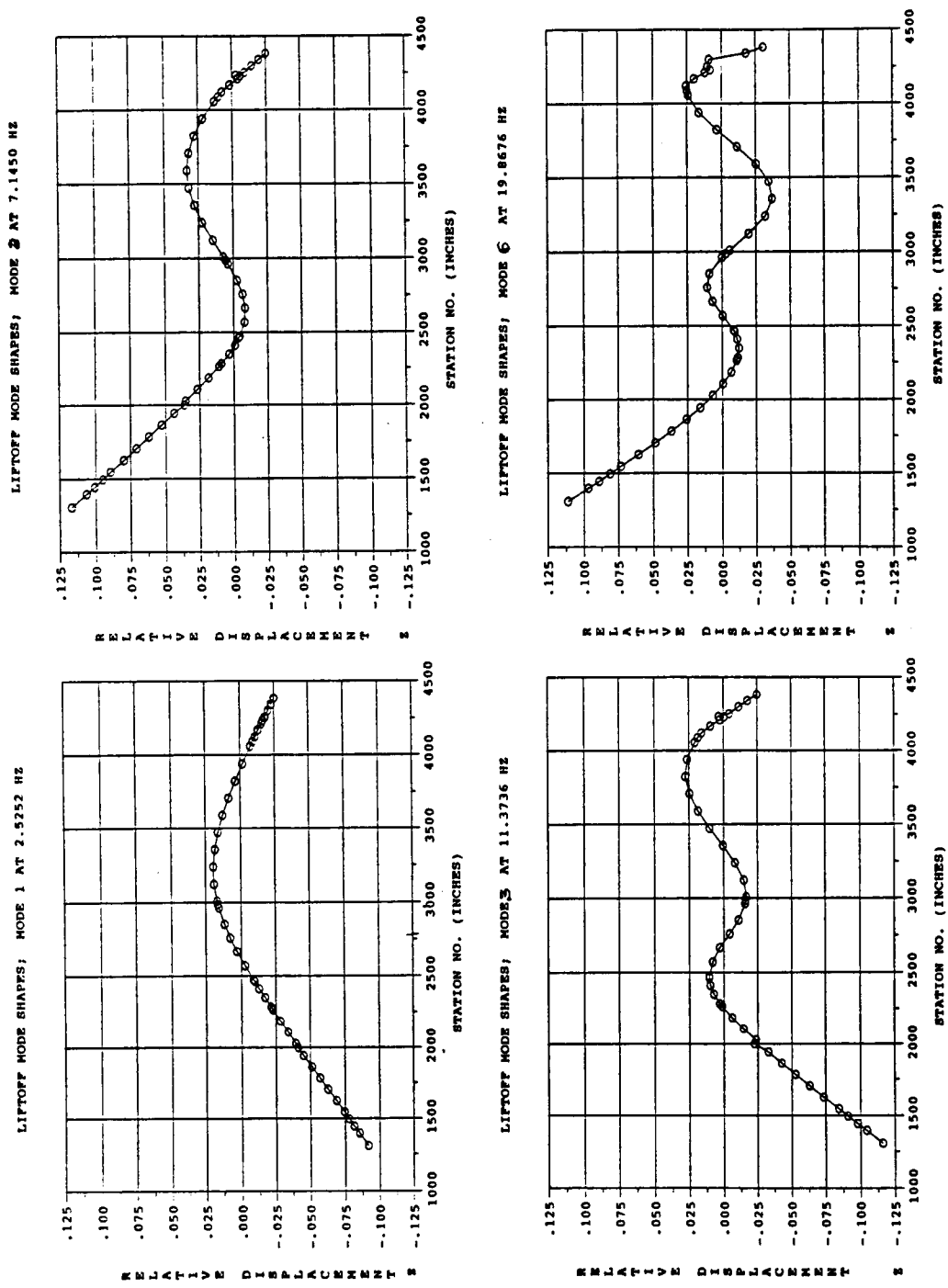


Figure 207. Forward cg mode shapes at lift-off.

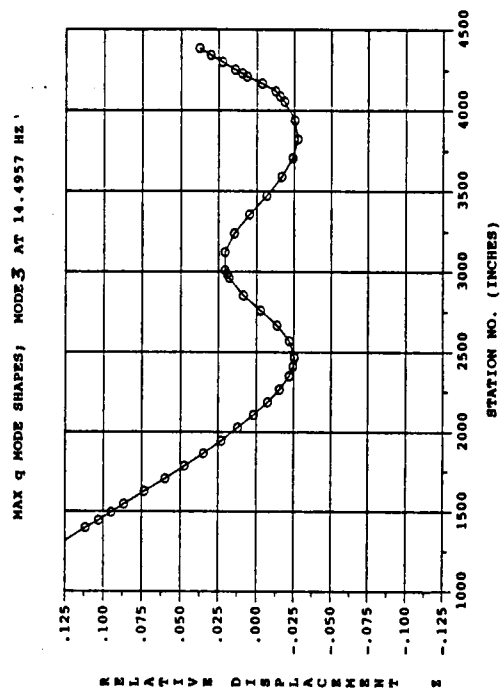
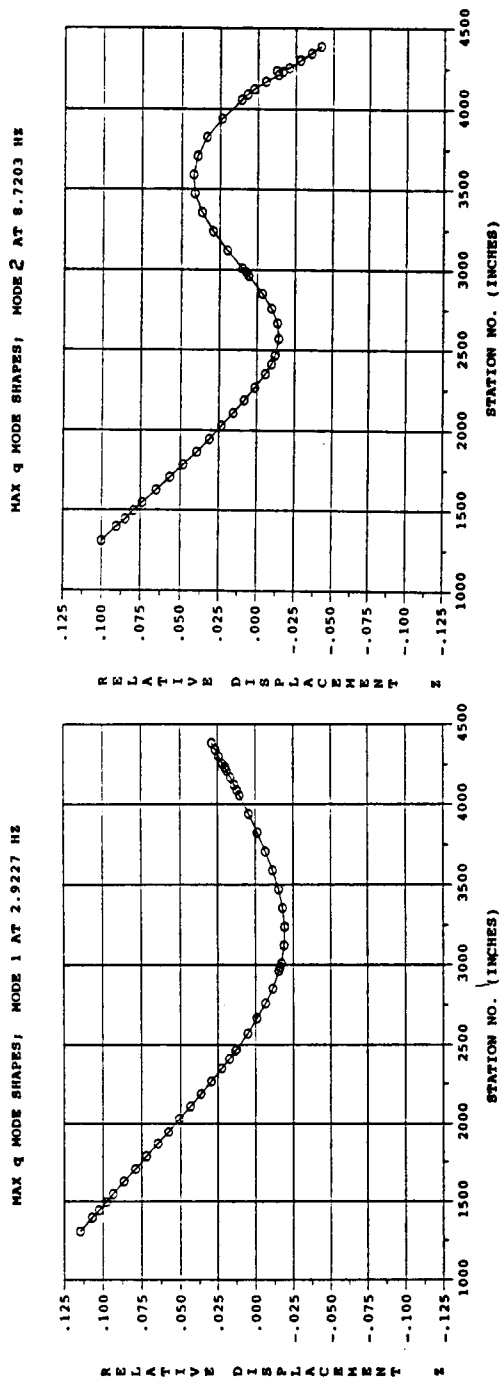


Figure 208. Forward cg mode shapes at max Q.

Figure 209 lists the significant natural frequencies and shows the corresponding significant mode shapes for the aft cg configuration of the experimental LV at liftoff. Figure 210 lists the significant natural frequencies and shows the corresponding significant mode shapes for the aft cg configuration of the experimental LV at max Q. Figure 211 shows the response spectrum for the forward cg vehicle, and Figure 212 shows the response spectrum for the aft cg vehicle. It can be seen that most of the dynamic response is 150% greater in the aft-cg experimental LV for both liftoff and max Q conditions.

Figure 213 [112] shows the first three bending modes of the Saturn V for comparison. The first bending frequency for Saturn I was 2.83 Hz, for Saturn S-II 2.085 Hz, and for Saturn S-IVB 7.67 Hz. In comparison, the B-52 aircraft elastic modes start at 0.8 Hz and for a C-5A aircraft start at 0.75 Hz. Figure 214 [41] shows Saturn control frequencies for the first three stages of flight.

As further work, analytical results could be coupled with experimental data. Azevedo [120] couples the structural dynamics with an unsteady flow solver for the transonic portion of a LV ascent trajectory. The coupling of the Navier-Stokes equations and the structural deflection equations is performed by integrating both of them simultaneously in time and ensuring that the data generated by one set is used in the next time step of the other. The aerodynamic equations are solved in a time-accurate fashion, and their solutions provide the forcing terms for the aeroelastic analysis, which are weighted integrals of the pressure distribution along the fuselage. Solution of the structural deflection equations gives the new mode shape of the body, and so the BC's for the aerodynamic solution at the next time step.

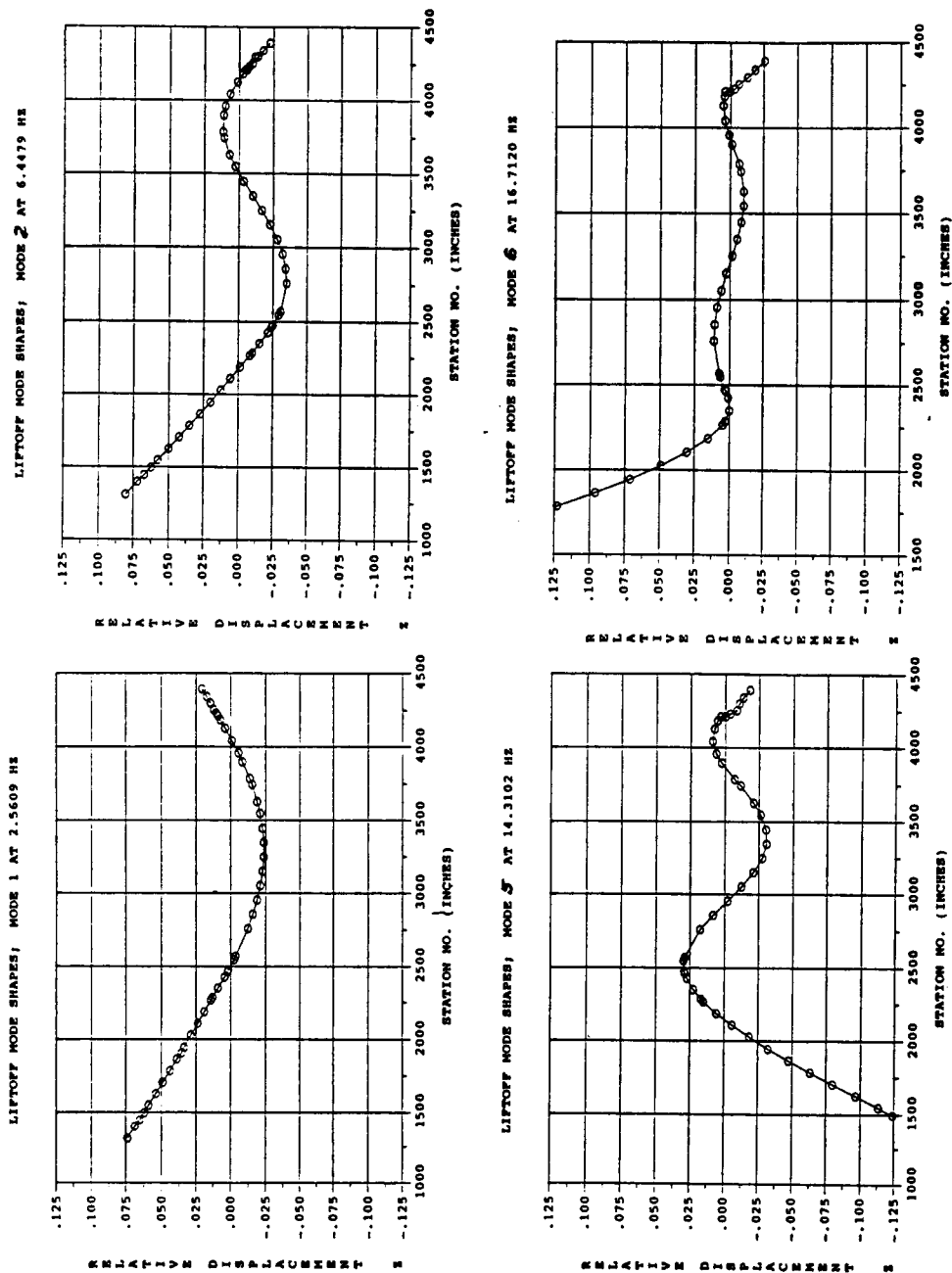


Figure 209. Aft cg mode shapes at liftoff.

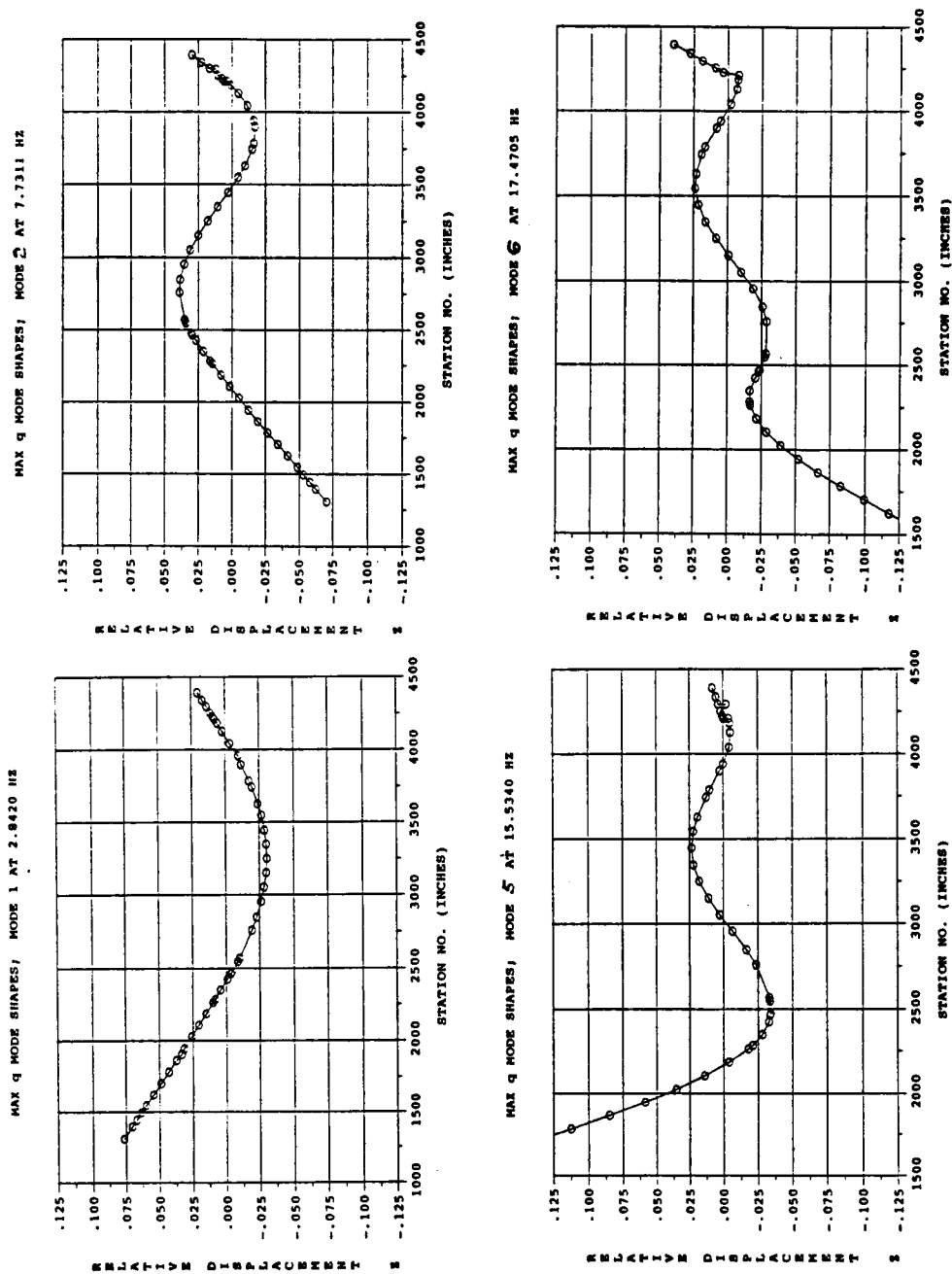


Figure 210. Aft cg mode shapes at max Q.

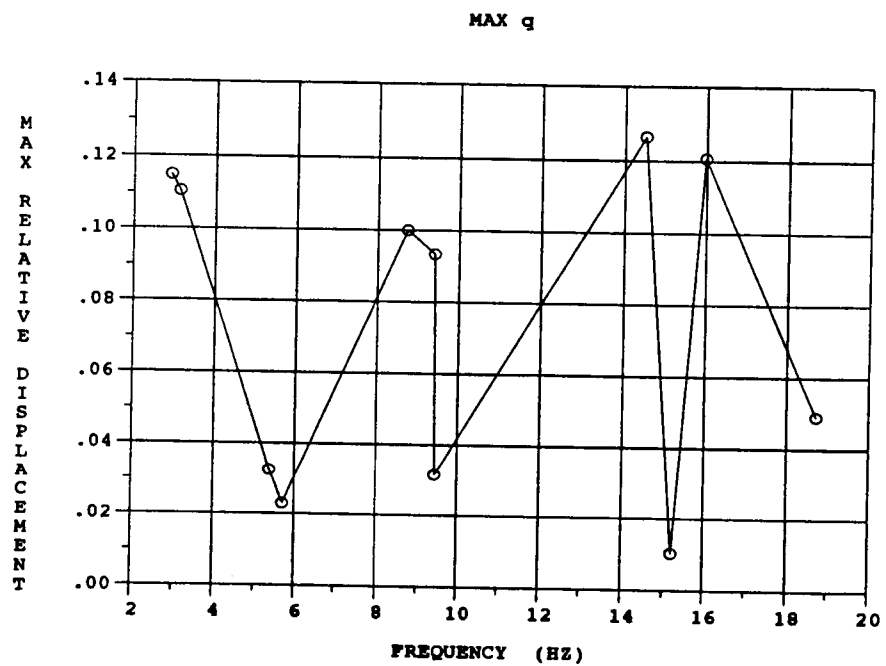
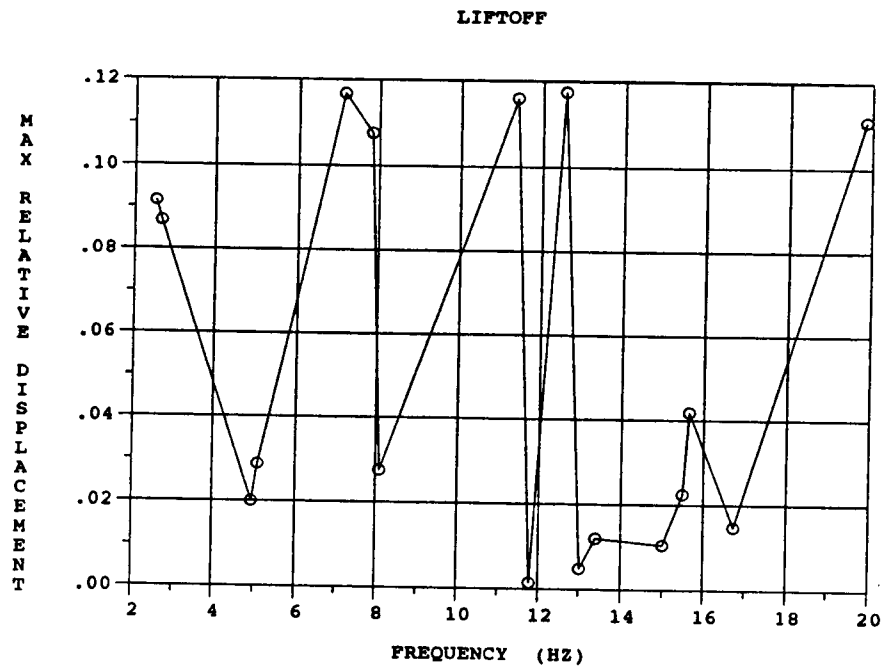


Figure 211. Response spectra for forward cg experimental LV.



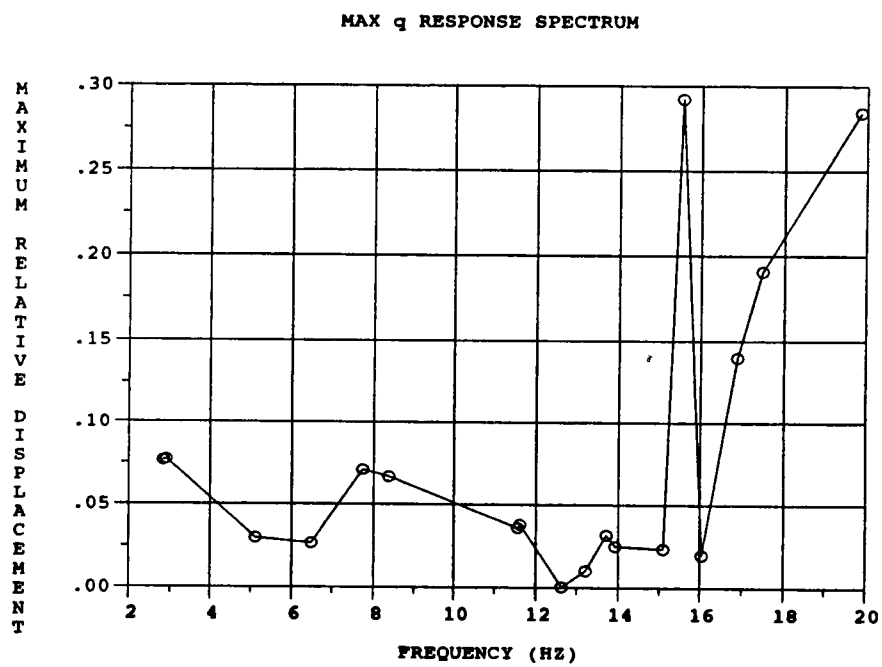
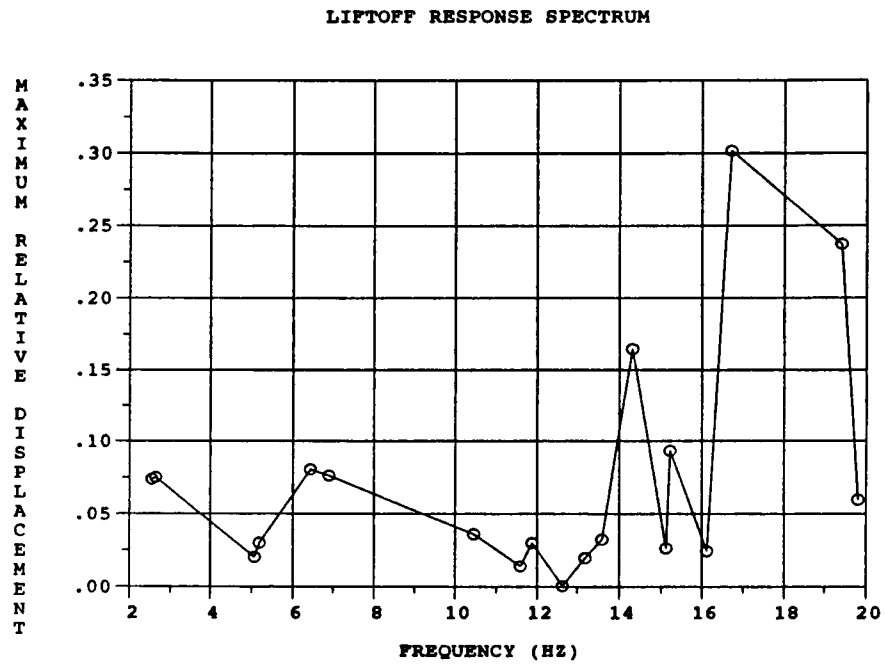


Figure 212. Response spectra for aft cg experimental LV.

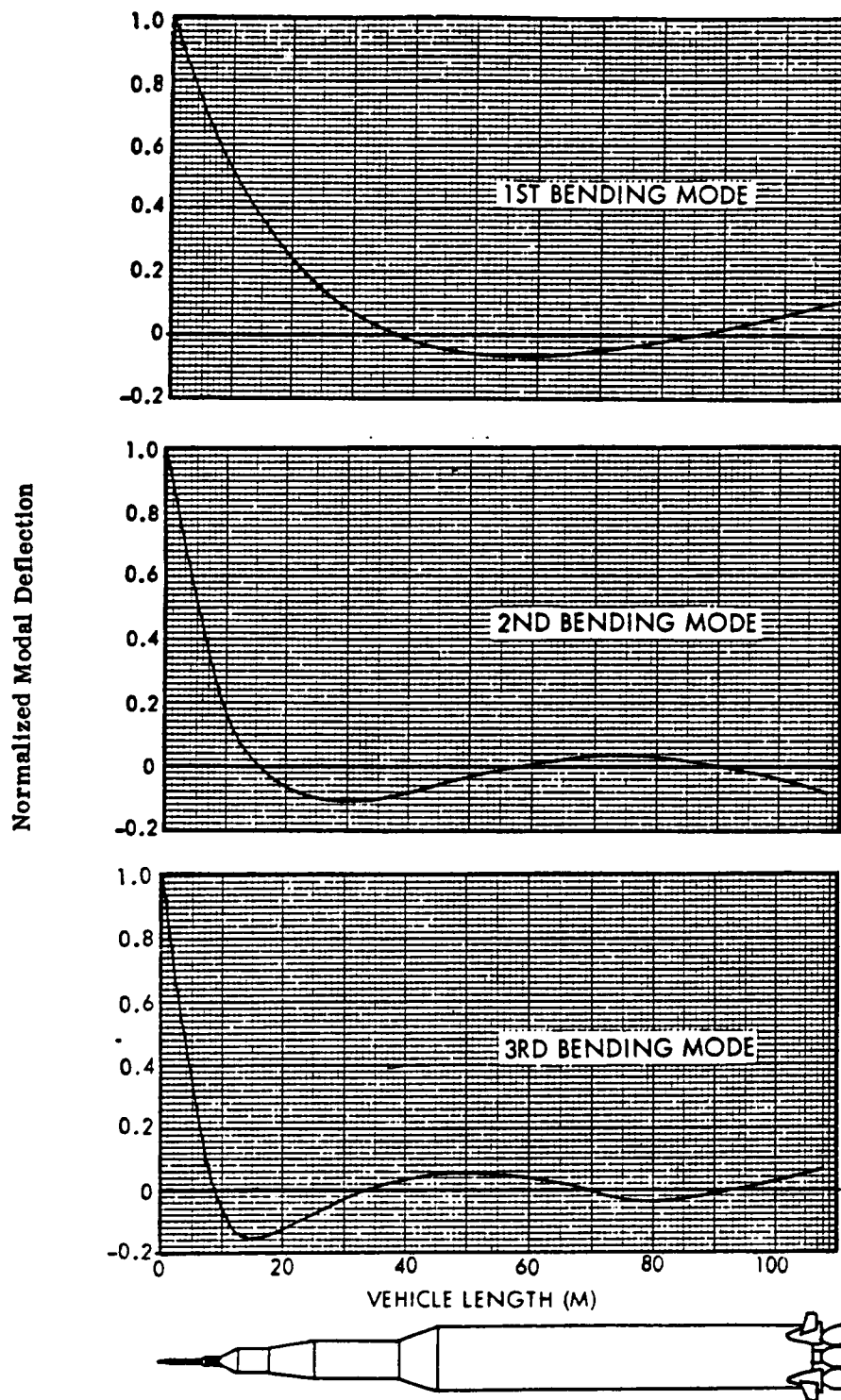


Figure 213. Saturn V free-free bending modes.

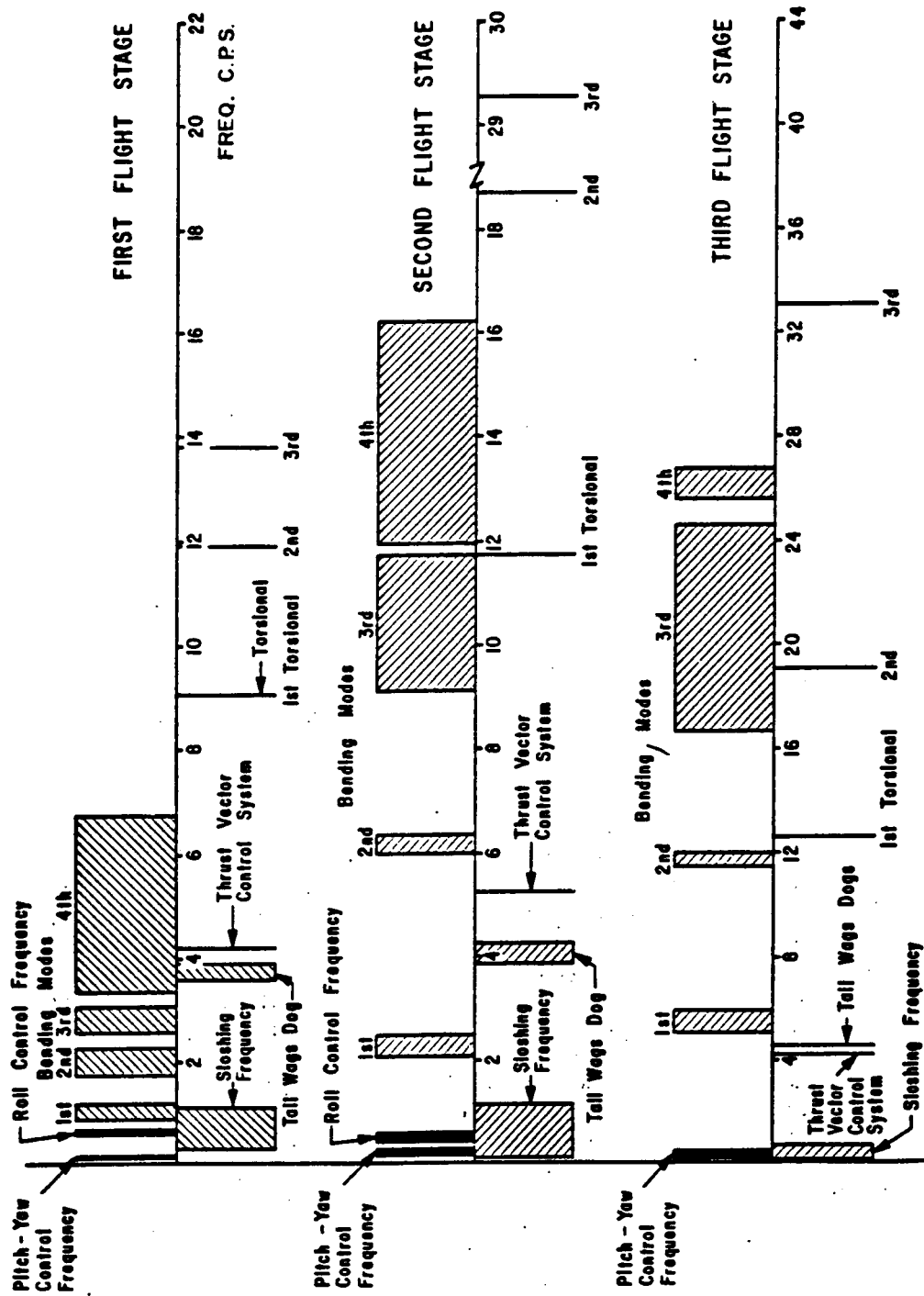


Figure 214. Saturn vehicle frequencies.

In summary, critical to high slenderness ratio LV's is an assessment of elastic body modes of vibration and propellant slosh damping. Aerodynamic damping analysis can be performed by employing the free-free bending modes along with the LV trajectory data. Propellant slosh frequency should be as far above the control frequency as possible to prevent resonance, and LOX slosh may be significant when the slosh mass / total mass  $> 0.1$ . As the slenderness ratio increases, the lowest bending frequency decreases.

In this investigation, the natural frequencies have been calculated for both the original forward cg and the aft cg configurations of the experimental LV, at the liftoff and max Q points of the ascent trajectory. Significant modes have been identified and compared for both configurations at liftoff and at max Q. The response spectra have been produced for both configurations at both critical points in the LV ascent trajectory. These show the frequency separation between the modes. The torsion frequencies and mode shapes were found to be not significant. The lateral bending modes were found to be of much greater significance to the vehicle flight stability than the longitudinal modes.

## **CHAPTER XII. CONCLUSIONS AND FUTURE WORK**

This research effort was a flight mechanics — stability and control, special project conducted by the author at NASA MSFC. The author has conducted comprehensive reviews, proposed, designed, wind-tunnel tested, and analyzed the static and dynamic flight stability of both forward and aft located, all-movable FCA's to provide flight control augmentation for future aft-cg LV's that would not be controllable with engine gimbaling alone. To the author's knowledge all-movable FCA's have never been used for flight control augmentation on a large Saturn-class LV.

This research included: a review of our national heritage of LV's that used aerodynamic surfaces and their design concepts; a review of current usage of these surfaces by other nations; calculation of stability derivatives and control derivatives from both the wind tunnel test programs data and the U.S. Air Force DATCOM computer code; analyses and quantification of enhanced static and dynamic flight stability of the experimental aft-cg LV when treated with the designed FCA's; and consideration of lightweight FCA materials, innovative FCA actuators, changes in aeroelastic effects, and future work. Specific conclusions resulting from this investigation are summarized below.

### **12.1 Conclusions**

1. Many future aft-cg LV's that may not be controllable with engine gimbaling alone can be controlled with the use of the FCA's designed in this research.
2. The designed FCA's can also be used in marginal control configurations to enhance controllability, as load alleviators, to reduce engine gimbaling requirements, to provide engine actuator failure protection, to reduce wind requirements, and to enhance crew safety and vehicle reliability by providing more control in engine-out events.

3. The greatest need for flight control augmentation is near the max Q point of the ascent trajectory. This is also when the FCA's would be of greatest service.
4. Although not used in the U.S. today, NASA has a rich national heritage of LV's that have used aerodynamic surfaces, including the V-2, Pershing, Redstone, Saturn I, Saturn IB, and the Saturn V which took us to the Moon with 300 ft<sup>2</sup> of aerodynamic surfaces.
5. Aerodynamic surfaces are, however, currently being used on large Saturn-class LV's by many other foreign nations to provide increased flight stability.
6. This research has shown that all-movable FCA's have never been used for flight control augmentation on a large Saturn-class LV. The author has filed patent disclosures on the use of FCA's to provide the required control augmentation for a large Saturn-class aft-cg LV.
7. The X-15 was comparable in performance to the Redstone, and achieved increased control effectiveness through the use of the blunt TE design. The blunt TE design has been investigated here for use on the FCA's designed in this research.
8. As a result of the conducted control requirements analyses, all-movable, forward and aft, FCA's incorporating blunt TE's have been designed.
9. It has been shown that FCA drag is not an issue for the LV in ascent with increased drag < 2%. For the experimental LV,  $S_F / S_{\text{wetted}} = 1.6\%$ ; the FCA flight time is 4 minutes; and the thrust available is 3,500,400 lb. Since drag is not an issue, blunt TE's can be used.
10. It has been determined that an aft FCA planform area of 80 ft<sup>2</sup> can accommodate a very far aft cg ( > 83% ) and an engine-out event. Canards of 32 ft<sup>2</sup> have also been designed as forward FCA's. A total of 12 candidate FCA's have been designed and wind-tunnel tested to determine the effects of the blunt TE design: 11 aft FCA's all

having the same planform area, but different planform shapes, airfoil sections, and TE thickness, and one set of forward canard FCA's.

11. Static and dynamic wind tunnel test programs were designed to provide the force, moment, and damping data from which the stability derivatives, control derivatives, and damping derivatives have been calculated to assess the static and dynamic stability of the experimental LV with and without the designed FCA's.
12. The LV static margin (SM) was used as the quantitative measure of longitudinal static stability for the various LV configurations and FCA designs tested. The requirements for control torque increase with a growing negative SM. It was found that the basic experimental LV without FCA's was longitudinally statically unstable. The static stability analysis has shown that with the use of the aft FCA's the SM was dramatically improved with all candidate designs. Improvements in SM with the use of aft FCA's ranged from 41% to 72% over that of the LV without FCA's.
13. The static longitudinal stability of the experimental LV with FCA's has been compared to that of the Saturn V. It was found that the static longitudinal stability of the experimental LV with the designed FCA's surpassed that of the Saturn V LV throughout the entire Mach range tested from  $M = 0$  to  $M = 5$ .
14. The best results in increased stability were obtained with the blunt TE, half diamond airfoil,  $\Lambda = 30^\circ$ , aft FCA design.
15. The LV control ratio (CR) was used as the quantitative measure of control authority of the various LV configurations and FCA designs tested. The control requirement is to have the  $CR > 1.5$ . The control analysis has shown that the far aft cg basic vehicle without FCA's had a  $CR = 1.12$ , and was not adequately controllable using TVC alone. With the use of undeflected aft FCA's, the CR increased to 1.64 which

more than satisfied the control requirement and was an improvement in the control authority of 46%. The control authority improvement increased to 110% for 3° FCA deflection; to 346% for 6° deflection; and at 9.1° FCA deflection the vehicle aerodynamic moment was completely trimmed out with the aft FCA's. The forward canard FCA's were able to trim the vehicle at 5° deflection.

16. It was found that the increased control effectiveness of the blunt TE FCA design provided the maximum flight control augmentation. In addition, the blunt TE design was found to be very attractive from a structural standpoint. The increased rigidity aids in flutter prevention, and the internal space allows for possible insertion of RCS or smart actuator mechanism.
17. For the experimental LV considered here in the ascent trajectory, the longitudinal short-period mode was of much more significance than the phugoid mode. Therefore, the LV short-period mode assessment was used as the quantitative measure of dynamic stability of the final LV configurations and FCA designs tested.
18. Dynamic stability data, either experimental or analytical, has been found to be extremely sparse in the literature. It is hoped that this work will contribute to this need.
19. Dynamic stability results have been presented for the basic LV without FCA's, and for the vehicle with three selected FCA's with deflections up to 15° and a Mach range of .8 to 2. It was shown that the pitch damping generally increased up to the max Q point in the ascent trajectory, and then decreased. The short-period analysis showed that the maximum improvement in short-period damping ratio was realized with the use of the blunt TE FCA design.



20. The Air Force Missile DATCOM simulation program (1993 revision) has been used to verify and compare wind tunnel test data. Within the predicted accuracies of the code the comparisons were good.
21. Since the Saturn V took us to the Moon using 300 ft<sup>2</sup> of aerodynamic surfaces, a wealth of smart materials and advanced composites have been developed that enables the design of very lightweight, strong, and innovative LV FCA's.
22. Many of the smart materials can be used as both sensors and actuators, and by physically deforming themselves can produce significant actuation forces. Patents have recently been granted on articulated flight control surfaces using SMA's. The replacement of conventional flight control actuation systems with smart material actuators could result in enormous saving in cost and weight.
23. Nowhere in the literature has the author found any application of smart materials or advanced composites to aerodynamic flight controls of an expendable, large, Saturn-class LV. The author has filed patent disclosures using SMA's for smart actuation of the designed FCA's.
24. The new MMC's can not only provide substantial weight savings and increased high-temperature strength, which allow increased payload capability and flight performance, but can also reduce the thermal protection requirements, thus providing even more indirect weight and cost savings. TMC's appear to be among the candidates in the advanced composites for potential use in LV FCA's. They have already been selected by NASA for the NASP. As part of this research the author studied TMC production on site at the Textron Specialty Materials Company and conducted a Flight Control Augmentor Design Review with Textron officials on the 12 sets of author-designed FCA's. Thus, any added weight concerns have been addressed, and cost estimates have been obtained for the full scale FCA's.

25. Aeroelastic effects on a LV during its ascent trajectory can have profound effects on its flight stability and control. The aeroelastic effects on the Saturn series of LV's have been reviewed and presented. During ascent, the mass of the propellants represent the major part of the LV mass, hence sloshing of the propellants can significantly contribute to fuselage deformations and shift the cg position. Frequencies of the sloshing propellants must be separated from those of the control system to prevent resonance. As the slenderness ratio of the vehicle increases, so do the aeroelastic effects especially at max Q, and the lowest bending frequency decreases. Since bending mode frequencies can couple with the control system frequencies, it is desirable to have the first bending mode frequency much higher than that of the control frequency.
26. Elastic body motion can be conveniently expressed in terms of vibration frequencies and mode shapes of the vehicle. NASTRAN has been used to generate the modal data for both the original and the aft-cg experimental LV. The critical trajectory points of liftoff and max Q were analyzed. Significant mode shapes and frequencies have been identified and presented along with response spectra. It has been shown that for the aft-cg vehicle, the number of natural frequencies below 20 Hz has increased by 12% during liftoff, and has increased by 64% during the max Q point of the ascent trajectory. The lowest bending mode frequency has increased by 1.4% during liftoff and has decreased by 2.8% during max Q for the aft cg configuration. The lateral bending modes were found to be the most significant, and the dynamic response was found to be 150% greater in the aft-cg experimental LV for both the liftoff and max Q conditions.

## 12.2 Future Work

To build upon this research, options for increased control effectiveness could be explored. The Shuttle aerodynamic flight control surfaces become effective approximately 10 seconds into the launch ascent trajectory. As an added way to increase the torque provided by the designed FCA's, RCS could be employed in the blunt TE's, or the concept of jet flaps could be investigated. In the jet flap concept, a propulsive jet can be emitted from the TE of the FCA, with the primary objective of increasing the lift of the surface, hence control moment. A recent investigation has shown that an effective technique for generating control torque when conventional control surfaces lose their effectiveness is to use pneumatic blowing through a chined forebody. These could be explored if the FCA's were required to be more effective sooner than 10 seconds into the ascent. Once the FCA's have ceased to be effective, they could be dropped off, perhaps with the engines, since this is an expendable LV. The logistics of this could be developed.

The concepts of flight control augmentation developed and applied here to an expendable LV with an aft cg, could be utilized to treat many of the configurations of the reusable LV program. Trade studies could be made of the benefits of using FCA's on a LV having adequate controllability. Integration of the FCA's and TVC with common actuators could be developed.

Many of the smart materials can be used as both sensors and actuators, and by physical deformation can produce significant actuation forces. Patents have recently been granted on articulated flight control surfaces using SMA's. The replacement of conventional flight control actuation systems with smart material actuators could result in enormous saving in cost and weight. The development of a FCA actuator utilizing a smart material could be investigated.

As further aeroelastic work, analytical results could be coupled with experimental data. Vehicle structural dynamics has been coupled with an unsteady flow solver for the

transonic portion of a LV ascent trajectory. The coupling of the Navier-Stokes equations and the structural deflection equations can be performed by integrating both of them simultaneously in time and ensuring that the data generated by one set is used in the next time step of the other. The aerodynamic equations could be solved in a time-accurate fashion, and their solution could provide the forcing terms for the aeroelastic analysis, which could be weighted integrals of the pressure distribution along the fuselage. Solution of the structural deflection equations could give the new mode shape of the body, and so the BC's for the aerodynamic solution at the next time step.

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## **APPENDICES**

## **APPENDIX A**

### **EXPERIMENTAL LAUNCH VEHICLE DATA**

**A.1 Preliminary Aerodynamic Data and Trajectory Data  
for the Aft CG Vehicle**

Table A1. Preliminary aerodynamic data used for experimental LV.

| COEFF/MACH        | 0        | 0.6       | 0.8       | 0.9       | 0.95      | 1.05      | 1.1       |
|-------------------|----------|-----------|-----------|-----------|-----------|-----------|-----------|
| $C_{Na} = C_{Na}$ | 0.05     | 0.051638  | 0.052973  | 0.053234  | 0.054478  | 0.056068  | 0.055539  |
| $C_{Ma} = C_{Ma}$ | 0.258425 | 0.258425  | 0.259746  | 0.262482  | 0.271271  | 0.286673  | 0.289559  |
| $C_{Yb}$          | -0.0599  | -0.064256 | -0.067568 | -0.068456 | -0.069352 | -0.070798 | -0.069871 |
| $C_{Ynb}$         | -0.28    | -0.293771 | -0.297957 | -0.301729 | -0.309866 | -0.329174 | -0.333979 |
| $C_{DLb}$         | 0        | 0         | 0         | 0         | 0         | 0         | 0         |
| $CA0$             | 0.2      | 0.221529  | 0.255329  | 0.323363  | 0.380371  | 0.524     | 0.575746  |

|  | 1.15      | 1.25      | 1.46      | 1.96      | 2.74       | 3.48       | 4.96       |
|--|-----------|-----------|-----------|-----------|------------|------------|------------|
|  | 0.058527  | 0.057911  | 0.056917  | 0.061311  | 0.0717236  | 0.07439906 | 0.06947641 |
|  | 0.309339  | 0.303354  | 0.326607  | 0.342068  | 0.383663   | 0.3936375  | 0.3738235  |
|  | -0.076712 | -0.07656  | -0.07395  | -0.078046 | -0.0863027 | -0.0888597 | -0.0693815 |
|  | -0.358316 | -0.350907 | -0.371529 | -0.38846  | -0.4204563 | -0.431989  | -0.3812833 |
|  | 0         | 0         | 0         | 0         | 0          | 0          | 0          |
|  | 0.553637  | 0.578022  | 0.499793  | 0.448496  | 0.3865777  | 0.3223577  | 0.1752813  |

Table A2. Preliminary trajectory data for experimental LV.

| MACH | ALT (ft)  | Q (pfs) | THRST (lbs) | BDRAG (lbs) |
|------|-----------|---------|-------------|-------------|
| 0.00 | 0.00      | 0.00    | 0.00        | 0.00        |
| 0.61 | 9767.00   | 384.20  | 3491867.00  | 37977.70    |
| 0.80 | 14926.00  | 547.30  | 2913900.75  | 41119.57    |
| 0.90 | 19231.00  | 584.80  | 2965691.25  | 41767.07    |
| 0.94 | 21062.00  | 599.10  | 2985669.25  | 42414.58    |
| 1.07 | 25916.00  | 625.30  | 3033270.50  | 43758.52    |
| 1.12 | 27967.00  | 630.10  | 3051234.25  | 40911.27    |
| 1.17 | 30080.00  | 633.90  | 3068509.25  | 36687.49    |
| 1.26 | 33371.00  | 638.90  | 3093054.00  | 29509.55    |
| 1.41 | 37944.00  | 639.90  | 3123076.25  | 15380.07    |
| 1.66 | 45454.00  | 620.70  | 3161505.75  | 5990.87     |
| 2.02 | 56381.00  | 526.00  | 3199134.25  | -5181.30    |
| 2.19 | 62254.00  | 460.30  | 3212174.25  | -10468.95   |
| 2.49 | 73142.00  | 350.80  | 3227807.00  | -15889.43   |
| 2.71 | 81363.00  | 281.20  | 3234955.50  | -15447.31   |
| 2.99 | 91684.00  | 213.70  | 3240632.25  | -13295.85   |
| 3.43 | 108017.00 | 136.00  | 3244574.00  | -9773.67    |
| 3.78 | 121351.00 | 93.70   | 3247423.00  | -7554.01    |
| 4.16 | 135170.00 | 65.20   | 3248522.75  | -6186.50    |
| 4.52 | 145375.00 | 51.10   | 3899016.25  | -6176.20    |
| 5.01 | 158136.00 | 38.60   | 3899394.00  | -4890.88    |



## **A.2 Updated Mass Properties for Very Far Aft CG**

Table A3. Mass properties for 83% aft CG.

|                                                                                                                                      | WEIGHT<br>POUNDS  | X                                        | Y<br>INCHES | Z<br>INCHES | 1X<br>SLUG-FEET-SQUARED               | 1Y<br>SLUG-FEET-SQUARED | 1Z<br>SLUG-FEET-SQUARED |
|--------------------------------------------------------------------------------------------------------------------------------------|-------------------|------------------------------------------|-------------|-------------|---------------------------------------|-------------------------|-------------------------|
| INTERSTAGE ASY<br>STRUCTURE<br>AVIONICS<br>TPS<br>RCS<br>PAYLOAD SEPARATION                                                          | ( )               |                                          |             |             |                                       |                         |                         |
| FWD SKIRT ASY<br>STRUCTURE<br>AVIONICS<br>TPS<br>PURGE MANIFOLD<br>LOX VENT OUTLET<br>AERO VENT                                      | ( )               |                                          |             |             |                                       |                         |                         |
| LOX TANK ASY<br>STRUCTURE<br>SLOSH BAFFLES<br>PRESSURIZATION SYS<br>VENT/RELIEF SYS<br>FEED SYSTEM<br>AVIONICS<br>TPS                | ( -1351)          | 1157.64<br>1327.93                       |             |             | -29<br>-450655                        | -450650                 | 44                      |
| INTERTANK ASY<br>STRUCTURE<br>AVIONICS<br>TPS<br>PURGE MANIFOLD<br>GROUND UMBILICAL<br>CROSS BEAM ASY<br>SRB ATTACH UWD<br>AERO VENT | ( -3710)<br>-3710 | 774.29<br>774.58<br>774.58<br>774.58     | .11         | .00         | -46317<br>-46317<br>-22829<br>-22829  | -39246<br>-39247        | 78<br>70<br>60<br>60    |
| LH2 TANK ASY<br>STRUCTURE<br>AVIONICS<br>TPS<br>FEED SYS<br>PRESSURIZATION SYS<br>VENT/RELIEF SYS                                    | ( -4939)<br>-4939 | -522.23<br>-533.35<br>-533.35<br>-533.35 | .01         |             | -31143<br>-31143<br>-28990<br>-160411 | -29245<br>-160667       | -6                      |
| AFT SKIRT ASY<br>STRUCTURE<br>AVIONICS<br>TPS<br>GROUND UMBILICAL<br>PROPELLANT FEED                                                 | ( )               |                                          |             |             |                                       |                         |                         |

|                           | WEIGHT<br>POUNDS | X       | Y<br>INCHES | Z   | IX<br>SLUG-FEET-SQUARED | IY<br>SLUG-FEET-SQUARED | IZ<br>SLUG-FEET-SQUARED |
|---------------------------|------------------|---------|-------------|-----|-------------------------|-------------------------|-------------------------|
| PROPULSION MODULE         | ( )              |         |             |     |                         |                         |                         |
| STRUCTURE                 |                  |         |             |     |                         |                         |                         |
| AVIONICS                  |                  |         |             |     |                         |                         |                         |
| BASE HEAT SHIELD/TPS      |                  |         |             |     |                         |                         |                         |
| PROP FEED SYS             |                  |         |             |     |                         |                         |                         |
| PRESSURIZATION SYS        |                  |         |             |     |                         |                         |                         |
| PNEUMATIC SYS             |                  |         |             |     |                         |                         |                         |
| TVC                       |                  |         |             |     |                         |                         |                         |
| PURGE & VENT              |                  |         |             |     |                         |                         |                         |
| SEPARATION                |                  |         |             |     |                         |                         |                         |
| INTERFACE & SYS HND       | ( )              |         |             |     |                         |                         |                         |
| CABLE TRAY ASY            |                  |         |             |     |                         |                         |                         |
| PROPELLANT LINES          |                  |         |             |     |                         |                         |                         |
| PRESS SYS LINES           |                  |         |             |     |                         |                         |                         |
| PURGE & PURGE LINES       |                  |         |             |     |                         |                         |                         |
| SYSTEM CABLES             |                  |         |             |     |                         |                         |                         |
| HAZ GAS DETECTION         |                  |         |             |     |                         |                         |                         |
| I/F HND & CLOSURES        |                  |         |             |     |                         |                         |                         |
| AFT SRD STRUTS            |                  |         |             |     |                         |                         |                         |
| ENGINE                    | ( )              |         |             |     |                         |                         |                         |
| TOTAL CORE THERT          | -10000           | 65.50   | .04         | .07 | -77497                  | -2030500                | -2847280                |
| SHROUD                    |                  |         |             |     |                         |                         |                         |
| TOTAL VEHICLE THERT       | -10000           | 55.70   | .04         | .06 | -77497                  | -2199030                | -2215720                |
| PROPELLANT-LO             | 10000            | 1029.14 |             |     | -237240                 | -12073076               | -11909096               |
| PAYLOAD (TARGET + MARGIN) |                  | -95.74  |             |     | 863                     | 862                     | 862                     |
| LO                        |                  | 884.13  |             |     | -313870                 | 1092240                 | 1239360                 |
| JETTISONED SHROUD         |                  |         |             |     |                         |                         |                         |
| STAGING-HARDWARE          | -6351            | 1039.68 |             |     | -205355                 | -4899256                | -4757352                |
| PROPELLANT                |                  |         |             |     |                         |                         |                         |
| USABLE PROPELLANT         | 6351             | -1.00   | .03         |     | -100548                 | 9124260                 | 9129092                 |
| MECO                      |                  |         |             |     |                         |                         |                         |
|                           |                  |         |             |     | -1260                   | -486                    | -11904                  |

| SUMMARY                             | WEIGHT<br>POUNDS | X       | Y<br>INCHES | %     | IX<br>SLUG-FEET-SQUARED | IY<br>SLUG-FEET-SQUARED | IZ<br>SLUG-FEET-SQUARED | IXY<br>SLUG-FEET-SQUARED | IYZ<br>SLUG-FEET-SQUARED | IYZ<br>SLUG-FEET-SQUARED |
|-------------------------------------|------------------|---------|-------------|-------|-------------------------|-------------------------|-------------------------|--------------------------|--------------------------|--------------------------|
| INTERSTAGE ASY<br>STRUCTURE         | ( 11400)         | 2400.57 | 0.19        | 0.11  | 21511                   | 14773                   | 14095                   | 122                      | -102                     | -342                     |
| AVIONICS                            | 4201             | 2306.95 | 2.36        | -1.50 | 20990                   | 13594                   | 13730                   | 157                      | -105                     | -233                     |
| TPS                                 | 1877             | 2308.00 | 0.00        | 0.00  | 88                      | 81                      | 76                      | 0                        | 0                        | 0                        |
| TPS                                 | 65               | 2387.00 | 0.00        | 0.00  | 1                       | 1                       | 1                       | 0                        | 0                        | 0                        |
| ICCS                                | 5257             | 2425.48 | -1.47       | 1.50  | 419                     | 465                     | 453                     | 18                       | -42                      | -102                     |
| PAYLOAD SEPARATION                  |                  |         |             |       |                         |                         |                         |                          |                          |                          |
| FWD SKIRT ASY<br>STRUCTURE          | ( 2700)          | 2519.80 | 0.00        | 0.00  | 10593                   | 5776                    | 5776                    | 0                        | 0                        | 0                        |
| AVIONICS                            | 2413             | 2519.78 | 0.00        | 0.00  | 10507                   | 5770                    | 5770                    | 0                        | 0                        | 0                        |
| TPS                                 | 0                | 0.00    | 0.00        | 0.00  | 0                       | 0                       | 0                       | 0                        | 0                        | 0                        |
| TPS                                 | 217              | 2520.00 | 0.00        | 0.00  | 4                       | 4                       | 4                       | 0                        | 0                        | 0                        |
| PURGE MANIFOLD                      | 70               | 2520.00 | 0.00        | 0.00  | 1                       | 1                       | 1                       | 0                        | 0                        | 0                        |
| LOX VENT OUTLET                     |                  |         |             |       |                         |                         |                         |                          |                          |                          |
| AERO VENT                           |                  |         |             |       |                         |                         |                         |                          |                          |                          |
| LOX TANK ASY<br>STRUCTURE           | ( 19231)         | 4077.72 | 0.04        | 0.06  | 69733                   | 131636                  | 140222                  | -25                      | -42                      | 16                       |
| SLUSH RAFFLES                       | 15330            | 4080.67 | 0.00        | 0.00  | 69073                   | 107491                  | 116101                  | 0                        | 0                        | 0                        |
| PRESSURIZATION SYS                  | 171              | 4274.00 | 0.00        | 0.00  | 3                       | 3                       | 3                       | 0                        | 0                        | 0                        |
| VENT/RELIEF SYS                     | 146              | 3007.93 | 0.00        | 0.00  | 3                       | 3                       | 3                       | 0                        | 0                        | 0                        |
| FWD SYSTEM                          | 3377             | 4075.30 | 0.00        | 0.00  | 72                      | 17804                   | 17804                   | 0                        | 0                        | 0                        |
| AVIONICS                            | 48               | 3762.52 | 15.92       | 25.56 | 30                      | 167                     | 151                     | 26                       | 40                       | 12                       |
| TPS                                 | 160              | 3975.93 | 0.00        | 0.00  | 540                     | 1034                    | 1034                    | 0                        | 0                        | 0                        |
| INTERTANK ASY<br>STRUCTURE          | ( 4090)          | 3754.26 | -1.53       | -1.12 | 44602                   | 21991                   | 37790                   | -76                      | -66                      | 411                      |
| AVIONICS                            | 3572             | 3754.87 | -1.75       | -1.20 | 44591                   | 21977                   | 37784                   | -75                      | -65                      | 410                      |
| TPS                                 | 0                | 0.00    | 0.00        | 0.00  | 0                       | 0                       | 0                       | 0                        | 0                        | 0                        |
| TPS                                 | 466              | 3749.58 | 0.00        | 0.00  | 10                      | 10                      | 10                      | 0                        | 0                        | 0                        |
| PURGE MANIFOLD                      | 52               | 3754.50 | 0.00        | 0.00  | 1                       | 1                       | 1                       | 0                        | 0                        | 0                        |
| GROUND UNBITICAL<br>CROSS DEAH ASY  |                  |         |             |       |                         |                         |                         |                          |                          |                          |
| SRB ATTACH IHD                      |                  |         |             |       |                         |                         |                         |                          |                          |                          |
| AERO VENT                           |                  |         |             |       |                         |                         |                         |                          |                          |                          |
| LR2 TANK ASY<br>STRUCTURE           | ( 35579)         | 3215.75 | 0.07        | 0.00  | 223238                  | 1345494                 | 1347022                 | -178                     | 672                      | 2                        |
| AVIONICS                            | 33173            | 3201.88 | 0.00        | 0.00  | 209191                  | 1077494                 | 1079217                 | 0                        | 0                        | 0                        |
| TPS                                 | 68               | 2005.85 | 37.99       | 0.61  | 236                     | 3210                    | 3007                    | 6                        | 675                      | 2                        |
| TPS                                 | 1405             | 3100.65 | 0.00        | 0.00  | 13770                   | 67080                   | 67080                   | 0                        | 0                        | 0                        |
| FWD SYS                             | 535              | 4274.00 | 0.00        | 0.00  | 11                      | 11                      | 11                      | 0                        | 0                        | 0                        |
| PRESSURIZATION SYS                  | 182              | 4274.00 | 0.00        | 0.00  | 3                       | 3                       | 3                       | 0                        | 0                        | 0                        |
| VENT/RELIEF SYS                     | 137              | 2451.65 | 0.00        | 0.00  | 2                       | 2                       | 2                       | 0                        | 0                        | 0                        |
| AFT SKIRT ASY<br>STRUCTURE          | ( 7000)          | 4170.63 | 0.00        | 0.00  | 22657                   | 14375                   | 14297                   | 0                        | 0                        | 0                        |
| AVIONICS                            | 4255             | 4171.20 | 0.00        | 0.00  | 22401                   | 14193                   | 14193                   | 0                        | 0                        | 0                        |
| TPS                                 | 2745             | 4169.74 | 0.00        | 0.00  | 175                     | 170                     | 99                      | 0                        | 0                        | 0                        |
| TPS                                 | 0                | 1933.00 | 0.00        | 0.00  | 0                       | 0                       | 0                       | 0                        | 0                        | 0                        |
| GROUND UNBITICAL<br>PROPELLANT FEED |                  |         |             |       |                         |                         |                         |                          |                          |                          |

|                           | WEIGHT<br>POUNDS | X       | Y<br>INCHES | Z      | TX<br>SLUG-FEET-SQUARED  | TY<br>SLUG-FEET-SQUARED | TXZ<br>SLUG-FEET-SQUARED | TYZ   |
|---------------------------|------------------|---------|-------------|--------|--------------------------|-------------------------|--------------------------|-------|
| PROPULSION MODULE         |                  |         |             |        |                          |                         |                          |       |
| STRUCTURE                 | ( 51056)         | 4331.67 | 0.00        | 0.00   | 158477                   | 157762                  | 0                        | 0     |
| AVIONICS                  | 23944            | 4307.17 | 0.00        | 0.00   | 157106                   | 91279                   | 0                        | 0     |
| RACE HEAT SHIELD/TPS      | 0                | 0.00    | 0.00        | 0.00   | 0                        | 0                       | 0                        | 0     |
| PROP FEED SYS             | 10017            | 4477.00 | 0.00        | 0.00   | 933                      | 1403                    | 0                        | 0     |
| PRESSURIZATION SYS        | 8775             | 4274.00 | 0.00        | 0.00   | 189                      | 187                     | 0                        | 0     |
| PNEUMATIC SYS             | 251              | 4274.00 | 0.00        | 0.00   | 5                        | 5                       | 0                        | 0     |
| TOL                       | 779              | 4274.00 | 0.00        | 0.00   | 16                       | 17                      | 0                        | 0     |
| PURGE & VENT              | 6735             | 4274.00 | 0.00        | 0.00   | 189                      | 187                     | 0                        | 0     |
| SEPARATION                | 181              | 4274.00 | 0.00        | 0.00   | 3                        | 3                       | 0                        | 0     |
|                           | 368              | 4274.00 | 0.00        | 0.00   | 31                       | 31                      | 0                        | 0     |
| INTERFACE & SYS IHD       | ( 1800)          | 3544.66 | 21.35       | 94.25  | 4389                     | 120677                  | -1034                    | 609   |
| CABLE TRAY ASY            | 1446             | 3517.00 | 26.57       | 117.31 | 3407                     | 102593                  | -805                     | 1622  |
| PROPELLANT LINES          |                  |         |             |        |                          |                         |                          |       |
| PRESS SYS LINES           |                  |         |             |        |                          |                         |                          |       |
| PURGE & PURGE LINES       |                  |         |             |        |                          |                         |                          |       |
| SYSTEM CABLES             | 103              | 4274.00 | 0.00        | 0.00   | 2                        | 2                       | 0                        | 0     |
| HAZ GAS DETECTION         | 251              | 3406.00 | 0.00        | 0.00   | 91                       | 4172                    | 0                        | 0     |
| I/F IHD & CLOSURES        |                  |         |             |        |                          |                         |                          |       |
| AFT SRB STRUTS            |                  |         |             |        |                          |                         |                          |       |
| ENGINES                   | ( 54600)         | 4453.00 | 0.00        | 0.00   | 239597                   | 136304                  | 0                        | 0     |
| TOTAL CORE THURT          | 180250           | 3961.50 | 0.20        | 0.00   | 798397                   | 17976252                | -5509                    | 986   |
| SHROUD                    | 12500            | 1972.93 | 0.00        | 0.00   | 20142                    | 246464                  | 0                        | 0     |
| TOTAL VEHICLE THURT       | 200750           | 3835.19 | 0.19        | 0.00   | 818542                   | 28634004                | -4488                    | 986   |
| PROPELLANT-LO             | 1757703          | 3913.00 | 0.00        | 0.00   | 25863                    | 40150172                | 0                        | 0     |
| PAYLOAD (TARGET + MARGIN) | 72248            | 1904.00 | 0.00        | 0.00   | 6237                     | 6234                    | 0                        | 0     |
| LO                        | 2030701          | 3833.83 | 0.02        | 0.00   | 850672137244704137263344 |                         | -4469                    | 10527 |
| JETTISONED SHROUD         | 12500            | 1932.93 | 0.00        | 0.00   | 20142                    | 246464                  | 0                        | 0     |
| STAGING-HARDWARE          | 70700            | 4398.02 | 0.00        | 0.00   | 312358                   | 276427                  | 0                        | 0     |
| -PROPELLANT               | 7612             | 4426.67 | 0.00        | 0.00   | 25769                    | 22455                   | 0                        | 0     |
| USEABLE PROPELLANT        | 1725400          | 3907.91 | 0.00        | 0.00   | -15                      | 46019380                | 0                        | 0     |
| NECO                      | 214401           | 3141.70 | 0.10        | 0.70   | 492350                   | 50475972                | 1167                     | 15266 |

## REF MISS 2:10' STRETCH; EQUAL; LOX TK APT (1000 GEGRRR)

| EVENT        | TIME<br>(Seconds) | WEIGHT<br>(Pounds) | CENTER OF GRAVITY<br>(Station - Inches) |     |     | MOMENTS OF INERTIA<br>(Slug - Feet <sup>2</sup> ) |            |            | PRODUCTS OF INERTIA<br>(Slug - Feet <sup>2</sup> ) |        |        |
|--------------|-------------------|--------------------|-----------------------------------------|-----|-----|---------------------------------------------------|------------|------------|----------------------------------------------------|--------|--------|
|              |                   |                    | X                                       | Y   | Z   | IX                                                | IY         | IZ         | IXY                                                | IYZ    | IYZ    |
| LA           | 0.000             | 2000001.0          | 3011.9                                  | 0.0 | 0.1 | 85076.9                                           | 145566.04  | 145566.176 | -1780.                                             | 104.   | 10520. |
| NO. TILT     | 10.000            | 199686.0           | 3042.7                                  | 0.0 | 0.1 | 85076.9                                           | 141176.96  | 141277.152 | -1861.                                             | -215.  | 10520. |
|              | 20.000            | 1994670.5          | 3050.5                                  | 0.0 | 0.1 | 85076.9                                           | 137411.200 | 1374507.36 | -1924.                                             | -499.  | 10520. |
| END TILT     | 30.000            | 1757655.0          | 3057.6                                  | 0.0 | 0.1 | 85076.8                                           | 134001096. | 134020560. | -1982.                                             | -755.  | 10520. |
| THRTL. DESTR | 30.000            | 1684843.0          | 3062.2                                  | 0.0 | 0.1 | 85076.8                                           | 131447776. | 131447776. | -2023.                                             | -940.  | 10520. |
|              | 40.000            | 1670263.5          | 3063.6                                  | 0.0 | 0.1 | 85076.8                                           | 130531224. | 130950680. | -2031.                                             | -975.  | 10520. |
|              | 50.000            | 1597365.6          | 3068.1                                  | 0.0 | 0.1 | 85076.8                                           | 128520104. | 128547560. | -2067.                                             | -1136. | 10520. |
|              | 52.790            | 1577027.0          | 3069.2                                  | 0.0 | 0.1 | 85076.8                                           | 127902088. | 127921552. | -2076.                                             | -1178. | 10520. |
| MAX Q        | 60.000            | 1524406.3          | 3071.9                                  | 0.0 | 0.1 | 85076.8                                           | 126317816. | 126337280. | -2090.                                             | -1275. | 10520. |
|              | 70.000            | 1451614.0          | 3075.0                                  | 0.0 | 0.1 | 85076.8                                           | 124265048. | 124284512. | -2123.                                             | -1307. | 10520. |
|              | 80.000            | 1378741.9          | 3077.2                                  | 0.0 | 0.1 | 85076.7                                           | 122304744. | 122404208. | -2141.                                             | -1468. | 10520. |
|              | 90.000            | 1305869.6          | 3078.4                                  | 0.0 | 0.1 | 85076.7                                           | 120653008. | 120673272. | -2152.                                             | -1517. | 10520. |
|              | 100.000           | 1232997.4          | 3078.5                                  | 0.0 | 0.1 | 85076.7                                           | 119019200. | 119038752. | -2162.                                             | -1570. | 10520. |
|              | 110.000           | 1160125.3          | 3077.3                                  | 0.0 | 0.1 | 85076.7                                           | 117403672. | 117502600. | -2142.                                             | -1470. | 10527. |
|              | 120.000           | 1087253.0          | 3074.3                                  | 0.0 | 0.2 | 85076.6                                           | 116023424. | 116042664. | -2110.                                             | -1363. | 10527. |
|              | 130.000           | 1014300.8          | 3069.3                                  | 0.0 | 0.2 | 85076.6                                           | 114598032. | 114617272. | -2077.                                             | -1183. | 10527. |
|              | 140.000           | 941500.5           | 3061.9                                  | 0.0 | 0.2 | 85076.5                                           | 113199568. | 113210008. | -2017.                                             | -913.  | 10527. |
|              | 150.000           | 868636.3           | 3051.3                                  | 0.0 | 0.2 | 85076.5                                           | 111746568. | 111766032. | -1931.                                             | -530.  | 10527. |
| DESTR CO     | 156.390           | 822071.0           | 3042.6                                  | 0.0 | 0.2 | 85076.4                                           | 110775056. | 110795320. | -1860.                                             | -212.  | 10527. |
| STAG100      | 157.390           | 740306.0           | 3041.9                                  | 0.0 | 0.2 | 85076.4                                           | 110775056. | 110723576. | -1854.                                             | -107.  | 10527. |
| STAG100      | 160.000           | 712467.7           | 3040.2                                  | 0.1 | 0.2 | 85126.36                                          | 104565272. | 104602392. | -1374.                                             | 1938.  | 20062. |
|              | 170.000           | 702129.3           | 3039.0                                  | 0.1 | 0.2 | 85126.36                                          | 103504264. | 103541152. | -1269.                                             | 2050.  | 20062. |
|              | 180.000           | 671790.9           | 3038.0                                  | 0.1 | 0.2 | 85126.35                                          | 102606200. | 102643328. | -1173.                                             | 2057.  | 20062. |
|              | 190.000           | 641452.5           | 3044.7                                  | 0.1 | 0.3 | 85126.34                                          | 101644776. | 101681096. | -1065.                                             | 3340.  | 20061. |
|              | 200.000           | 611114.3           | 3029.6                                  | 0.1 | 0.3 | 85126.33                                          | 100598024. | 100635144. | -942.                                              | 3889.  | 20061. |
|              | 210.000           | 580775.8           | 3012.3                                  | 0.1 | 0.3 | 85126.33                                          | 99449656.  | 99400768.  | -803.                                              | 4514.  | 20061. |
|              | 220.000           | 550437.4           | 3002.7                                  | 0.1 | 0.3 | 85126.31                                          | 98102344.  | 98219464.  | -643.                                              | 5227.  | 20061. |
|              | 230.000           | 520099.0           | 3000.2                                  | 0.1 | 0.3 | 85126.32                                          | 96702416.  | 96819528.  | -461.                                              | 6041.  | 20061. |
| SHROUD JET   | 236.590           | 500106.0           | 3053.7                                  | 0.1 | 0.3 | 85126.32                                          | 95768592.  | 95805712.  | -126.                                              | 6643.  | 20061. |
| SHROUD JET   | 246.590           | 487606.0           | 3097.0                                  | 0.1 | 0.3 | 85126.32                                          | 94329160.  | 94366280.  | -684.                                              | 5042.  | 20061. |
|              | 250.000           | 469221.1           | 3061.7                                  | 0.1 | 0.4 | 85126.31                                          | 92889804.  | 92915992.  | -472.                                              | 5993.  | 20061. |
|              | 260.000           | 446922.1           | 3061.7                                  | 0.1 | 0.4 | 85126.31                                          | 91440336.  | 91477336.  | -391.                                              | 6352.  | 20061. |
|              | 270.000           | 416503.7           | 3029.3                                  | 0.1 | 0.4 | 85126.31                                          | 89980816.  | 89980816.  | -129.                                              | 7525.  | 20060. |
|              | 280.000           | 386245.1           | 3051.2                                  | 0.1 | 0.4 | 85126.31                                          | 88520816.  | 88520816.  | 181.                                               | 8909.  | 20060. |
|              | 290.000           | 355906.7           | 3045.7                                  | 0.1 | 0.5 | 85126.31                                          | 87060816.  | 87060816.  | 550.                                               | 10560. | 20060. |
|              | 300.000           | 325568.2           | 3090.7                                  | 0.1 | 0.5 | 85126.31                                          | 85600816.  | 85600816.  | 997.                                               | 12555. | 20059. |
|              | 310.000           | 295229.0           | 3042.3                                  | 0.1 | 0.6 | 85126.31                                          | 84140816.  | 84140816.  | 1544.                                              | 15080. | 20059. |
| THRTL. CORE  | 302.000           | 280089.0           | 3007.3                                  | 0.1 | 0.6 | 85126.31                                          | 82680816.  | 82680816.  | 1674.                                              | 15581. | 20059. |
|              | 310.000           | 272072.0           | 3007.3                                  | 0.1 | 0.6 | 85126.31                                          | 81220816.  | 81220816.  | 2051.                                              | 17267. | 20059. |
|              | 320.000           | 250811.4           | 3007.3                                  | 0.1 | 0.7 | 85126.31                                          | 79760816.  | 79760816.  | 2606.                                              | 19747. | 20059. |
|              | 330.000           | 229551.0           | 3010.5                                  | 0.2 | 0.7 | 85126.31                                          | 78300816.  | 78300816.  | 3272.                                              | 22222. | 20059. |
| HECO         | 337.090           | 214481.0           | 3141.7                                  | 0.2 | 0.8 | 85126.31                                          | 58004444.  | 58004444.  | 3030.                                              | 25210. | 20057. |

### **A.3 Updated Trajectory Data for Very Far Aft CG**

Table A4. Trajectory data for 83% aft cg.

| LOAD INDICATOR | FIRST STAGE |         | MAXIMUM |         | MINIMUM |        | SECOND STAGE |        | UNITS        |
|----------------|-------------|---------|---------|---------|---------|--------|--------------|--------|--------------|
|                | TIME        | VALUE   | TIME    | VALUE   | TIME    | VALUE  | TIME         | VALUE  |              |
| Q ALPHA        | 9.00        | -27.25  | 30.00   | 1328.30 | 157.39  | -29.86 | 337.09       | 0.01   | LB-DEG/FT**2 |
| Q BETA         | 15.00       | -113.81 | 2.00    | 0.52    | 157.39  | -2.87  | 297.59       | 0.00   | LB-DEG/FT**2 |
| Q              | 0.00        | 0.00    | 52.79   | 722.57  | 310.09  | 0.00   | 157.39       | 5.70   | LB/FT**2     |
| LONG ACC       | 157.39      | 1.27    | 156.39  | 3.80    | 157.39  | 1.76   | 302.09       | 4.50   | G'S          |
| LAT ACC        | 3.00        | 0.00    | 16.00   | 0.01    | 299.59  | 0.00   | 157.39       | 0.00   | G'S          |
| NORM ACC       | 30.00       | -0.14   | 10.00   | 0.00    | 302.09  | 0.00   | 157.39       | 0.01   | G'S          |
| TOTAL ACC      | 157.39      | 1.27    | 156.39  | 3.80    | 157.39  | 1.76   | 302.09       | 4.50   | G'S          |
| STAG HEAT      | 0.00        | 0.00    | 157.39  | 261.94  | 157.39  | 261.94 | 337.09       | 386.92 | BTU/FT**2    |
| HEAT RATE      | 0.00        | 0.00    | 146.00  | 4.21    | 256.59  | 0.22   | 157.39       | 3.92   | BTU/FT**2/S  |
| PDOME          | 157.39      | 29.98   | 17.00   | 62.40   | 337.09  | 23.22  | 157.39       | 33.01  | PSI          |
| YRING          | 157.39      | 24.21   | 17.00   | 54.50   | 337.09  | 3.99   | 157.39       | 25.07  | PSI          |

STAGING CONDITIONS

|          |             |          |              |          |              |
|----------|-------------|----------|--------------|----------|--------------|
| Time     | 157.393 sec | Wgt-pre  | 818697.9 lbs | Wgt-post | 740385.9 lb  |
| Altitude | 231295.0 ft | Vel Iner | 9760.88 fps  | Gamma I  | 14.37150 deg |
| Dyn Pres | 5.701 psf   | Vel Rel  | 8455.13 fps  | Gamma R  | 16.65090 deg |
| Ax Acc   | 1.272 gs    | Mach No. | 8.76         |          |              |



|                                          | NOMINAL  | ENGINE OUT |          |
|------------------------------------------|----------|------------|----------|
| PAYLOAD                                  |          |            | 72248.   |
| OMS KIT                                  | 0.       | 0.         |          |
| UPPER STAGE INERT STRUCTURE              | 0.       | 0.         |          |
| UPPER STAGE - FLIGHT PERFORMANCE RESERVE | 0.       | 0.         |          |
| - DELTA VELOCITY PROPELLANT              | 0.       | 0.         | (0. F/S) |
| - EXCESS PROPELLANT (PROP LD)            | 0.       | 0.         | (0.)     |
| WEIGHT AT ORBIT INSERTION                |          |            | 72248.   |
| UPPER STAGE - PROPELLANT BURNED          | 0.       | 0.         |          |
| WEIGHT AFTER CORE SEPARATION             |          |            | 72248.   |
| PERFORMANCE MARGIN                       | 0.       | 0.         |          |
| 2ND STAGE - CORE DRY WEIGHT (MINUS STME) | 99350.   | 99350.     |          |
| - NONPROPULSIVE CONSUMABLES              | 0.       | 0.         |          |
| STME EMPTY                               | 18200.   | 18200.     |          |
| 2ND STAGE AT CORE SEPARATION             |          |            | 189798.  |
| TRAPPED MPS PROPELLANT - 2ND STAGE       | 0.       | 0.         |          |
| - STME                                   | 0.       | 0.         |          |
| FLIGHT PERFORMANCE RESERVE - ET          | 5571.    | 5571.      |          |
| CORE RESIDUALS                           | 12761.   | 12761.     | MARGIN   |
| EXCESS MPS PROPELLANT (MPS PROP LOADED)  | 6351.0   | 1898.      | (0.)     |
| EXPULSED MPS PROPELLANT                  | 0.       | 0.         |          |
| INJECTED WEIGHT AT MECO                  |          |            | 214481.  |
| SHROUD JETTISON WEIGHT                   | 12500.   | 12500.     |          |
| PROPELLANT BURNED-SEPARATION TO MECO     | 513405.  | 517858.    |          |
| EXPULSED MPS PROPELLANT                  | 0.       | 0.         |          |
| WEIGHT AT SEPARATION                     |          |            | 740386.  |
| 1ST STAGE JETTISON WEIGHT                | 78312.   |            |          |
| PROPELLANT BURNED-LIFTOFF TO SEPARATION  | 1212003. |            |          |
| RESIDUAL MPS PROPELLANT                  | 0.       |            |          |
| LIFTOFF (TW=1.0)                         |          |            | 2030701. |
| MPS PROPELLANT                           | 0.       |            |          |
| IGNITION COMMAND (T-0)                   |          |            | 2030701. |
| MPS THRUST BUILDUP                       | 30954.   |            |          |
| OVERFILL, DRAINBACK, AND BOILOFF         | 0.       |            |          |
| NONPROPULSIVE CONSUMABLES                | 0.       |            |          |
| PRELAUNCH WEIGHT (T-5 MIN)               |          |            | 2061655. |

|                              |                 |                   |                             |                      |                |
|------------------------------|-----------------|-------------------|-----------------------------|----------------------|----------------|
| case =NOM                    |                 |                   |                             |                      |                |
| TIME                         | 337.086671 SEC  | RANGE             | 499.05 km                   | RANGE ANGLE          | 9.540426 DEG   |
| INERTIAL<br>VELOCITY         | 7853.5207 M/SEC | RADIUS            | 6526317.96 M                | FLIGHT PATH<br>ANGLE | -0.000344 DEG  |
| INCL                         | 28.465702 DEG   | DES.<br>NODE      | 87.742551 DEG               | FLIGHT AZIMUTH       | 90.000000 DEG  |
| GDLAT                        | 28.000408 DEG   | GCLAT             | 27.841175 DEG               | LONG                 | -71.210835 DEG |
| INERTIAL<br>AZIMUTH          | 96.182579 DEG   | C1                | 0.51254573E+11 M**2/SEC     |                      |                |
| WEIGHT                       | 214480.83 LBS   | C3                | -0.60474153E+08 M**2/SEC**2 |                      |                |
| KEPLERIAN ORBITAL PARAMETERS |                 |                   |                             |                      |                |
| ECNTRICY                     | 0.009854        | APOGEE<br>RADIUS  | 6656213.14 M                | PERIGEE RADIUS       | 6526317.95 M   |
| APOGEE HEIGHT                | 150.136684 NM   | PERIGEE<br>HEIGHT | 79.998894 NM                |                      |                |

|            | TIME  | MACH  | ALPHA  | BETA   | Q      | QALPH | QBETA | EMACH | INBD | OUTBD |
|------------|-------|-------|--------|--------|--------|-------|-------|-------|------|-------|
| LIFT-OFF   | 0.00  | 0.001 | 0.000  | 0.000  | 0.00   | 0.    | 0.    | 0.00  | 0.00 | 0.00  |
|            | 1.00  | 0.018 | -3.237 | 0.629  | 0.48   | -2.   | 0.    | 0.02  | 0.00 | 0.00  |
|            | 2.00  | 0.036 | -1.721 | 0.268  | 1.92   | -3.   | 1.    | 0.04  | 0.00 | 0.00  |
|            | 3.00  | 0.054 | -1.261 | 0.117  | 4.37   | -6.   | 1.    | 0.05  | 0.00 | 0.00  |
|            | 4.00  | 0.073 | -1.061 | 0.004  | 7.85   | -8.   | 0.    | 0.07  | 0.00 | 0.00  |
|            | 5.00  | 0.092 | -0.958 | -0.105 | 12.38  | -12.  | -1.   | 0.09  | 0.00 | 0.00  |
|            | 6.00  | 0.111 | -0.892 | -0.225 | 17.99  | -16.  | -4.   | 0.11  | 0.00 | 0.00  |
|            | 7.00  | 0.130 | -0.833 | -0.361 | 24.70  | -21.  | -9.   | 0.13  | 0.00 | 0.00  |
|            | 8.00  | 0.150 | -0.760 | -0.510 | 32.53  | -25.  | -17.  | 0.15  | 0.00 | 0.00  |
|            | 9.00  | 0.170 | -0.657 | -0.665 | 41.50  | -27.  | -28.  | 0.17  | 0.00 | 0.00  |
| BEGIN TILT | 10.00 | 0.190 | -0.514 | -0.816 | 51.61  | -27.  | -42.  | 0.19  | 0.00 | 0.00  |
|            | 10.00 | 0.190 | -0.514 | -0.816 | 51.61  | -27.  | -42.  | 0.19  | 0.00 | 0.00  |
|            | 11.00 | 0.211 | 0.539  | -0.943 | 62.87  | 34.   | -59.  | 0.21  | 0.00 | 0.00  |
|            | 12.00 | 0.232 | 1.427  | -1.033 | 75.28  | 107.  | -78.  | 0.23  | 0.00 | 0.00  |
|            | 13.00 | 0.253 | 2.184  | -1.070 | 88.82  | 194.  | -95.  | 0.25  | 0.00 | 0.00  |
|            | 14.00 | 0.275 | 2.825  | -1.045 | 103.46 | 292.  | -108. | 0.27  | 0.00 | 0.00  |
|            | 15.00 | 0.296 | 3.355  | -0.955 | 119.17 | 400.  | -114. | 0.30  | 0.00 | 0.00  |
|            | 16.00 | 0.318 | 3.770  | -0.809 | 135.93 | 512.  | -96.  | 0.32  | 0.00 | 0.00  |
|            | 17.00 | 0.341 | 4.069  | -0.624 | 153.70 | 625.  | -77.  | 0.34  | 0.00 | 0.00  |
|            | 18.00 | 0.364 | 4.245  | -0.447 | 172.47 | 732.  | -61.  | 0.36  | 0.00 | 0.00  |
| END TILT   | 19.00 | 0.387 | 4.317  | -0.318 | 192.20 | 830.  | -53.  | 0.39  | 0.00 | 0.00  |
|            | 20.00 | 0.410 | 4.310  | -0.247 | 212.88 | 918.  | -54.  | 0.41  | 0.00 | 0.00  |
|            | 21.00 | 0.434 | 4.246  | -0.232 | 234.46 | 995.  | -65.  | 0.44  | 0.00 | 0.00  |
|            | 22.00 | 0.459 | 4.139  | -0.252 | 256.91 | 1063. | -76.  | 0.46  | 0.00 | 0.00  |
|            | 23.00 | 0.484 | 4.007  | -0.273 | 280.17 | 1123. | -74.  | 0.49  | 0.00 | 0.00  |
|            | 24.00 | 0.509 | 3.868  | -0.244 | 304.19 | 1177. | -64.  | 0.51  | 0.00 | 0.00  |
|            | 25.00 | 0.535 | 3.720  | -0.196 | 328.91 | 1223. | -58.  | 0.54  | 0.00 | 0.00  |
|            | 26.00 | 0.561 | 3.556  | -0.167 | 354.29 | 1260. | -60.  | 0.57  | 0.00 | 0.00  |
|            | 27.00 | 0.588 | 3.384  | -0.152 | 380.29 | 1287. | -60.  | 0.59  | 0.00 | 0.00  |
|            | 28.00 | 0.615 | 3.209  | -0.148 | 406.83 | 1305. | -60.  | 0.62  | 0.00 | 0.00  |
|            | 29.00 | 0.643 | 3.038  | -0.140 | 433.90 | 1318. | -60.  | 0.65  | 0.00 | 0.00  |
|            | 30.00 | 0.672 | 2.878  | -0.130 | 461.50 | 1328. | -60.  | 0.68  | 0.00 | 0.00  |
|            | 30.00 | 0.672 | 2.878  | -0.130 | 461.50 | 1328. | -60.  | 0.68  | 0.00 | 0.00  |
|            | 31.00 | 0.702 | 0.000  | 0.000  | 489.89 | 0.    | 0.    | 0.71  | 0.00 | 0.00  |
|            | 32.00 | 0.732 | 0.000  | 0.000  | 518.40 | 0.    | 0.    | 0.74  | 0.00 | 0.00  |
|            | 33.00 | 0.763 | 0.000  | 0.000  | 546.97 | 0.    | 0.    | 0.77  | 0.00 | 0.00  |
|            | 34.00 | 0.795 | 0.000  | 0.000  | 575.51 | 0.    | 0.    | 0.81  | 0.00 | 0.00  |
|            | 35.00 | 0.827 | 0.000  | 0.000  | 603.86 | 0.    | 0.    | 0.84  | 0.00 | 0.00  |
|            | 36.00 | 0.860 | 0.000  | 0.000  | 631.83 | 0.    | 0.    | 0.88  | 0.00 | 0.00  |
|            | 37.00 | 0.894 | 0.000  | 0.000  | 659.24 | 0.    | 0.    | 0.91  | 0.00 | 0.00  |
|            | 38.00 | 0.928 | 0.000  | 0.000  | 685.88 | 0.    | 0.    | 0.95  | 0.00 | 0.00  |
|            | 38.00 | 0.928 | 0.000  | 0.000  | 685.88 | 0.    | 0.    | 0.95  | 0.00 | 0.00  |
|            | 39.00 | 0.949 | 0.000  | 0.000  | 691.46 | 0.    | 0.    | 0.97  | 0.00 | 0.00  |

MAX DYN PRES

| TIME  | MACH  | ALPHA | BETA  | Q      | QALPH | QBETA | EMACH | INBD | OUTBD |
|-------|-------|-------|-------|--------|-------|-------|-------|------|-------|
| 40.00 | 0.970 | 0.000 | 0.000 | 696.62 | 0.    | 0.    | 0.99  | 0.00 | 0.00  |
| 41.00 | 0.992 | 0.000 | 0.000 | 701.28 | 0.    | 0.    | 1.01  | 0.00 | 0.00  |
| 42.00 | 1.015 | 0.000 | 0.000 | 705.46 | 0.    | 0.    | 1.04  | 0.00 | 0.00  |
| 43.00 | 1.038 | 0.000 | 0.000 | 709.14 | 0.    | 0.    | 1.06  | 0.00 | 0.00  |
| 44.00 | 1.062 | 0.000 | 0.000 | 712.23 | 0.    | 0.    | 1.09  | 0.00 | 0.00  |
| 45.00 | 1.086 | 0.000 | 0.000 | 714.76 | 0.    | 0.    | 1.12  | 0.00 | 0.00  |
| 46.00 | 1.111 | 0.000 | 0.000 | 716.78 | 0.    | 0.    | 1.14  | 0.00 | 0.00  |
| 47.00 | 1.137 | 0.000 | 0.000 | 718.63 | 0.    | 0.    | 1.17  | 0.00 | 0.00  |
| 48.00 | 1.164 | 0.000 | 0.000 | 720.17 | 0.    | 0.    | 1.20  | 0.00 | 0.00  |
| 49.00 | 1.192 | 0.000 | 0.000 | 720.82 | 0.    | 0.    | 1.23  | 0.00 | 0.00  |
| 50.00 | 1.221 | 0.000 | 0.000 | 721.28 | 0.    | 0.    | 1.26  | 0.00 | 0.00  |
| 51.00 | 1.251 | 0.000 | 0.000 | 721.80 | 0.    | 0.    | 1.30  | 0.00 | 0.00  |
| 52.00 | 1.283 | 0.000 | 0.000 | 722.35 | 0.    | 0.    | 1.33  | 0.00 | 0.00  |
| 52.79 | 1.309 | 0.000 | 0.000 | 722.57 | 0.    | 0.    | 1.36  | 0.00 | 0.00  |
| 53.00 | 1.316 | 0.000 | 0.000 | 722.55 | 0.    | 0.    | 1.37  | 0.00 | 0.00  |
| 54.00 | 1.350 | 0.000 | 0.000 | 722.07 | 0.    | 0.    | 1.40  | 0.00 | 0.00  |
| 55.00 | 1.386 | 0.000 | 0.000 | 721.14 | 0.    | 0.    | 1.44  | 0.00 | 0.00  |
| 56.00 | 1.423 | 0.000 | 0.000 | 719.86 | 0.    | 0.    | 1.48  | 0.00 | 0.00  |
| 57.00 | 1.462 | 0.000 | 0.000 | 718.23 | 0.    | 0.    | 1.52  | 0.00 | 0.00  |
| 58.00 | 1.503 | 0.000 | 0.000 | 716.05 | 0.    | 0.    | 1.56  | 0.00 | 0.00  |
| 59.00 | 1.545 | 0.000 | 0.000 | 713.18 | 0.    | 0.    | 1.60  | 0.00 | 0.00  |
| 60.00 | 1.589 | 0.000 | 0.000 | 709.66 | 0.    | 0.    | 1.65  | 0.00 | 0.00  |
| 61.00 | 1.634 | 0.000 | 0.000 | 705.04 | 0.    | 0.    | 1.69  | 0.00 | 0.00  |
| 62.00 | 1.680 | 0.000 | 0.000 | 698.97 | 0.    | 0.    | 1.73  | 0.00 | 0.00  |
| 63.00 | 1.726 | 0.000 | 0.000 | 691.52 | 0.    | 0.    | 1.78  | 0.00 | 0.00  |
| 64.00 | 1.773 | 0.000 | 0.000 | 682.36 | 0.    | 0.    | 1.82  | 0.00 | 0.00  |
| 65.00 | 1.819 | 0.000 | 0.000 | 671.37 | 0.    | 0.    | 1.87  | 0.00 | 0.00  |
| 66.00 | 1.866 | 0.000 | 0.000 | 658.92 | 0.    | 0.    | 1.91  | 0.00 | 0.00  |
| 67.00 | 1.912 | 0.000 | 0.000 | 645.23 | 0.    | 0.    | 1.96  | 0.00 | 0.00  |
| 68.00 | 1.959 | 0.000 | 0.000 | 630.19 | 0.    | 0.    | 2.00  | 0.00 | 0.00  |
| 69.00 | 2.006 | 0.000 | 0.000 | 614.09 | 0.    | 0.    | 2.05  | 0.00 | 0.00  |
| 70.00 | 2.053 | 0.000 | 0.000 | 597.55 | 0.    | 0.    | 2.09  | 0.00 | 0.00  |
| 71.00 | 2.099 | 0.000 | 0.000 | 580.00 | 0.    | 0.    | 2.14  | 0.00 | 0.00  |
| 72.00 | 2.142 | 0.000 | 0.000 | 560.21 | 0.    | 0.    | 2.17  | 0.00 | 0.00  |
| 73.00 | 2.186 | 0.000 | 0.000 | 540.26 | 0.    | 0.    | 2.21  | 0.00 | 0.00  |
| 74.00 | 2.232 | 0.000 | 0.000 | 521.87 | 0.    | 0.    | 2.25  | 0.00 | 0.00  |
| 75.00 | 2.278 | 0.000 | 0.000 | 503.93 | 0.    | 0.    | 2.30  | 0.00 | 0.00  |
| 76.00 | 2.326 | 0.000 | 0.000 | 485.94 | 0.    | 0.    | 2.34  | 0.00 | 0.00  |
| 77.00 | 2.373 | 0.000 | 0.000 | 468.55 | 0.    | 0.    | 2.38  | 0.00 | 0.00  |
| 78.00 | 2.420 | 0.000 | 0.000 | 451.85 | 0.    | 0.    | 2.42  | 0.00 | 0.00  |
| 79.00 | 2.464 | 0.000 | 0.000 | 433.75 | 0.    | 0.    | 2.46  | 0.00 | 0.00  |
| 80.00 | 2.507 | 0.000 | 0.000 | 415.23 | 0.    | 0.    | 2.51  | 0.00 | 0.00  |
| 81.00 | 2.552 | 0.000 | 0.000 | 397.21 | 0.    | 0.    | 2.55  | 0.00 | 0.00  |
| 82.00 | 2.597 | 0.000 | 0.000 | 379.72 | 0.    | 0.    | 2.60  | 0.00 | 0.00  |
| 83.00 | 2.642 | 0.000 | 0.000 | 363.26 | 0.    | 0.    | 2.64  | 0.00 | 0.00  |

| TIME   | MACH  | ALPHA | BETA  | Q      | QALPH | QBETA | EMACH | INBD | OUTBD |
|--------|-------|-------|-------|--------|-------|-------|-------|------|-------|
| 84.00  | 2.687 | 0.000 | 0.000 | 347.45 | 0.    | 0.    | 2.69  | 0.00 | 0.00  |
| 85.00  | 2.732 | 0.000 | 0.000 | 331.66 | 0.    | 0.    | 2.73  | 0.00 | 0.00  |
| 86.00  | 2.778 | 0.000 | 0.000 | 316.20 | 0.    | 0.    | 2.78  | 0.00 | 0.00  |
| 87.00  | 2.826 | 0.000 | 0.000 | 301.89 | 0.    | 0.    | 2.83  | 0.00 | 0.00  |
| 88.00  | 2.874 | 0.000 | 0.000 | 288.31 | 0.    | 0.    | 2.88  | 0.00 | 0.00  |
| 89.00  | 2.923 | 0.000 | 0.000 | 274.91 | 0.    | 0.    | 2.93  | 0.00 | 0.00  |
| 90.00  | 2.973 | 0.000 | 0.000 | 262.02 | 0.    | 0.    | 2.98  | 0.00 | 0.00  |
| 91.00  | 3.024 | 0.000 | 0.000 | 250.06 | 0.    | 0.    | 3.04  | 0.00 | 0.00  |
| 92.00  | 3.075 | 0.000 | 0.000 | 238.48 | 0.    | 0.    | 3.09  | 0.00 | 0.00  |
| 93.00  | 3.126 | 0.000 | 0.000 | 226.88 | 0.    | 0.    | 3.14  | 0.00 | 0.00  |
| 94.00  | 3.177 | 0.000 | 0.000 | 215.76 | 0.    | 0.    | 3.19  | 0.00 | 0.00  |
| 95.00  | 3.225 | 0.000 | 0.000 | 204.88 | 0.    | 0.    | 3.24  | 0.00 | 0.00  |
| 96.00  | 3.272 | 0.000 | 0.000 | 194.13 | 0.    | 0.    | 3.29  | 0.00 | 0.00  |
| 97.00  | 3.323 | 0.000 | 0.000 | 184.15 | 0.    | 0.    | 3.35  | 0.00 | 0.00  |
| 98.00  | 3.377 | 0.000 | 0.000 | 175.15 | 0.    | 0.    | 3.40  | 0.00 | 0.00  |
| 99.00  | 3.432 | 0.000 | 0.000 | 166.60 | 0.    | 0.    | 3.45  | 0.00 | 0.00  |
| 100.00 | 3.485 | 0.000 | 0.000 | 157.96 | 0.    | 0.    | 3.50  | 0.00 | 0.00  |
| 101.00 | 3.535 | 0.000 | 0.000 | 149.56 | 0.    | 0.    | 3.56  | 0.00 | 0.00  |
| 102.00 | 3.585 | 0.000 | 0.000 | 141.59 | 0.    | 0.    | 3.61  | 0.00 | 0.00  |
| 103.00 | 3.634 | 0.000 | 0.000 | 133.92 | 0.    | 0.    | 3.66  | 0.00 | 0.00  |
| 104.00 | 3.685 | 0.000 | 0.000 | 126.58 | 0.    | 0.    | 3.72  | 0.00 | 0.00  |
| 105.00 | 3.736 | 0.000 | 0.000 | 119.79 | 0.    | 0.    | 3.77  | 0.00 | 0.00  |
| 106.00 | 3.789 | 0.000 | 0.000 | 113.39 | 0.    | 0.    | 3.83  | 0.00 | 0.00  |
| 107.00 | 3.843 | 0.000 | 0.000 | 107.21 | 0.    | 0.    | 3.88  | 0.00 | 0.00  |
| 108.00 | 3.898 | 0.000 | 0.000 | 101.53 | 0.    | 0.    | 3.94  | 0.00 | 0.00  |
| 109.00 | 3.954 | 0.000 | 0.000 | 96.21  | 0.    | 0.    | 3.99  | 0.00 | 0.00  |
| 110.00 | 4.010 | 0.000 | 0.000 | 90.99  | 0.    | 0.    | 4.05  | 0.00 | 0.00  |
| 111.00 | 4.065 | 0.000 | 0.000 | 85.97  | 0.    | 0.    | 4.10  | 0.00 | 0.00  |
| 112.00 | 4.121 | 0.000 | 0.000 | 81.29  | 0.    | 0.    | 4.16  | 0.00 | 0.00  |
| 113.00 | 4.178 | 0.000 | 0.000 | 77.05  | 0.    | 0.    | 4.21  | 0.00 | 0.00  |
| 114.00 | 4.236 | 0.000 | 0.000 | 73.13  | 0.    | 0.    | 4.27  | 0.00 | 0.00  |
| 115.00 | 4.294 | 0.000 | 0.000 | 69.35  | 0.    | 0.    | 4.33  | 0.00 | 0.00  |
| 116.00 | 4.352 | 0.000 | 0.000 | 65.70  | 0.    | 0.    | 4.39  | 0.00 | 0.00  |
| 117.00 | 4.411 | 0.000 | 0.000 | 62.15  | 0.    | 0.    | 4.44  | 0.00 | 0.00  |
| 118.00 | 4.470 | 0.000 | 0.000 | 58.70  | 0.    | 0.    | 4.50  | 0.00 | 0.00  |
| 119.00 | 4.530 | 0.000 | 0.000 | 55.34  | 0.    | 0.    | 4.56  | 0.00 | 0.00  |
| 120.00 | 4.593 | 0.000 | 0.000 | 52.14  | 0.    | 0.    | 4.63  | 0.00 | 0.00  |
| 121.00 | 4.656 | 0.000 | 0.000 | 49.29  | 0.    | 0.    | 4.69  | 0.00 | 0.00  |
| 122.00 | 4.720 | 0.000 | 0.000 | 46.72  | 0.    | 0.    | 4.75  | 0.00 | 0.00  |
| 123.00 | 4.784 | 0.000 | 0.000 | 44.36  | 0.    | 0.    | 4.82  | 0.00 | 0.00  |
| 124.00 | 4.851 | 0.000 | 0.000 | 42.12  | 0.    | 0.    | 4.88  | 0.00 | 0.00  |
| 125.00 | 4.920 | 0.000 | 0.000 | 39.90  | 0.    | 0.    | 4.95  | 0.00 | 0.00  |
| 126.00 | 4.994 | 0.000 | 0.000 | 37.68  | 0.    | 0.    | 5.03  | 0.00 | 0.00  |
| 127.00 | 5.072 | 0.000 | 0.000 | 35.69  | 0.    | 0.    | 5.11  | 0.00 | 0.00  |
| 128.00 | 5.154 | 0.000 | 0.000 | 33.92  | 0.    | 0.    | 5.19  | 0.00 | 0.00  |

| TIME   | MACH   | ALPHA   | BETA   | Q     | QALPH | QBETA | EMACH | INBD | OUTBD |
|--------|--------|---------|--------|-------|-------|-------|-------|------|-------|
| 129.00 | 5.239  | 0.000   | 0.000  | 32.31 | 0.    | 0.    | 5.28  | 0.00 | 0.00  |
| 130.00 | 5.328  | 0.000   | 0.000  | 30.81 | 0.    | 0.    | 5.37  | 0.00 | 0.00  |
| 131.00 | 5.419  | 0.000   | 0.000  | 29.32 | 0.    | 0.    | 5.46  | 0.00 | 0.00  |
| 132.00 | 5.512  | 0.000   | 0.000  | 27.81 | 0.    | 0.    | 5.55  | 0.00 | 0.00  |
| 133.00 | 5.609  | 0.000   | 0.000  | 26.40 | 0.    | 0.    | 5.65  | 0.00 | 0.00  |
| 134.00 | 5.710  | 0.000   | 0.000  | 25.08 | 0.    | 0.    | 5.75  | 0.00 | 0.00  |
| 135.00 | 5.813  | 0.000   | 0.000  | 23.84 | 0.    | 0.    | 5.85  | 0.00 | 0.00  |
| 136.00 | 5.921  | 0.000   | 0.000  | 22.67 | 0.    | 0.    | 5.96  | 0.00 | 0.00  |
| 137.00 | 6.030  | 0.000   | 0.000  | 21.52 | 0.    | 0.    | 6.07  | 0.00 | 0.00  |
| 138.00 | 6.142  | 0.000   | 0.000  | 20.38 | 0.    | 0.    | 6.18  | 0.00 | 0.00  |
| 139.00 | 6.256  | 0.000   | 0.000  | 19.22 | 0.    | 0.    | 6.29  | 0.00 | 0.00  |
| 140.00 | 6.372  | 0.000   | 0.000  | 18.13 | 0.    | 0.    | 6.41  | 0.00 | 0.00  |
| 141.00 | 6.489  | 0.000   | 0.000  | 17.12 | 0.    | 0.    | 6.52  | 0.00 | 0.00  |
| 142.00 | 6.609  | 0.000   | 0.000  | 16.18 | 0.    | 0.    | 6.64  | 0.00 | 0.00  |
| 143.00 | 6.736  | 0.000   | 0.000  | 15.30 | 0.    | 0.    | 6.77  | 0.00 | 0.00  |
| 144.00 | 6.871  | 0.000   | 0.000  | 14.49 | 0.    | 0.    | 6.89  | 0.00 | 0.00  |
| 145.00 | 7.008  | 0.000   | 0.000  | 13.67 | 0.    | 0.    | 7.02  | 0.00 | 0.00  |
| 146.00 | 7.141  | 0.000   | 0.000  | 12.82 | 0.    | 0.    | 7.15  | 0.00 | 0.00  |
| 147.00 | 7.274  | 0.000   | 0.000  | 12.00 | 0.    | 0.    | 7.29  | 0.00 | 0.00  |
| 148.00 | 7.408  | 0.000   | 0.000  | 11.23 | 0.    | 0.    | 7.42  | 0.00 | 0.00  |
| 149.00 | 7.539  | 0.000   | 0.000  | 10.52 | 0.    | 0.    | 7.56  | 0.00 | 0.00  |
| 150.00 | 7.666  | 0.000   | 0.000  | 9.82  | 0.    | 0.    | 7.70  | 0.00 | 0.00  |
| 151.00 | 7.798  | 0.000   | 0.000  | 9.17  | 0.    | 0.    | 7.85  | 0.00 | 0.00  |
| 152.00 | 7.941  | 0.000   | 0.000  | 8.55  | 0.    | 0.    | 7.99  | 0.00 | 0.00  |
| 153.00 | 8.092  | 0.000   | 0.000  | 7.94  | 0.    | 0.    | 8.14  | 0.00 | 0.00  |
| 154.00 | 8.243  | 0.000   | 0.000  | 7.36  | 0.    | 0.    | 8.29  | 0.00 | 0.00  |
| 155.00 | 8.386  | 0.000   | 0.000  | 6.81  | 0.    | 0.    | 8.45  | 0.00 | 0.00  |
| 156.00 | 8.528  | 0.000   | 0.000  | 6.31  | 0.    | 0.    | 8.61  | 0.00 | 0.00  |
| 156.39 | 8.587  | 0.000   | 0.000  | 6.12  | 0.    | 0.    | 8.67  | 0.00 | 0.00  |
| 156.99 | 8.587  | 0.000   | 0.000  | 6.12  | 0.    | 0.    | 8.67  | 0.00 | 0.00  |
| 157.00 | 8.731  | 0.000   | 0.000  | 5.91  | 0.    | 0.    | 8.73  | 0.00 | 0.00  |
| 157.39 | 8.762  | 0.000   | 0.000  | 5.70  | 0.    | 0.    | 8.76  | 0.00 | 0.00  |
| 157.39 | 8.762  | -12.752 | -0.470 | 5.70  | -73.  | -3.   | 8.76  | 0.00 | 0.00  |
| 157.39 | 8.762  | -12.752 | -0.470 | 5.70  | -73.  | -3.   | 8.76  | 0.00 | 0.00  |
| 167.00 | 9.380  | -13.828 | -0.501 | 2.16  | -30.  | -1.   | 9.38  | 0.00 | 0.00  |
| 177.00 | 10.000 | -14.511 | -0.527 | 0.72  | -10.  | 0.    | 10.00 | 0.00 | 0.00  |
| 187.00 | 10.199 | -14.960 | -0.547 | 0.23  | -3.   | 0.    | 10.20 | 0.00 | 0.00  |
| 197.00 | 9.953  | -15.288 | -0.563 | 0.08  | -1.   | 0.    | 9.95  | 0.00 | 0.00  |
| 207.00 | 9.694  | -15.535 | -0.577 | 0.03  | 0.    | 0.    | 9.69  | 0.00 | 0.00  |
| 217.00 | 9.432  | -15.712 | -0.589 | 0.01  | 0.    | 0.    | 9.43  | 0.00 | 0.00  |
| 227.00 | 8.994  | -15.824 | -0.601 | 0.01  | 0.    | 0.    | 8.99  | 0.00 | 0.00  |
| 237.00 | 8.615  | -15.874 | -0.611 | 0.00  | 0.    | 0.    | 8.62  | 0.00 | 0.00  |
| 247.00 | 8.143  | -15.865 | -0.619 | 0.00  | 0.    | 0.    | 8.14  | 0.00 | 0.00  |
| 247.39 | 8.116  | -15.866 | -0.620 | 0.00  | 0.    | 0.    | 8.12  | 0.00 | 0.00  |
| 247.39 | 8.116  | -15.862 | -0.619 | 0.00  | 0.    | 0.    | 8.12  | 0.00 | 0.00  |

TAILOFF

SEPARATION  
BEGIN MINH

SHROUD SEP

| TIME   | MACH   | ALPHA   | BETA   | Q    | QALPH | QBETA | EMACH | INBD | OUTBD |
|--------|--------|---------|--------|------|-------|-------|-------|------|-------|
| 257.00 | 7.615  | -15.785 | -0.627 | 0.00 | 0.    | 0.    | 7.62  | 0.00 | 0.00  |
| 267.00 | 7.296  | -15.650 | -0.633 | 0.00 | 0.    | 0.    | 7.30  | 0.00 | 0.00  |
| 277.00 | 7.089  | -15.461 | -0.639 | 0.00 | 0.    | 0.    | 7.09  | 0.00 | 0.00  |
| 287.00 | 6.965  | -15.220 | -0.644 | 0.00 | 0.    | 0.    | 6.96  | 0.00 | 0.00  |
| 297.00 | 6.899  | -14.930 | -0.649 | 0.00 | 0.    | 0.    | 6.90  | 0.00 | 0.00  |
| 307.00 | 6.879  | -14.592 | -0.653 | 0.00 | 0.    | 0.    | 6.88  | 0.00 | 0.00  |
| 317.00 | 6.898  | -14.211 | -0.657 | 0.00 | 0.    | 0.    | 6.90  | 0.00 | 0.00  |
| 327.00 | 6.949  | -13.787 | -0.660 | 0.00 | 0.    | 0.    | 6.95  | 0.00 | 0.00  |
| 337.00 | 7.025  | -13.322 | -0.663 | 0.00 | 0.    | 0.    | 7.03  | 0.00 | 0.00  |
| 347.00 | 7.132  | -12.815 | -0.666 | 0.00 | 0.    | 0.    | 7.13  | 0.00 | 0.00  |
| 357.00 | 7.269  | -12.270 | -0.669 | 0.00 | 0.    | 0.    | 7.27  | 0.00 | 0.00  |
| 367.00 | 7.434  | -11.689 | -0.672 | 0.00 | 0.    | 0.    | 7.43  | 0.00 | 0.00  |
| 377.00 | 7.623  | -11.077 | -0.674 | 0.00 | 0.    | 0.    | 7.62  | 0.00 | 0.00  |
| 387.00 | 7.833  | -10.435 | -0.677 | 0.00 | 0.    | 0.    | 7.83  | 0.00 | 0.00  |
| 397.00 | 8.064  | -9.765  | -0.680 | 0.00 | 0.    | 0.    | 8.06  | 0.00 | 0.00  |
| 407.00 | 8.316  | -9.070  | -0.683 | 0.00 | 0.    | 0.    | 8.32  | 0.00 | 0.00  |
| 417.00 | 8.590  | -8.351  | -0.686 | 0.00 | 0.    | 0.    | 8.59  | 0.00 | 0.00  |
| 427.00 | 8.885  | -7.611  | -0.690 | 0.00 | 0.    | 0.    | 8.89  | 0.00 | 0.00  |
| 437.00 | 9.203  | -6.852  | -0.693 | 0.00 | 0.    | 0.    | 9.20  | 0.00 | 0.00  |
| 447.00 | 9.544  | -6.074  | -0.697 | 0.00 | 0.    | 0.    | 9.54  | 0.00 | 0.00  |
| 457.00 | 9.909  | -5.282  | -0.700 | 0.00 | 0.    | 0.    | 9.91  | 0.00 | 0.00  |
| 467.00 | 10.297 | -4.479  | -0.704 | 0.00 | 0.    | 0.    | 10.30 | 0.00 | 0.00  |
| 477.00 | 10.711 | -3.665  | -0.707 | 0.00 | 0.    | 0.    | 10.71 | 0.00 | 0.00  |
| 487.00 | 11.149 | -2.844  | -0.711 | 0.00 | 0.    | 0.    | 11.15 | 0.00 | 0.00  |
| 497.00 | 11.613 | -2.030  | -0.714 | 0.00 | 0.    | 0.    | 11.61 | 0.00 | 0.00  |
| 498.78 | 11.698 | -1.877  | -0.715 | 0.00 | 0.    | 0.    | 11.70 | 0.00 | 0.00  |
| 498.78 | 11.698 | -1.877  | -0.715 | 0.00 | 0.    | 0.    | 11.70 | 0.00 | 0.00  |
| 157.39 | 8.762  | -5.237  | -0.503 | 5.70 | -30.  | -3.   | 8.76  | 0.00 | 0.00  |
| 166.61 | 9.657  | -4.755  | -0.501 | 2.36 | -11.  | -1.   | 9.66  | 0.00 | 0.00  |
| 177.00 | 10.682 | -4.243  | -0.508 | 0.77 | -3.   | 0.    | 10.68 | 0.00 | 0.00  |
| 187.00 | 11.184 | -3.611  | -0.513 | 0.24 | -1.   | 0.    | 11.18 | 0.00 | 0.00  |
| 197.00 | 11.237 | -2.875  | -0.517 | 0.08 | 0.    | 0.    | 11.24 | 0.00 | 0.00  |
| 207.00 | 11.156 | -2.052  | -0.521 | 0.03 | 0.    | 0.    | 11.16 | 0.00 | 0.00  |
| 217.00 | 11.060 | -1.160  | -0.525 | 0.01 | 0.    | 0.    | 11.06 | 0.00 | 0.00  |
| 227.00 | 10.646 | -0.210  | -0.530 | 0.01 | 0.    | 0.    | 10.65 | 0.00 | 0.00  |
| 236.59 | 10.165 | 0.737   | -0.536 | 0.00 | 0.    | 0.    | 10.16 | 0.00 | 0.00  |
| 236.59 | 10.165 | 0.741   | -0.536 | 0.00 | 0.    | 0.    | 10.16 | 0.00 | 0.00  |
| 245.61 | 9.630  | 1.661   | -0.541 | 0.00 | 0.    | 0.    | 9.63  | 0.00 | 0.00  |
| 256.00 | 9.349  | 2.747   | -0.548 | 0.00 | 0.    | 0.    | 9.35  | 0.00 | 0.00  |
| 266.00 | 9.285  | 3.808   | -0.555 | 0.00 | 0.    | 0.    | 9.28  | 0.00 | 0.00  |
| 276.00 | 9.361  | 4.866   | -0.563 | 0.00 | 0.    | 0.    | 9.36  | 0.00 | 0.00  |
| 286.00 | 9.563  | 5.908   | -0.570 | 0.00 | 0.    | 0.    | 9.56  | 0.00 | 0.00  |
| 296.00 | 9.872  | 6.922   | -0.578 | 0.00 | 0.    | 0.    | 9.87  | 0.00 | 0.00  |
| 302.09 | 10.126 | 7.522   | -0.583 | 0.00 | 0.    | 0.    | 10.13 | 0.00 | 0.00  |
| 302.09 | 10.126 | 7.522   | -0.583 | 0.00 | 0.    | 0.    | 10.13 | 0.00 | 0.00  |

INJECTION

G-LIMIT

| TIME   | MACH   | ALPHA  | BETA   | Q    | QALPH | QBETA | EMACH | INBD | OUTBD |
|--------|--------|--------|--------|------|-------|-------|-------|------|-------|
| 306.00 | 10.233 | 7.962  | -0.590 | 0.00 | 0.    | 0.    | 10.23 | 0.00 | 0.00  |
| 316.00 | 10.585 | 9.049  | -0.609 | 0.00 | 0.    | 0.    | 10.58 | 0.00 | 0.00  |
| 326.00 | 11.038 | 10.082 | -0.627 | 0.00 | 0.    | 0.    | 11.04 | 0.00 | 0.00  |
| 336.00 | 11.603 | 11.043 | -0.643 | 0.00 | 0.    | 0.    | 11.60 | 0.00 | 0.00  |
| 337.09 | 11.672 | 11.143 | -0.645 | 0.00 | 0.    | 0.    | 11.67 | 0.00 | 0.00  |
| 337.09 | 11.672 | 11.143 | -0.645 | 0.00 | 0.    | 0.    | 11.67 | 0.00 | 0.00  |
| 337.09 | 11.672 | 11.143 | -0.645 | 0.00 | 0.    | 0.    | 11.67 | 0.00 | 0.00  |

INJECTION



|            | TIME  | THMPS    | TAU  | THSRM | THLBM | THOMS | THRCS | THRST    |
|------------|-------|----------|------|-------|-------|-------|-------|----------|
| LIFT-OFF   | 0.00  | 3307089. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3307089. |
|            | 1.00  | 3307299. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3307299. |
|            | 2.00  | 3307933. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3307933. |
|            | 3.00  | 3308995. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3308995. |
|            | 4.00  | 3310487. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3310487. |
|            | 5.00  | 3312413. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3312413. |
|            | 6.00  | 3314775. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3314775. |
|            | 7.00  | 3317574. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3317574. |
|            | 8.00  | 3320811. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3320811. |
|            | 9.00  | 3324485. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3324485. |
|            | 10.00 | 3328596. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3328596. |
|            | 11.00 | 3333141. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3333141. |
|            | 12.00 | 3338119. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3338119. |
|            | 13.00 | 3343523. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3343523. |
|            | 14.00 | 3349348. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3349348. |
|            | 15.00 | 3355588. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3355588. |
|            | 16.00 | 3362234. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3362234. |
|            | 17.00 | 3369277. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3369277. |
|            | 18.00 | 3376704. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3376704. |
| BEGIN TILT | 19.00 | 3384501. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3384501. |
|            | 20.00 | 3392656. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3392656. |
|            | 21.00 | 3401154. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3401154. |
|            | 22.00 | 3409975. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3409975. |
|            | 23.00 | 3419102. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3419102. |
|            | 24.00 | 3428520. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3428520. |
|            | 25.00 | 3438204. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3438204. |
|            | 26.00 | 3448135. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3448135. |
|            | 27.00 | 3458299. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3458299. |
|            | 28.00 | 3468696. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3468696. |
|            | 29.00 | 3479221. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3479221. |
|            | 30.00 | 3489836. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3489836. |
|            | 30.00 | 3489836. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3489836. |
|            | 31.00 | 3500682. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3500682. |
|            | 32.00 | 3511739. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3511739. |
|            | 33.00 | 3522952. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3522952. |
|            | 34.00 | 3534261. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3534261. |
|            | 35.00 | 3545645. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3545645. |
|            | 36.00 | 3557098. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3557098. |
| END TILT   | 37.00 | 3568612. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3568612. |
|            | 38.00 | 3580174. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3580174. |
|            | 38.00 | 3580174. | 1.00 | 0.    | 0.    | 0.    | 0.    | 3580174. |
|            | 39.00 | 2800174. | 0.80 | 0.    | 0.    | 0.    | 0.    | 2800174. |
|            | 39.00 | 2811647. | 0.80 | 0.    | 0.    | 0.    | 0.    | 2811647. |

| TIME  | THMP5    | TAU  | THSRM | THLBM | THOMS | THRCS | THRST    |
|-------|----------|------|-------|-------|-------|-------|----------|
| 40.00 | 2822950. | 0.80 | 0.    | 0.    | 0.    | 0.    | 2822950. |
| 41.00 | 2834096. | 0.80 | 0.    | 0.    | 0.    | 0.    | 2834096. |
| 42.00 | 2845059. | 0.80 | 0.    | 0.    | 0.    | 0.    | 2845059. |
| 43.00 | 2855829. | 0.80 | 0.    | 0.    | 0.    | 0.    | 2855829. |
| 44.00 | 2866418. | 0.80 | 0.    | 0.    | 0.    | 0.    | 2866418. |
| 45.00 | 2876820. | 0.80 | 0.    | 0.    | 0.    | 0.    | 2876820. |
| 46.00 | 2887010. | 0.80 | 0.    | 0.    | 0.    | 0.    | 2887010. |
| 47.00 | 2897003. | 0.80 | 0.    | 0.    | 0.    | 0.    | 2897003. |
| 48.00 | 2906805. | 0.80 | 0.    | 0.    | 0.    | 0.    | 2906805. |
| 49.00 | 2916390. | 0.80 | 0.    | 0.    | 0.    | 0.    | 2916390. |
| 50.00 | 2925766. | 0.80 | 0.    | 0.    | 0.    | 0.    | 2925766. |
| 51.00 | 2934939. | 0.80 | 0.    | 0.    | 0.    | 0.    | 2934939. |
| 52.00 | 2943883. | 0.80 | 0.    | 0.    | 0.    | 0.    | 2943883. |
| 52.79 | 2950839. | 0.80 | 0.    | 0.    | 0.    | 0.    | 2950839. |
| 53.00 | 2952612. | 0.80 | 0.    | 0.    | 0.    | 0.    | 2952612. |
| 54.00 | 2961128. | 0.80 | 0.    | 0.    | 0.    | 0.    | 2961128. |
| 55.00 | 2969409. | 0.80 | 0.    | 0.    | 0.    | 0.    | 2969409. |
| 56.00 | 2977464. | 0.80 | 0.    | 0.    | 0.    | 0.    | 2977464. |
| 57.00 | 2985280. | 0.80 | 0.    | 0.    | 0.    | 0.    | 2985280. |
| 58.00 | 2992847. | 0.80 | 0.    | 0.    | 0.    | 0.    | 2992847. |
| 59.00 | 3000171. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3000171. |
| 60.00 | 3007229. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3007229. |
| 61.00 | 3014029. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3014029. |
| 62.00 | 3020567. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3020567. |
| 63.00 | 3026828. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3026828. |
| 64.00 | 3032829. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3032829. |
| 65.00 | 3038548. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3038548. |
| 66.00 | 3043998. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3043998. |
| 67.00 | 3049157. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3049157. |
| 68.00 | 3054053. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3054053. |
| 69.00 | 3058728. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3058728. |
| 70.00 | 3063092. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3063092. |
| 71.00 | 3067162. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3067162. |
| 72.00 | 3070985. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3070985. |
| 73.00 | 3074578. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3074578. |
| 74.00 | 3077900. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3077900. |
| 75.00 | 3080994. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3080994. |
| 76.00 | 3083903. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3083903. |
| 77.00 | 3086576. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3086576. |
| 78.00 | 3089017. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3089017. |
| 79.00 | 3091325. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3091325. |
| 80.00 | 3093499. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3093499. |
| 81.00 | 3095535. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3095535. |
| 82.00 | 3097418. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3097418. |
| 83.00 | 3099134. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3099134. |

MAX DYN PRES

| TIME   | THMPs    | TAU  | THSRM | THLBM | THOMS | THRCS | THRST    |
|--------|----------|------|-------|-------|-------|-------|----------|
| 84.00  | 3100714. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3100714. |
| 85.00  | 3102194. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3102194. |
| 86.00  | 3103579. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3103579. |
| 87.00  | 3104840. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3104840. |
| 88.00  | 3106002. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3106002. |
| 89.00  | 3107093. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3107093. |
| 90.00  | 3108105. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3108105. |
| 91.00  | 3109027. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3109027. |
| 92.00  | 3109881. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3109881. |
| 93.00  | 3110683. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3110683. |
| 94.00  | 3111420. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3111420. |
| 95.00  | 3112095. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3112095. |
| 96.00  | 3112722. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3112722. |
| 97.00  | 3113306. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3113306. |
| 98.00  | 3113837. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3113837. |
| 99.00  | 3114324. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3114324. |
| 100.00 | 3114780. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3114780. |
| 101.00 | 3115197. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3115197. |
| 102.00 | 3115578. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3115578. |
| 103.00 | 3115931. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3115931. |
| 104.00 | 3116259. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3116259. |
| 105.00 | 3116556. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3116556. |
| 106.00 | 3116830. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3116830. |
| 107.00 | 3117086. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3117086. |
| 108.00 | 3117318. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3117318. |
| 109.00 | 3117531. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3117531. |
| 110.00 | 3117729. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3117729. |
| 111.00 | 3117912. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3117912. |
| 112.00 | 3118079. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3118079. |
| 113.00 | 3118228. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3118228. |
| 114.00 | 3118365. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3118365. |
| 115.00 | 3118491. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3118491. |
| 116.00 | 3118609. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3118609. |
| 117.00 | 3118719. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3118719. |
| 118.00 | 3118822. | 8.80 | 0.    | 0.    | 0.    | 0.    | 3118822. |
| 119.00 | 3118918. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3118918. |
| 120.00 | 3119007. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119007. |
| 121.00 | 3119087. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119087. |
| 122.00 | 3119158. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119158. |
| 123.00 | 3119222. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119222. |
| 124.00 | 3119282. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119282. |
| 125.00 | 3119339. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119339. |
| 126.00 | 3119393. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119393. |
| 127.00 | 3119442. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119442. |
| 128.00 | 3119487. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119487. |

| TIME   | THMP5    | TAU  | THSRM | THLBM | THOMS | THRCS | THRST    |
|--------|----------|------|-------|-------|-------|-------|----------|
| 129.00 | 3119527. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119527. |
| 130.00 | 3119564. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119564. |
| 131.00 | 3119599. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119599. |
| 132.00 | 3119633. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119633. |
| 133.00 | 3119663. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119663. |
| 134.00 | 3119691. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119691. |
| 135.00 | 3119716. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119716. |
| 136.00 | 3119740. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119740. |
| 137.00 | 3119762. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119762. |
| 138.00 | 3119783. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119783. |
| 139.00 | 3119803. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119803. |
| 140.00 | 3119821. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119821. |
| 141.00 | 3119837. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119837. |
| 142.00 | 3119851. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119851. |
| 143.00 | 3119865. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119865. |
| 144.00 | 3119877. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119877. |
| 145.00 | 3119888. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119888. |
| 146.00 | 3119899. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119899. |
| 147.00 | 3119909. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119909. |
| 148.00 | 3119918. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119918. |
| 149.00 | 3119926. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119926. |
| 150.00 | 3119933. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119933. |
| 151.00 | 3119939. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119939. |
| 152.00 | 3119946. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119946. |
| 153.00 | 3119951. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119951. |
| 154.00 | 3119957. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119957. |
| 155.00 | 3119961. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119961. |
| 156.00 | 3119965. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119965. |
| 156.39 | 3119967. | 0.80 | 0.    | 0.    | 0.    | 0.    | 3119967. |
| 156.39 | 3119967. | 1.00 | 0.    | 0.    | 0.    | 0.    | 1113040. |
| 157.00 | 1113040. | 1.00 | 0.    | 0.    | 0.    | 0.    | 1039970. |
| 157.39 | 1039970. | 1.00 | 0.    | 0.    | 0.    | 0.    | 649995.  |
| 157.39 | 649995.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 649995.  |
| 157.39 | 649995.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 649998.  |
| 167.00 | 649998.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 177.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 187.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 197.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 207.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 217.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 227.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 237.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 247.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 247.39 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 247.39 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |

TAILOFF

SEPARATION  
BEGIN MINH

SHROUD SEP

| TIME   | THMP5    | TAU  | THSRM | THLBM | THOMS | THRCS | THRST    |
|--------|----------|------|-------|-------|-------|-------|----------|
| 257.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 267.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 277.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 287.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 297.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 307.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 317.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 327.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 337.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 347.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 357.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 367.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 377.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 387.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 397.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 407.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 417.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 427.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 437.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 447.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 457.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 467.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 477.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 487.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 497.00 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 498.78 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 498.78 | 650000.  | 1.00 | 0.    | 0.    | 0.    | 0.    | 650000.  |
| 157.39 | 1299999. | 1.00 | 0.    | 0.    | 0.    | 0.    | 1299999. |
| 166.61 | 1299997. | 1.00 | 0.    | 0.    | 0.    | 0.    | 1299997. |
| 177.00 | 1299999. | 1.00 | 0.    | 0.    | 0.    | 0.    | 1299999. |
| 187.00 | 1300000. | 1.00 | 0.    | 0.    | 0.    | 0.    | 1300000. |
| 197.00 | 1300000. | 1.00 | 0.    | 0.    | 0.    | 0.    | 1300000. |
| 207.00 | 1300000. | 1.00 | 0.    | 0.    | 0.    | 0.    | 1300000. |
| 217.00 | 1300000. | 1.00 | 0.    | 0.    | 0.    | 0.    | 1300000. |
| 227.00 | 1300000. | 1.00 | 0.    | 0.    | 0.    | 0.    | 1300000. |
| 236.59 | 1300000. | 1.00 | 0.    | 0.    | 0.    | 0.    | 1300000. |
| 236.59 | 1300000. | 1.00 | 0.    | 0.    | 0.    | 0.    | 1300000. |
| 245.61 | 1300000. | 1.00 | 0.    | 0.    | 0.    | 0.    | 1300000. |
| 256.00 | 1300000. | 1.00 | 0.    | 0.    | 0.    | 0.    | 1300000. |
| 266.00 | 1300000. | 1.00 | 0.    | 0.    | 0.    | 0.    | 1300000. |
| 276.00 | 1300000. | 1.00 | 0.    | 0.    | 0.    | 0.    | 1300000. |
| 286.00 | 1300000. | 1.00 | 0.    | 0.    | 0.    | 0.    | 1300000. |
| 296.00 | 1300000. | 1.00 | 0.    | 0.    | 0.    | 0.    | 1300000. |
| 302.09 | 1300000. | 1.00 | 0.    | 0.    | 0.    | 0.    | 1300000. |
| 302.09 | 910000.  | 0.70 | 0.    | 0.    | 0.    | 0.    | 910000.  |

INJECTION

G-LIMIT

| TIME   | THMPS   | TAU  | THSRM | THLEB | THOMS | THRCS | THRST   |
|--------|---------|------|-------|-------|-------|-------|---------|
| 306.00 | 910000. | 0.70 | 0.    | 0.    | 0.    | 0.    | 910000. |
| 316.00 | 910000. | 0.70 | 0.    | 0.    | 0.    | 0.    | 910000. |
| 326.00 | 910000. | 0.70 | 0.    | 0.    | 0.    | 0.    | 910000. |
| 336.00 | 910000. | 0.70 | 0.    | 0.    | 0.    | 0.    | 910000. |
| 337.09 | 910000. | 0.70 | 0.    | 0.    | 0.    | 0.    | 910000. |
| 337.09 | 0.      | 0.70 | 0.    | 0.    | 0.    | 0.    | 0.      |
| 337.09 | 0.      | 0.70 | 0.    | 0.    | 0.    | 0.    | 0.      |

INJECTION

#### **A.4 Candidate Flight Control Augmentors**

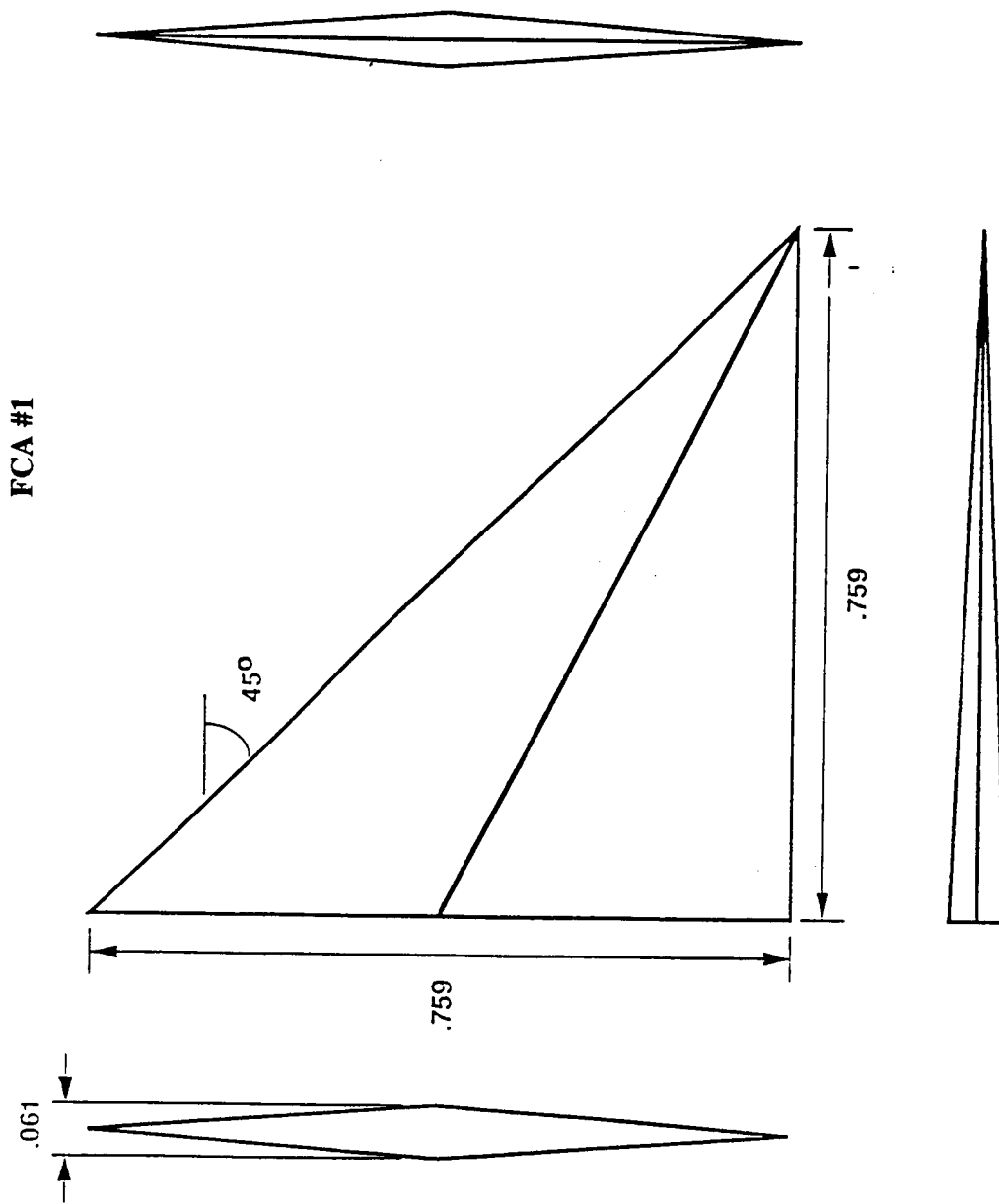


Figure A1. Candidate FCA's.



FCA #2

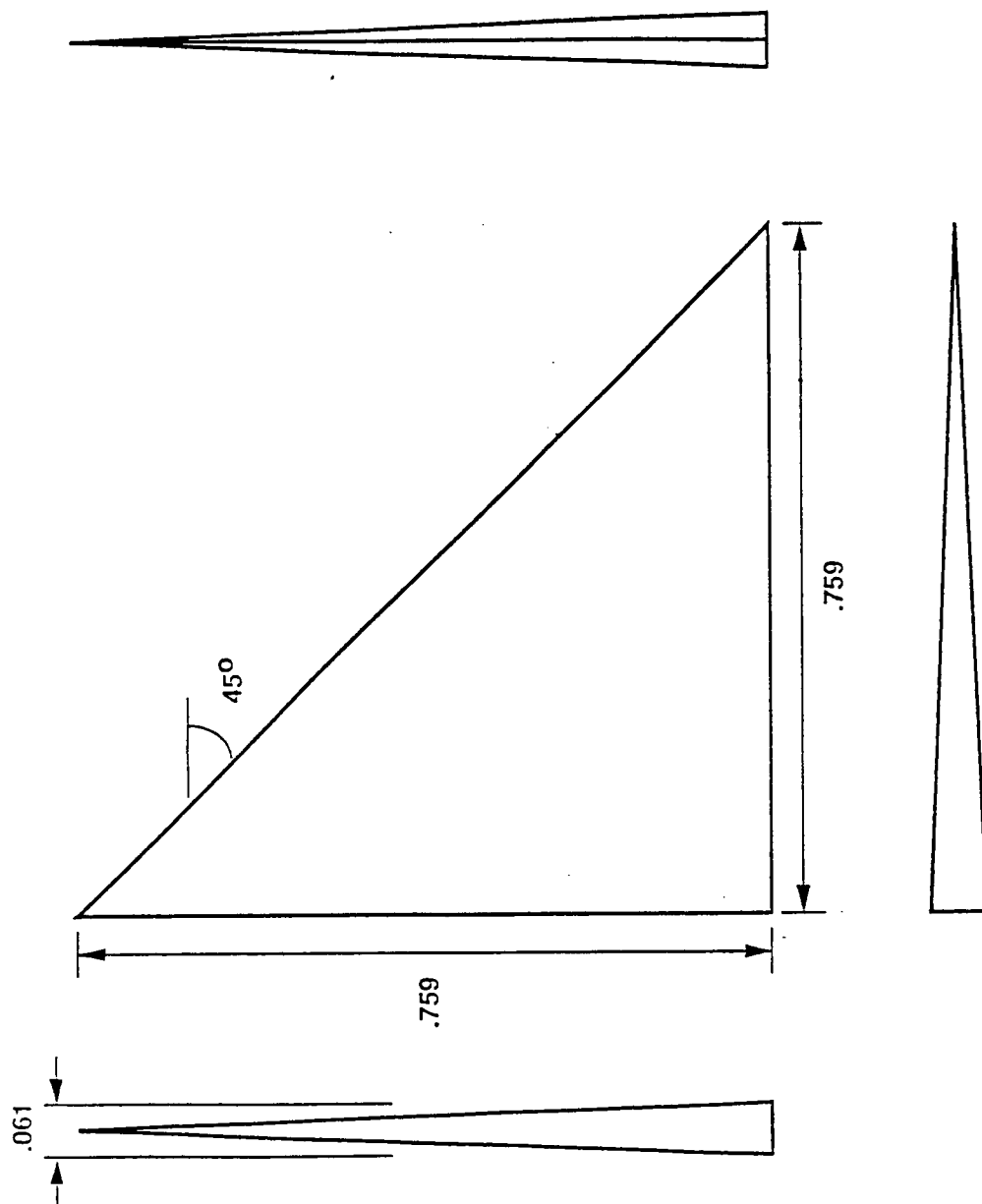


Figure A1. Candidate FCA's (continued).

FCA #3

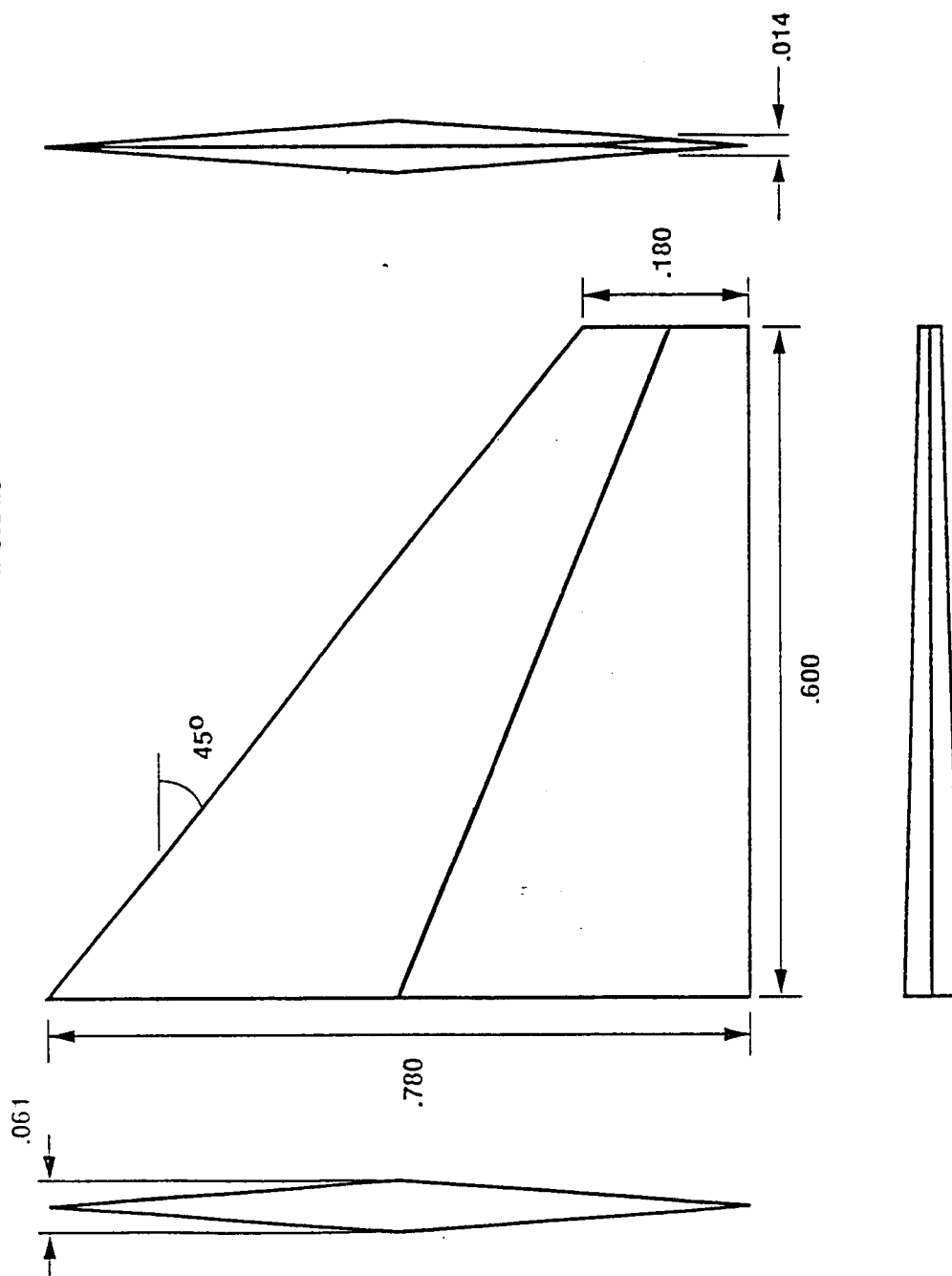


Figure A1. Candidate FCA's (continued).

FCA #4

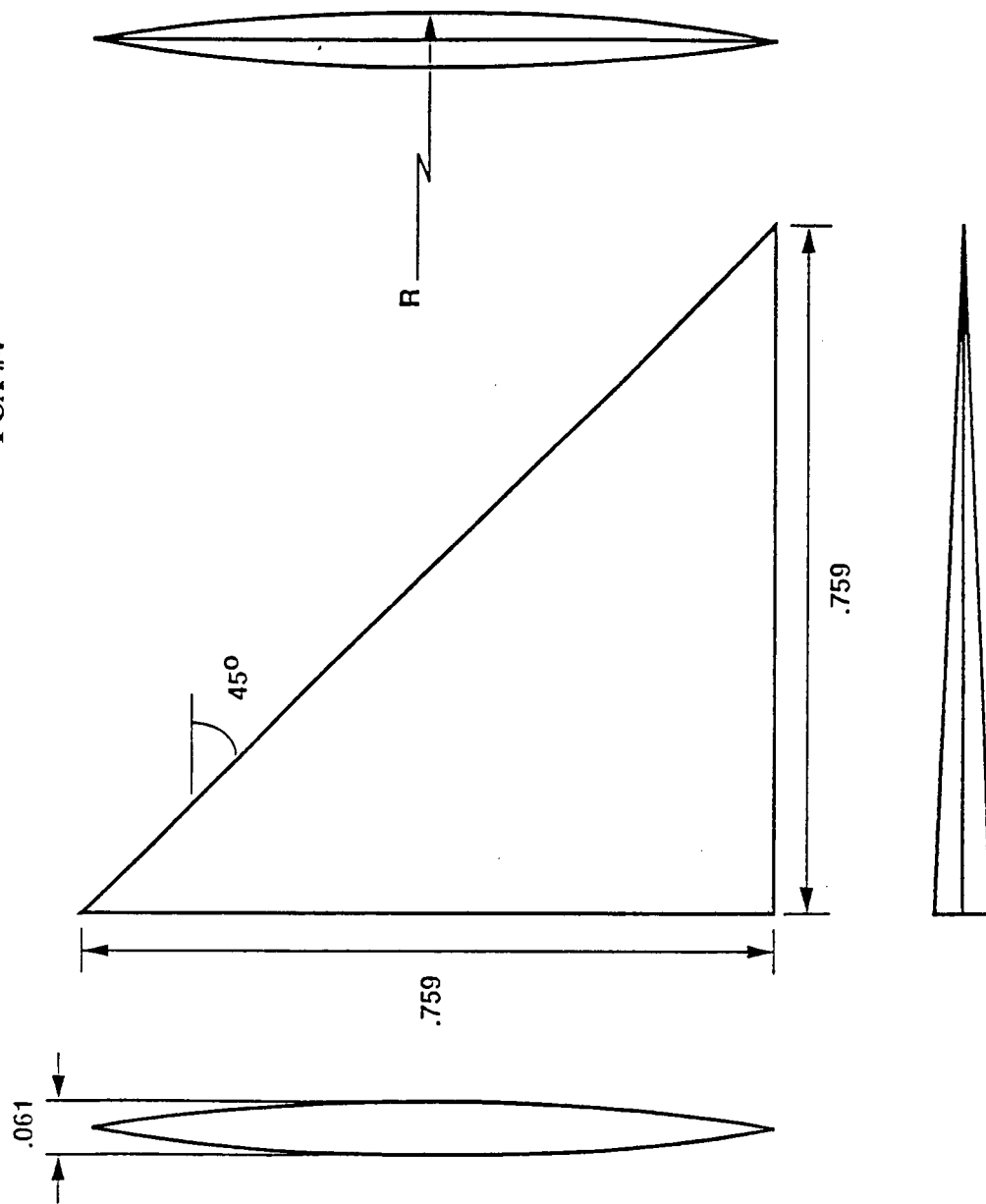


Figure A1. Candidate FCA's (continued).

FCA #5

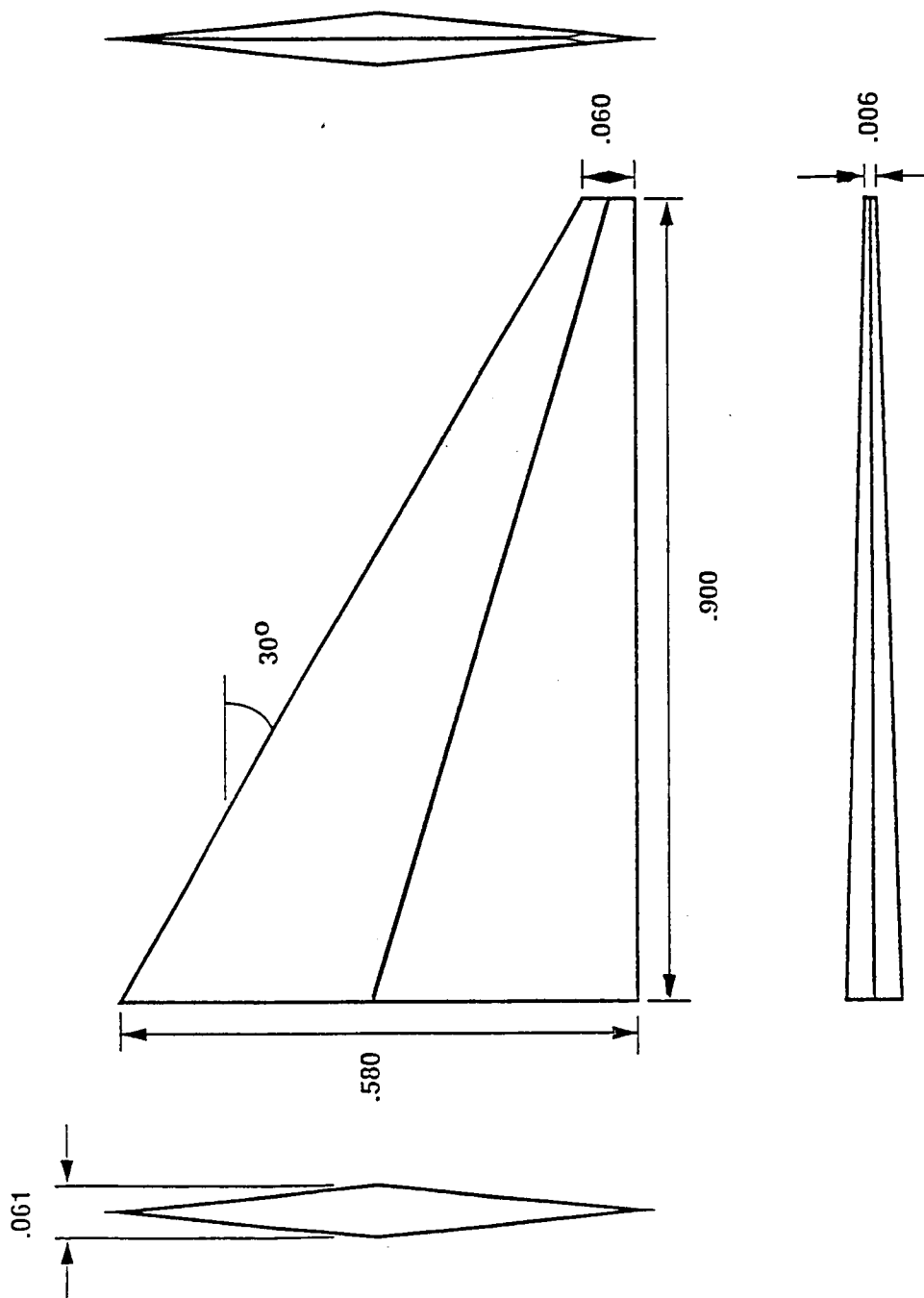


Figure A1. Candidate FCA's (continued).

FCA #6

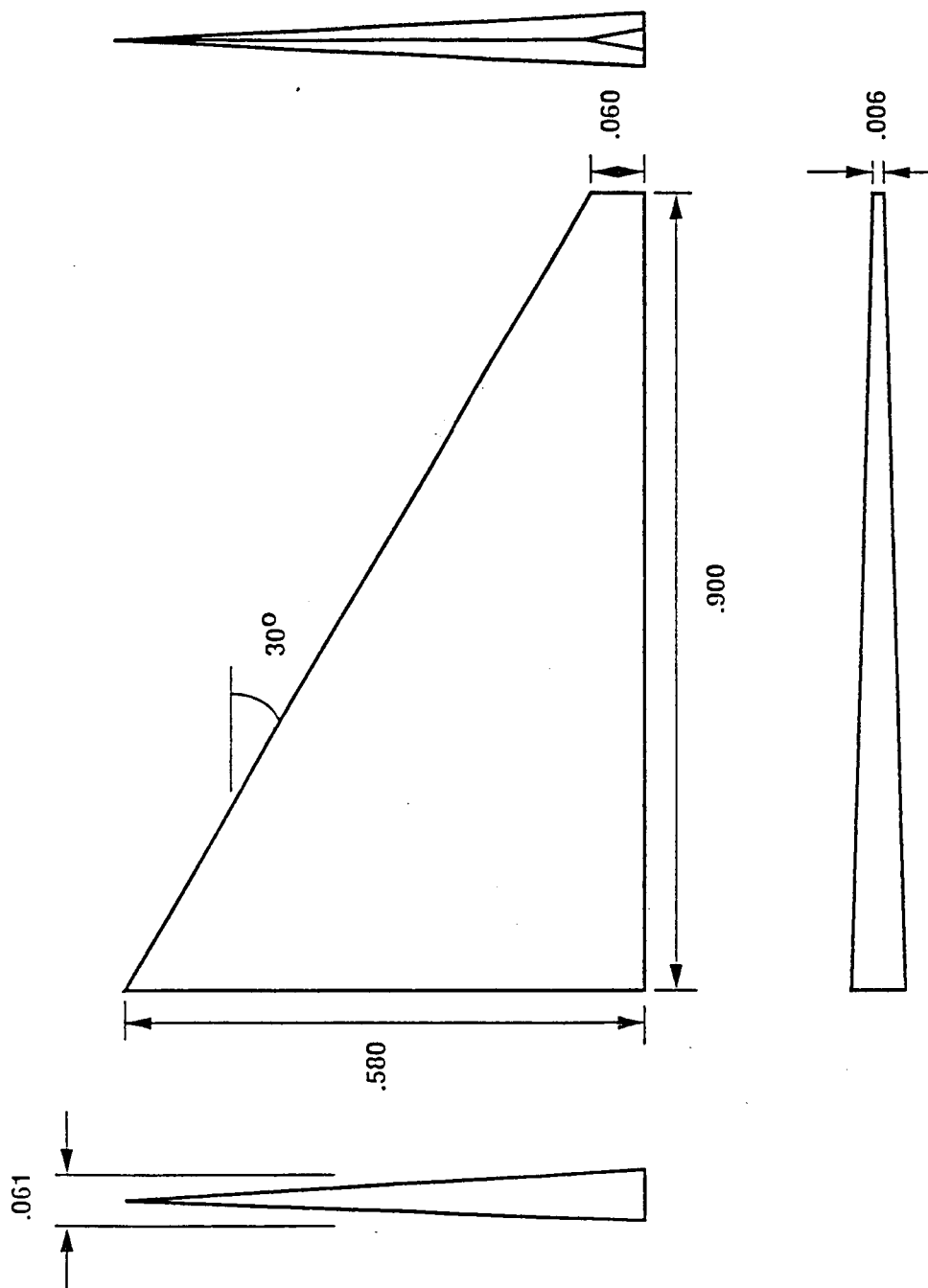


Figure A1. Candidate FCA's (continued).

FCA #7

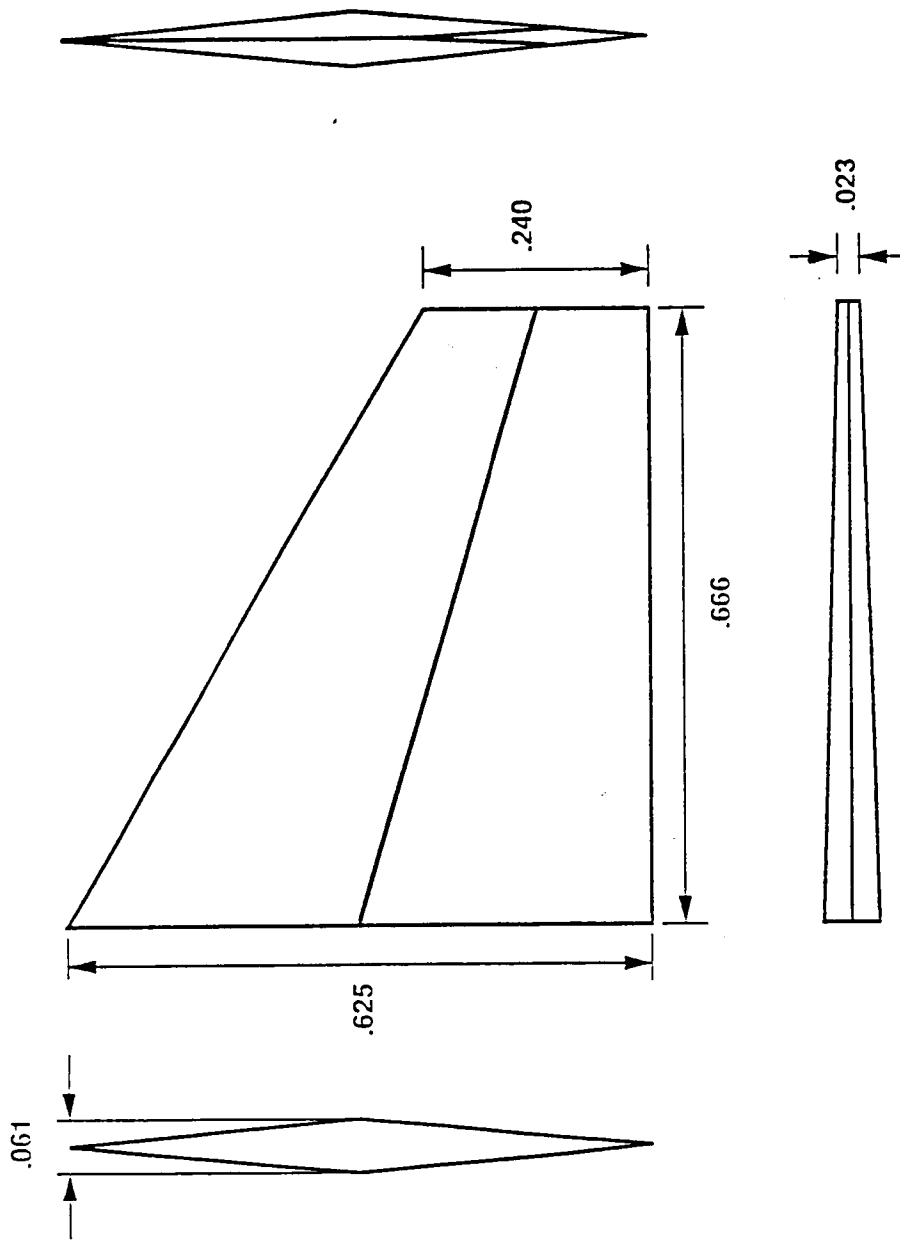


Figure A1. Candidate FCA's (continued).

FCA #8

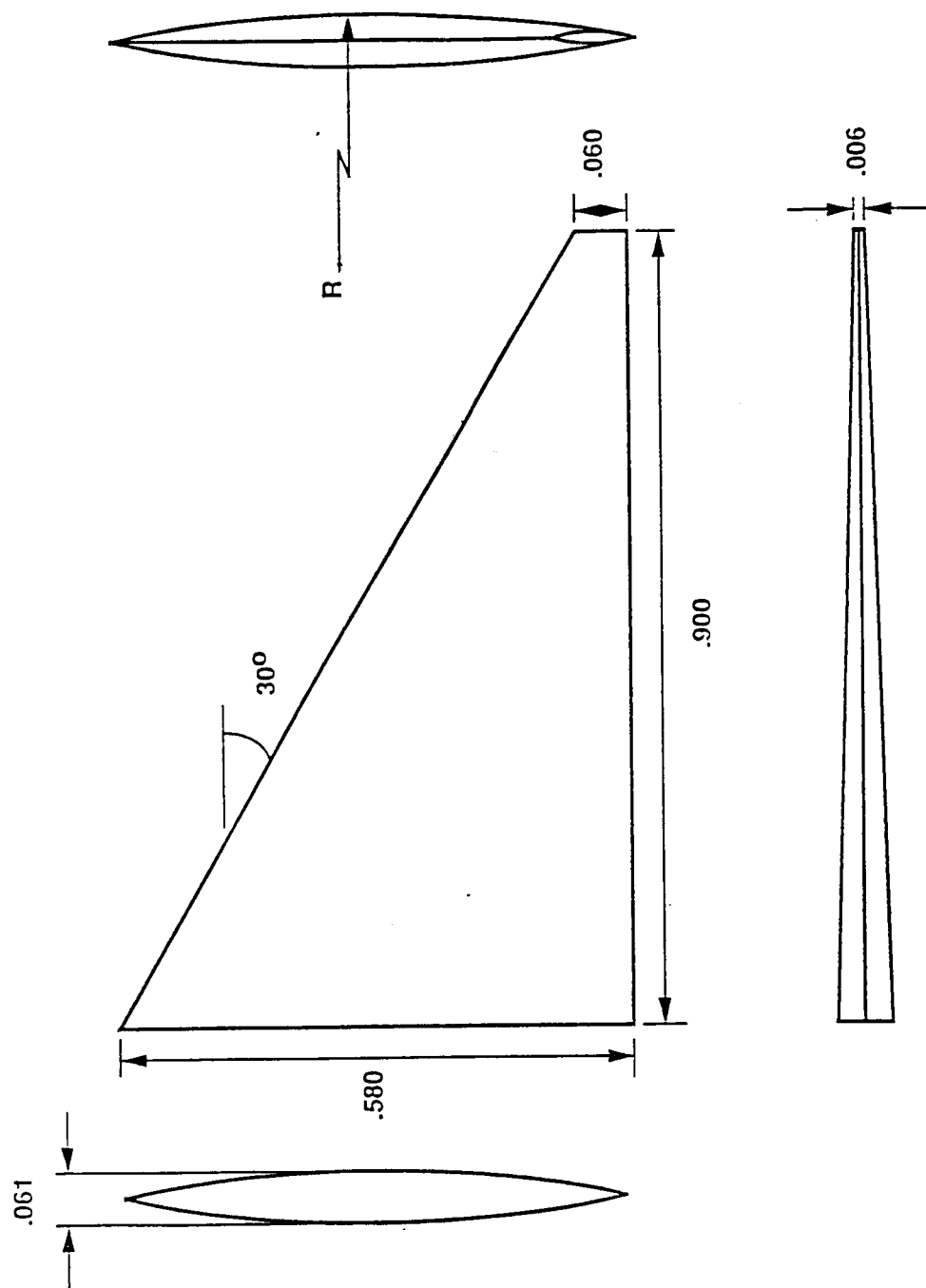


Figure A1. Candidate FCA's (continued).

FCA #9

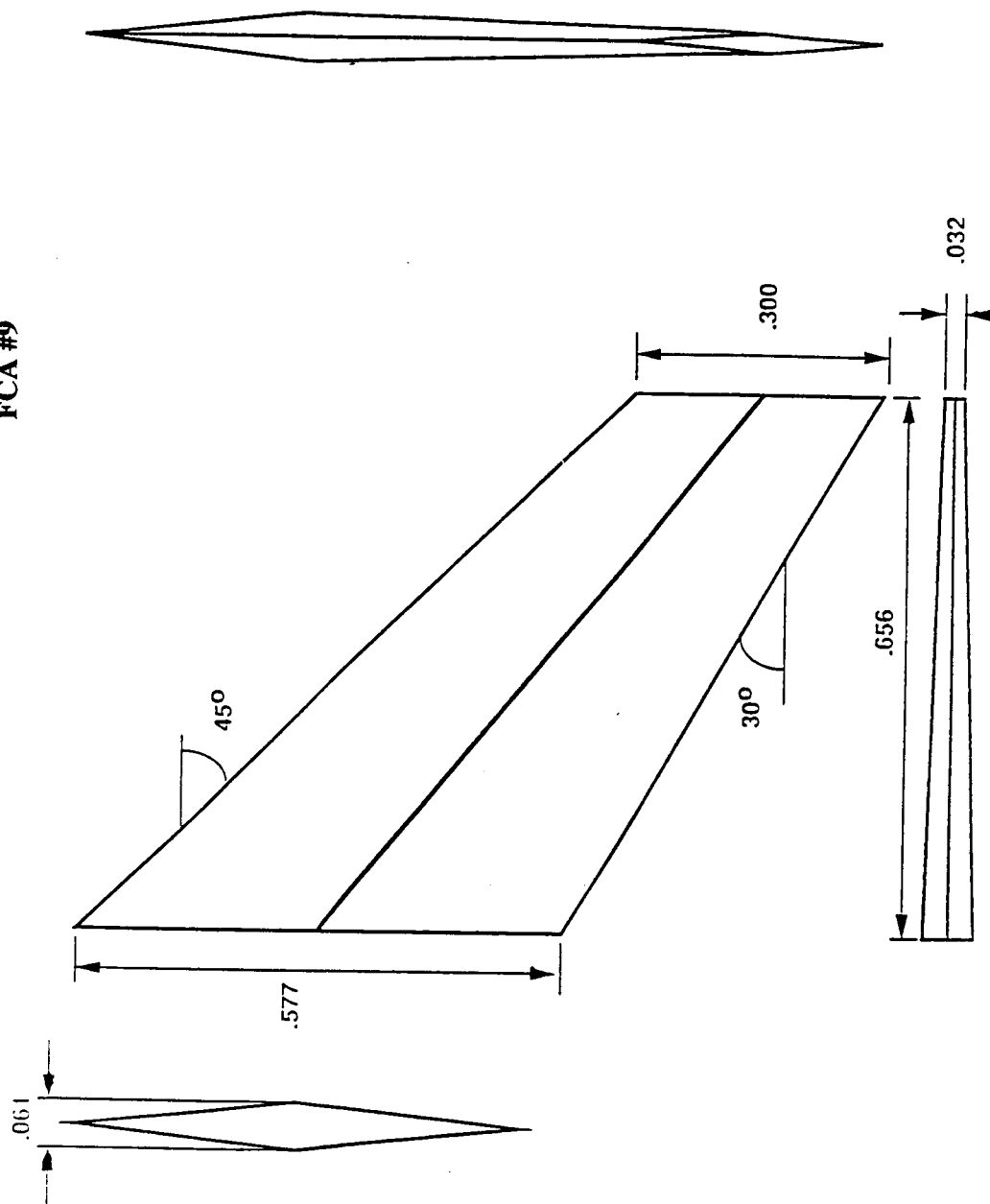


Figure A1. Candidate FCA's (continued).



FCA #10

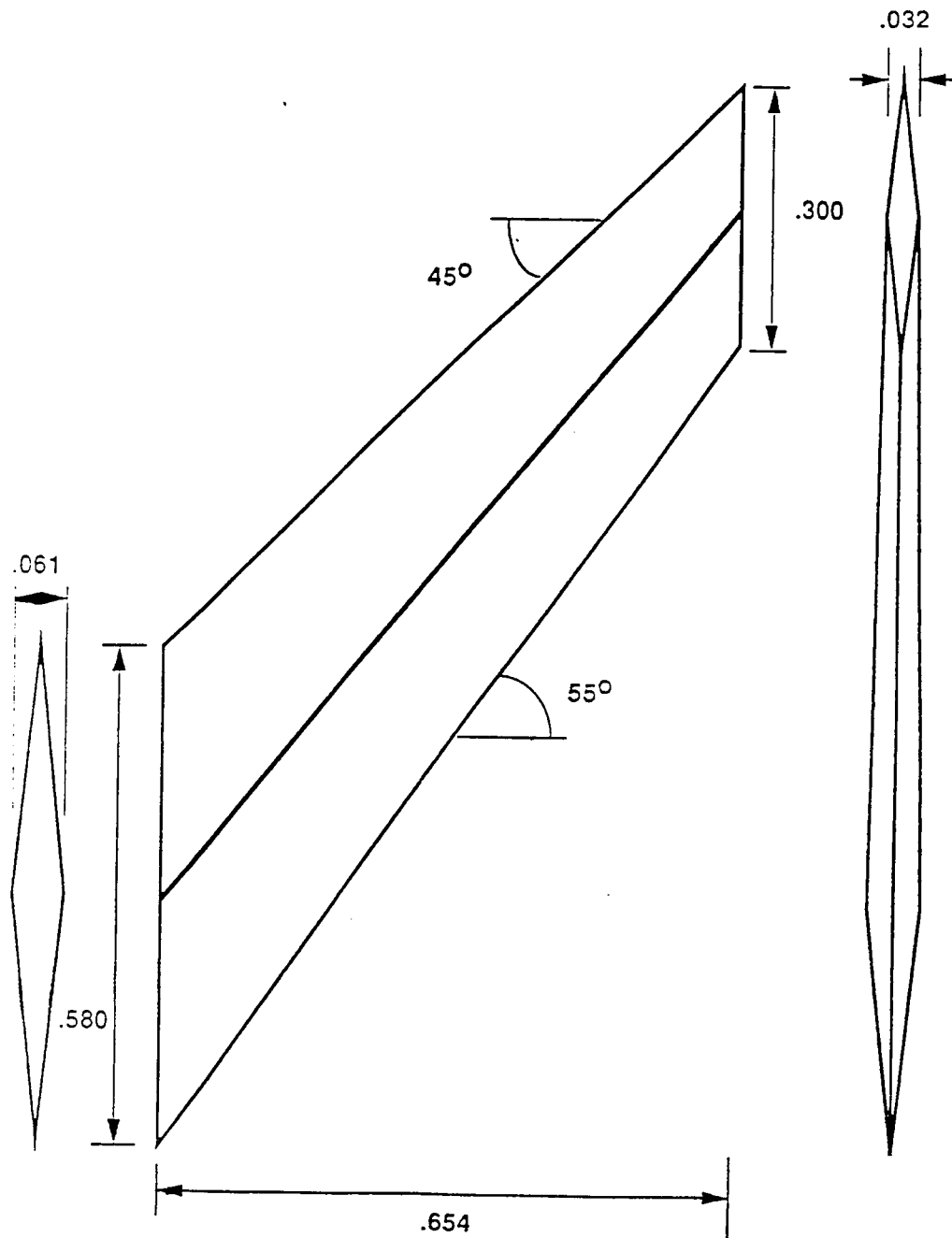


Figure A1. Candidate FCA's (continued).

FCA #11

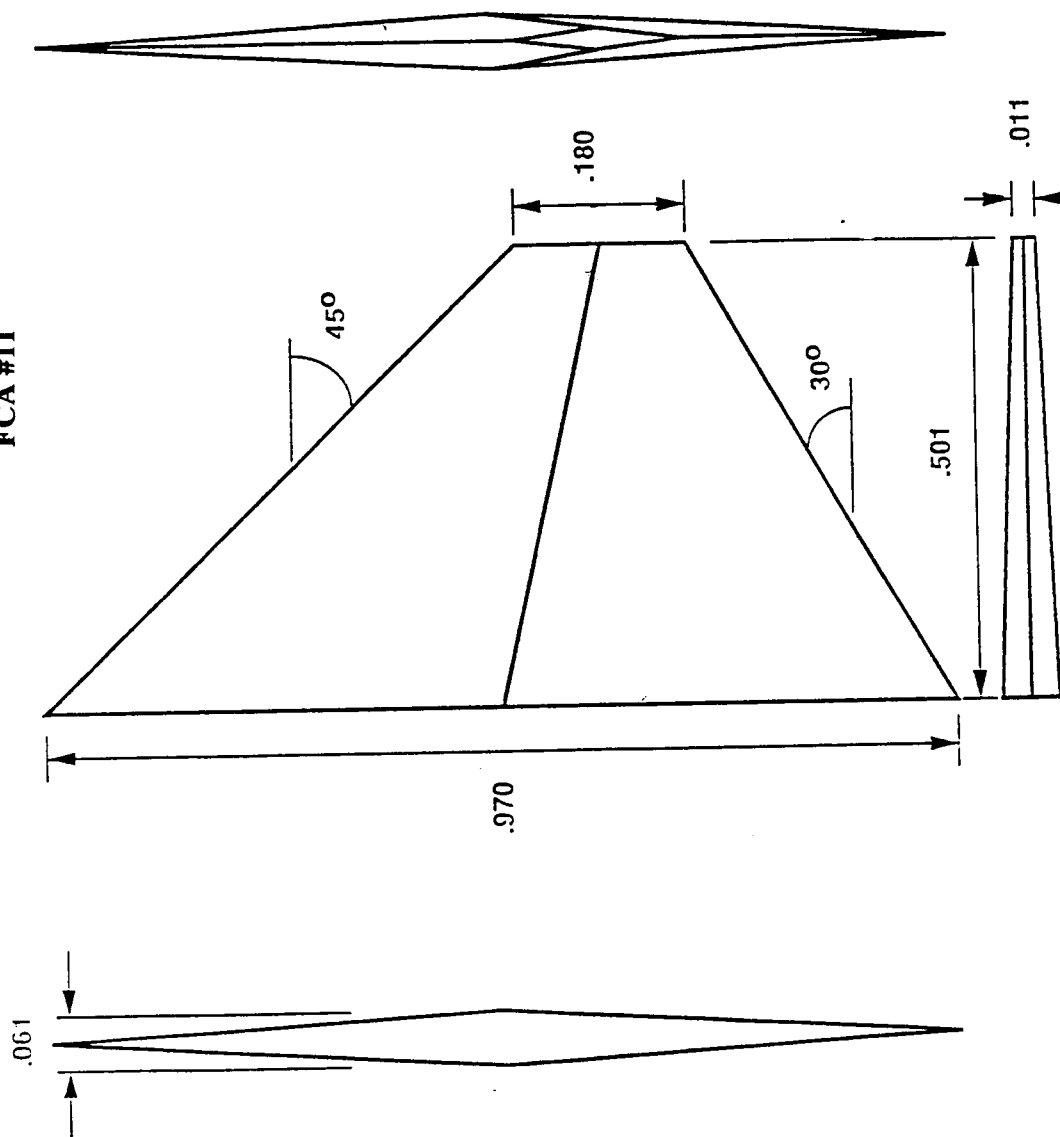


Figure A1. Candidate FCA's (continued).

# CANARD

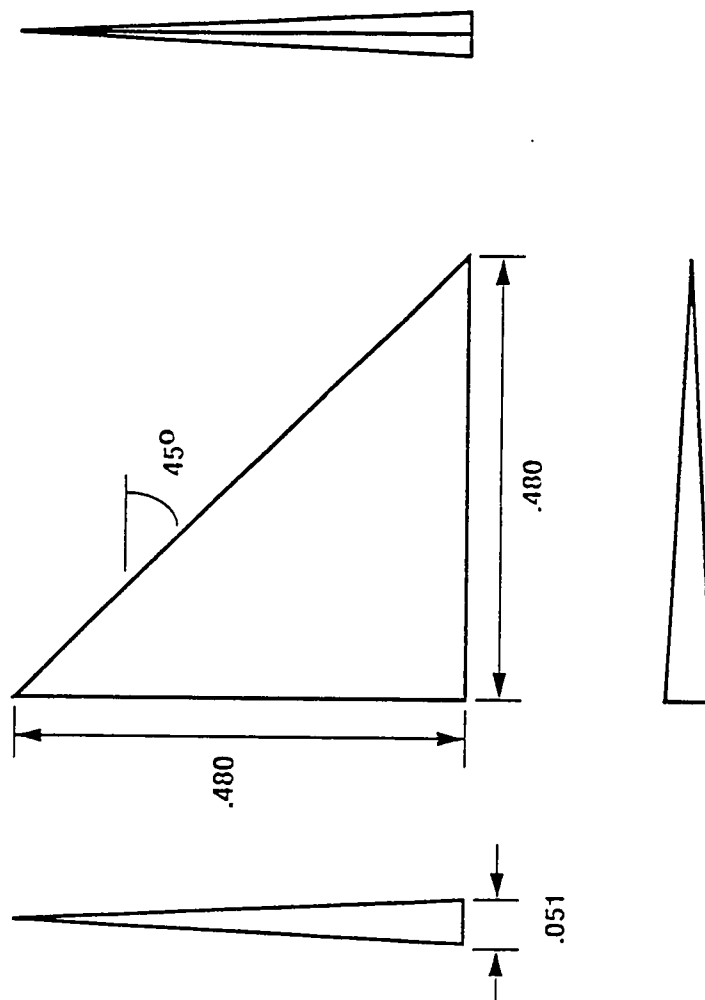


Figure A1. Candidate FCA's (continued).

## **APPENDIX B**

### **FABRICATED WIND TUNNEL TEST ARTICLES**

Table B1. Sample wind tunnel test run schedule.

| TWT 738       |        | MACH NUMBER |         |          |       |      |      |      |      |      |       |       |         |          |           |
|---------------|--------|-------------|---------|----------|-------|------|------|------|------|------|-------|-------|---------|----------|-----------|
| Configuration |        | $\alpha$    | $\beta$ | $\delta$ | #Runs | 0.20 | 0.40 | 0.80 | 0.95 | 1.20 | 1.46  | 1.96  | 2.74    | 3.48     | 4.96      |
| No            | FCA #1 | A           | 0       | -        | 13    | 1\3  | 2\3  | 3\1  | 4\1  | 5\1  | 371\0 | 716\0 | 496\8.9 | 497\9.10 | 498\40.41 |
|               |        |             |         | 0        | 13    | 10\0 | 9\0  | 8\0  | 7\0  | 6\1  | 372\0 | 717\0 | 501\0.1 | 500\0.1  | 499\1.2   |
|               |        |             |         | 3        | 13    | 15\0 | 14\0 | 13\0 | 12\0 | 11\0 | 373\0 | 718\0 | 502\0.1 | 503\0.1  | 504\0.1   |
|               |        |             |         | 6        | 13    | 16\0 | 17\0 | 18\0 | 19\0 | 20\0 | 374\0 | 719\0 | 505\0.1 | 506\0.1  | 507\0.1   |
|               |        |             |         | 9        | 13    | 25\0 | 24\0 | 23\0 | 22\0 | 21\0 | 375\0 | 720\0 | 508\0.1 | 509\0.1  | 510\0.1   |
|               |        |             |         | 12       | 13    | 26\0 | 27\0 | 28\0 | 29\0 | 30\0 | 376\0 | 721\0 | 511\0.1 | 512\0.1  | 513\0.2   |
|               |        |             |         | 15       | 13    | 35\0 | 34\0 | 33\0 | 32\0 | 31\0 | 377\0 | 722\0 | 514\0.1 | 515\0.1  | 516\0.2   |
|               | FCA #2 |             |         | 0        | 13    | 36\0 | 37\0 | 38\0 | 39\0 | 40\0 | 378\0 | 723\0 | 517\0.1 | 518\0.1  | 519\0.1   |
|               |        |             |         | 3        | 13    | 45\0 | 44\0 | 43\0 | 42\0 | 41\0 | 379\0 | 724\0 | 520\0.1 | 521\0.1  | 522\0.1   |
|               |        |             |         | 6        | 13    | 46\0 | 47\0 | 48\0 | 49\0 | 50\0 | 380\1 | 725\0 | 523\0.1 | 524\0.1  | 525\0.1   |
|               |        |             |         | 9        | 13    | 55\0 | 54\0 | 53\0 | 52\0 | 51\0 | 381\0 | 726\0 | 526\0.1 | 527\0.1  | 528\0.1   |
|               |        |             |         | 12       | 13    | 56\0 | 57\0 | 58\0 | 59\0 | 60\0 | 382\0 | 727\0 | 529\0.1 | 530\0.1  | 531\0.1   |
|               |        |             |         | 15       | 13    | 65\0 | 64\0 | 63\0 | 62\0 | 61\0 | 383\0 | 728\0 | 532\0.1 | 533\0.1  | 534\0.1   |
|               | FCA #3 |             |         | 0        | 13    | 66\0 | 67\0 | 68\0 | 69\0 | 70\0 | 384\0 | 729\0 | 535\0.1 | 536\0.1  | 537\0.1   |
|               |        |             |         | 3        | 13    | 75\0 | 74\0 | 73\0 | 72\0 | 71\1 | 385\0 | 730\0 | 538\0.1 | 539\0.1  | 540\0.1   |
|               |        |             |         | 6        | 13    | 76\0 | 77\0 | 78\0 | 79\0 | 80\0 | 386\0 | 731\0 | 541\0.1 | 542\0.1  | 543\0.1   |
|               |        |             |         | 9        | 13    | 85\0 | 84\0 | 83\0 | 82\0 | 81\0 | 387\0 | 732\0 | 544\0.1 | 545\0.1  | 546\0.1   |
|               |        |             |         | 12       | 13    | 86\0 | 87\0 | 88\0 | 89\0 | 90\0 | 388\0 | 733\0 | 547\0.1 | 548\0.1  | 549\0.1   |
|               |        | ▼           | ▼       | 15       | 13    | 95\0 | 94\0 | 93\2 | 92\0 | 91\0 | 389\0 | 734\0 | 550\2.3 | 551\0.1  | 552\0.1   |

$\alpha$  schedules: A = -10, -8, -6, -4, -2, 0, 2, 4, 6, 8, 10



Figure B1. Silicone carbon grit size #100 applied.



Figure B2. Nose cone silicon carbon grit size #100 applied.



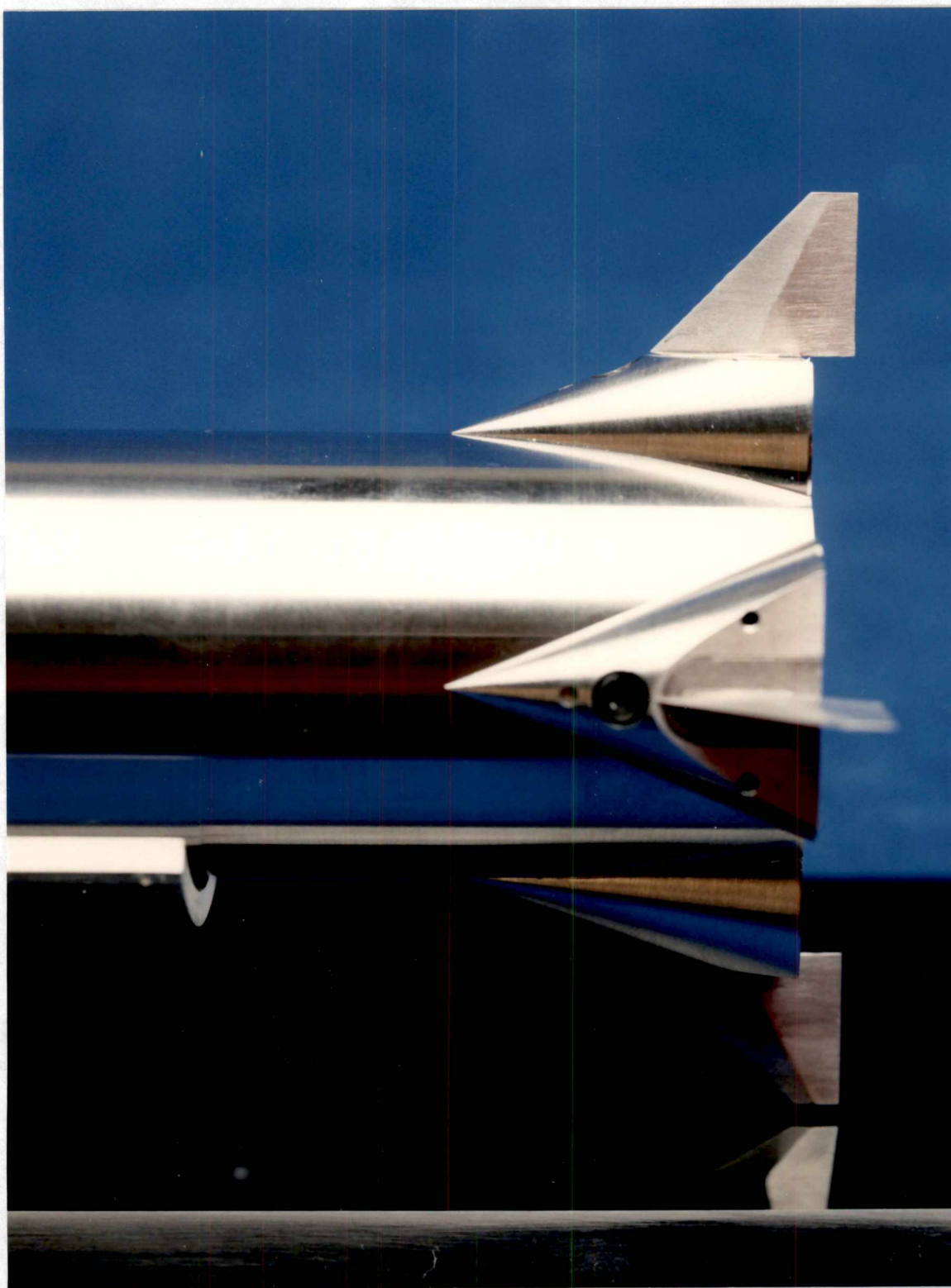


Figure B3. Fabricated test LV with FCA's #3.





Figure B4. Fabricated test LV with FCA's #9.



Figure B5. Fabricated test LV with fore and aft FCA's.



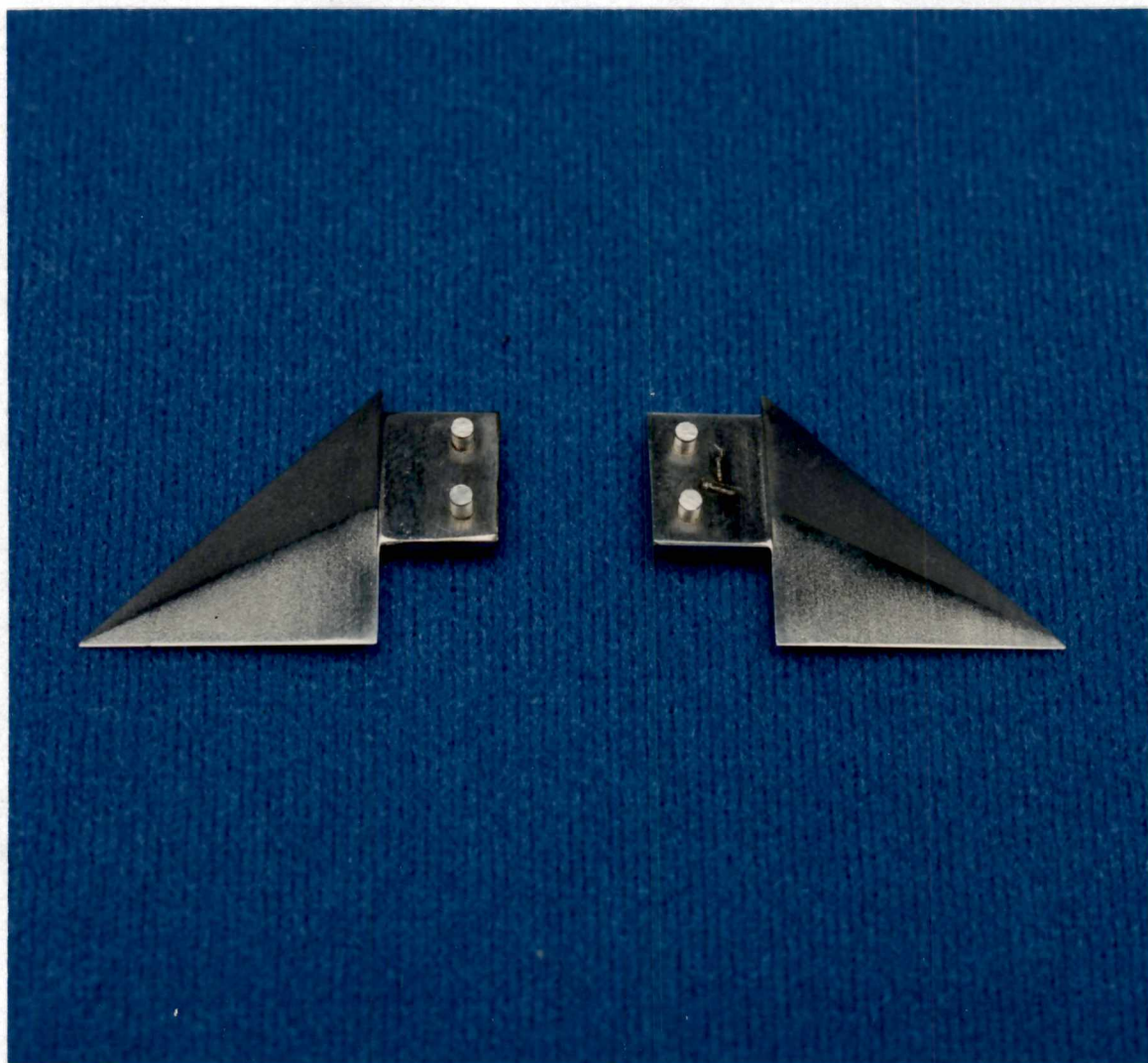


Figure B6. Fabricated FCA's #1.





Figure B7. Fabricated FCA's #2.



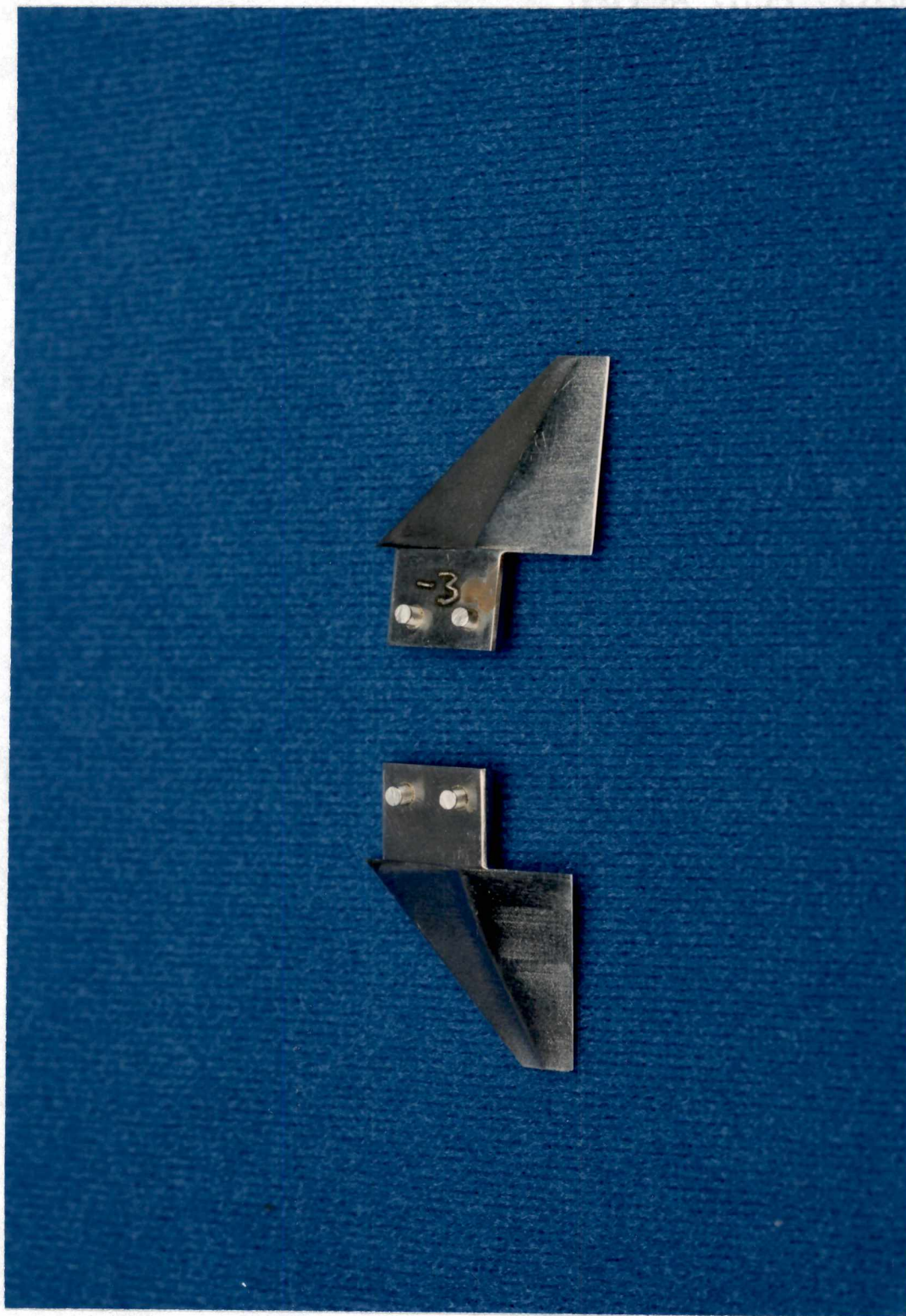


Figure B8. Horizontal fabricated FCA's #3.



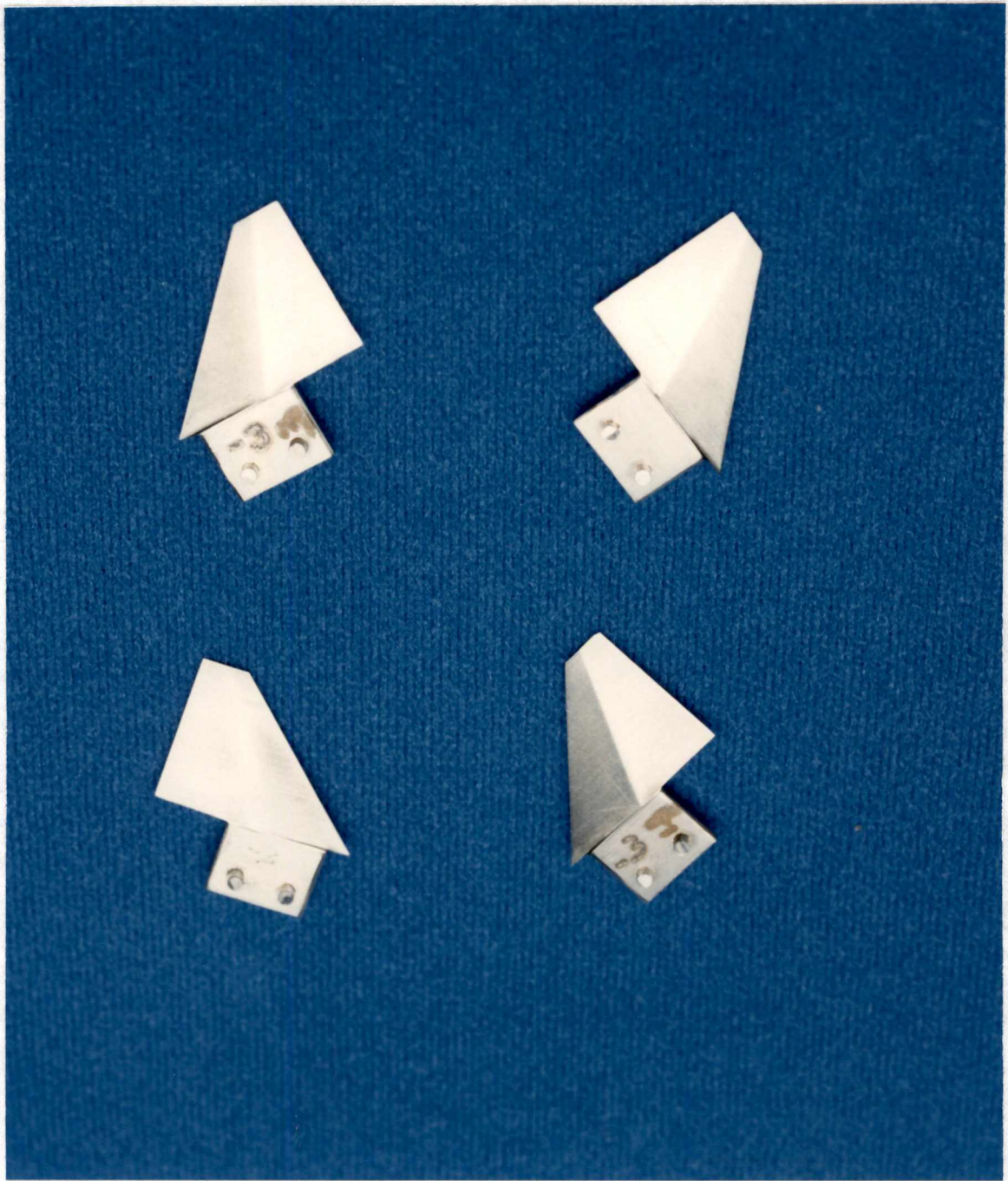


Figure B9. Horizontal and vertical fabricated FCA's #3.



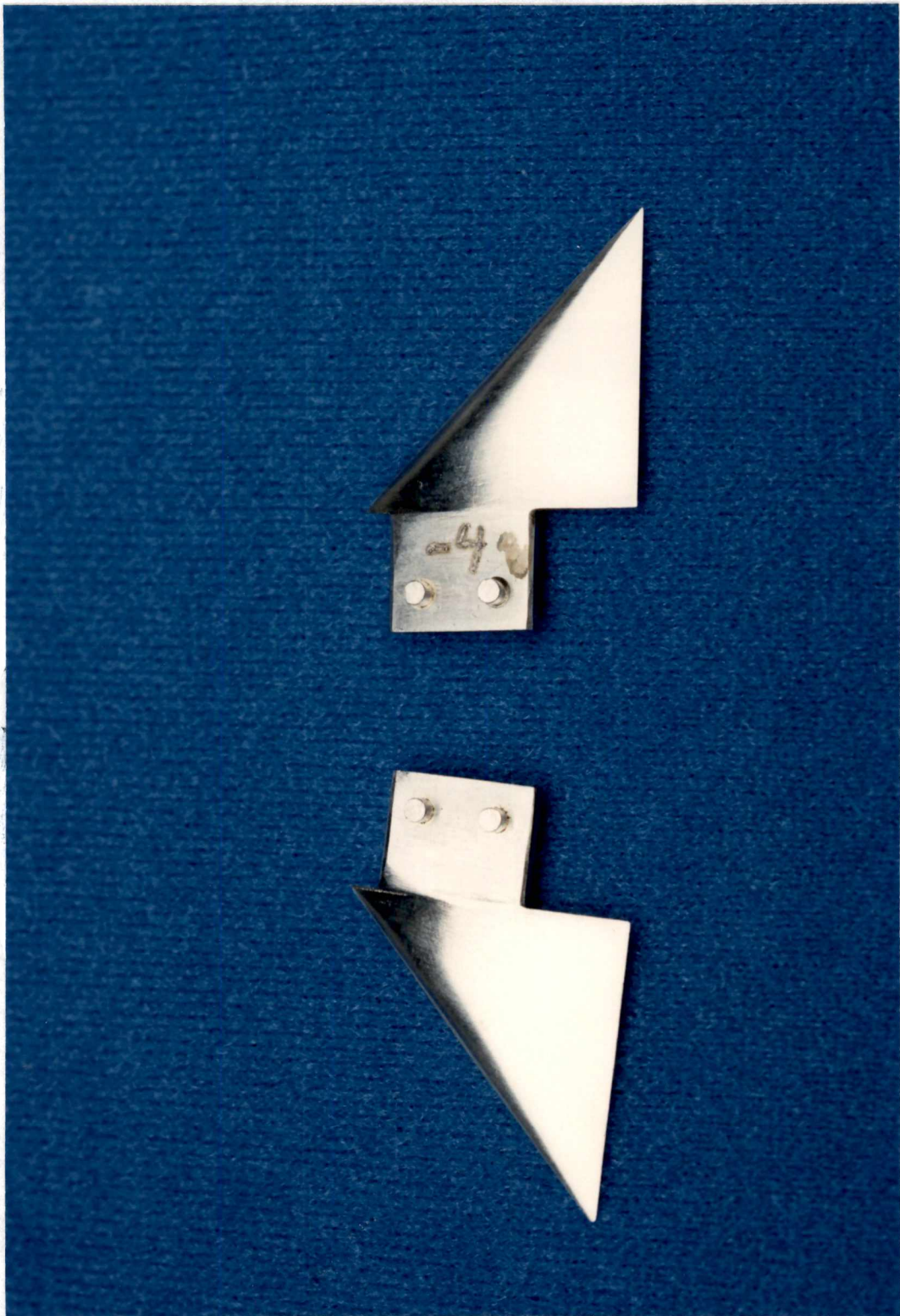


Figure B10. Fabricated FCA's #4.



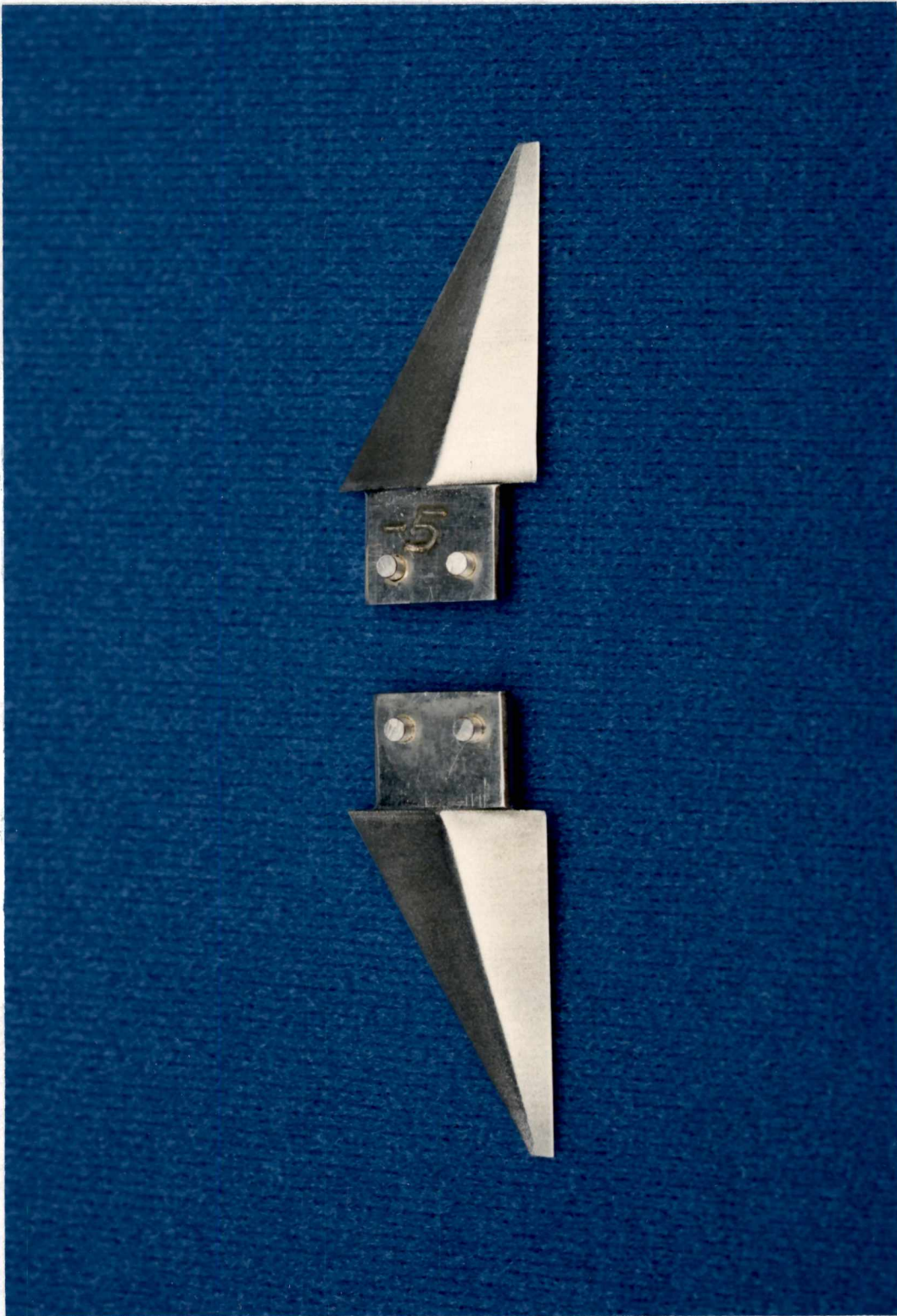


Figure B11. Fabricated FCA's #5.



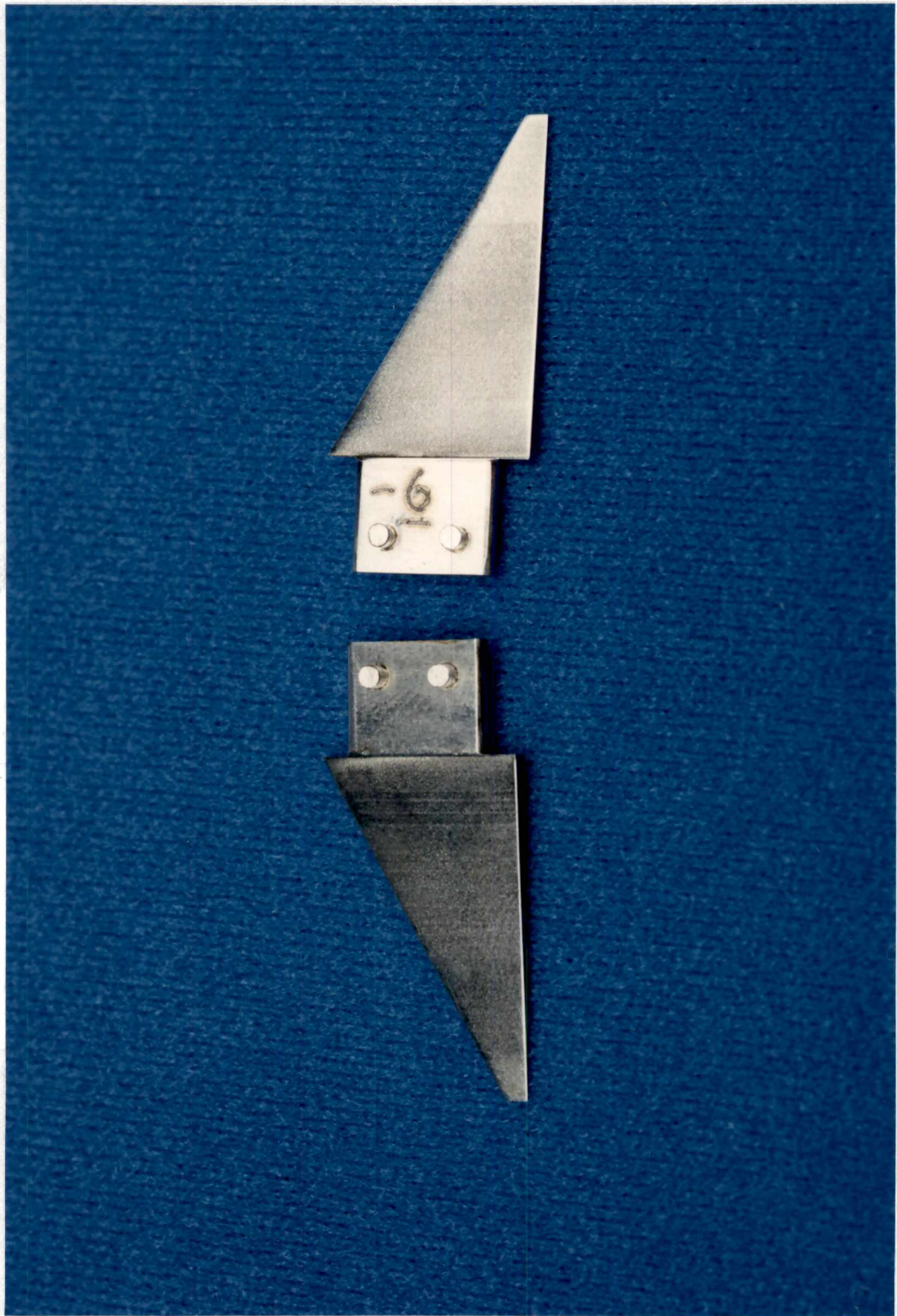


Figure B12. Fabricated FCA's #6.



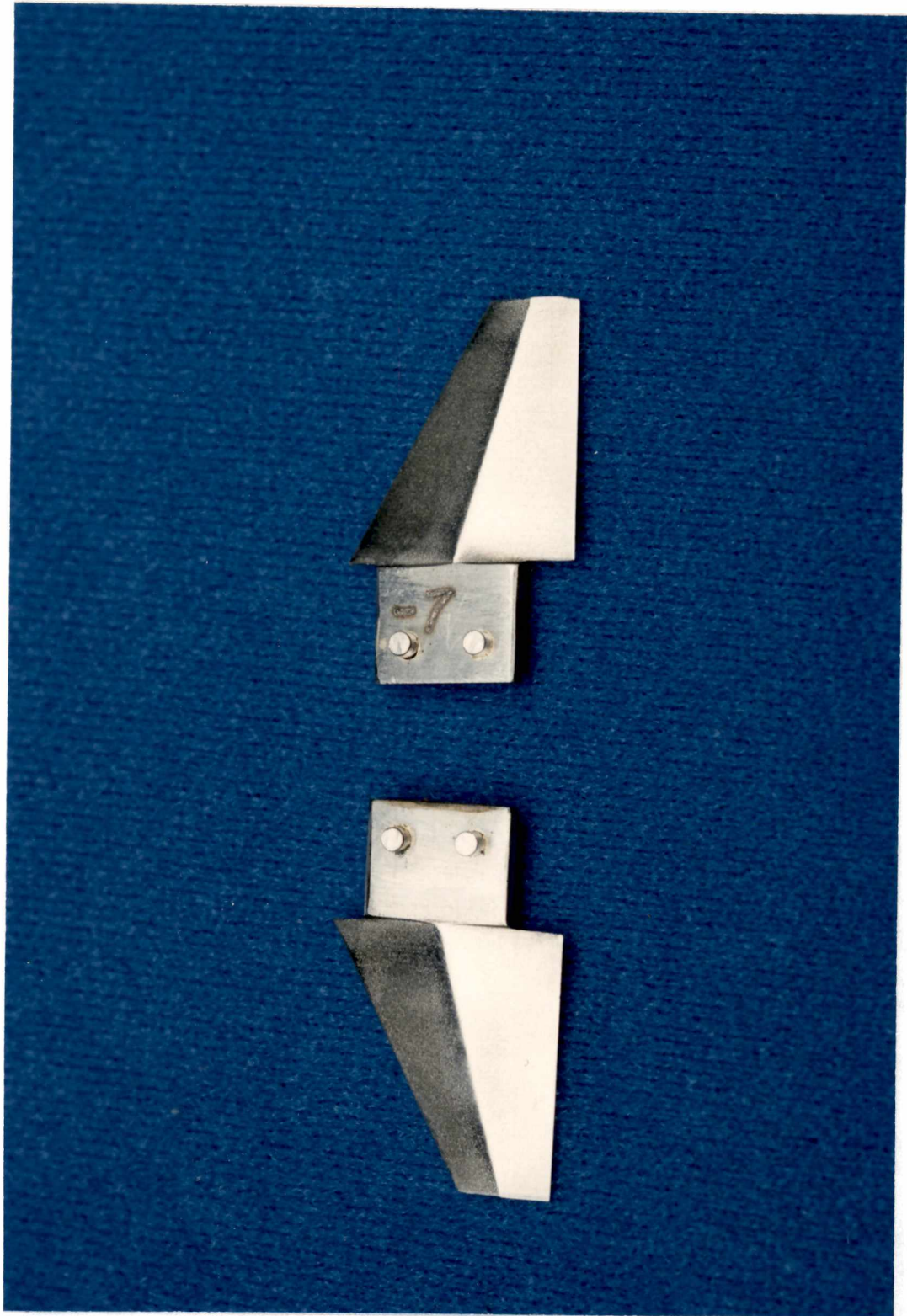


Figure B13. Fabricated FCA's #7.



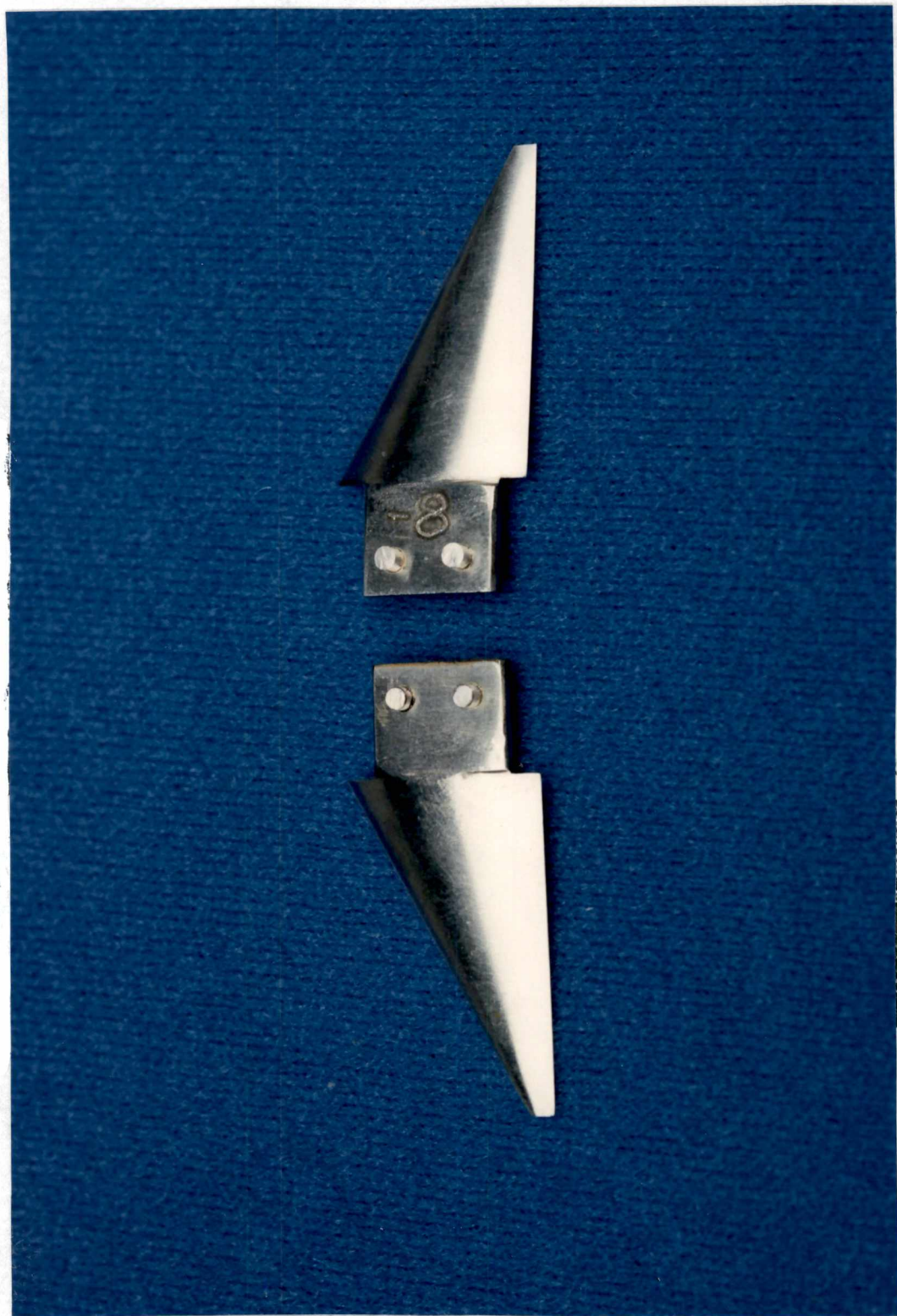


Figure B14. Fabricated FCA #8.



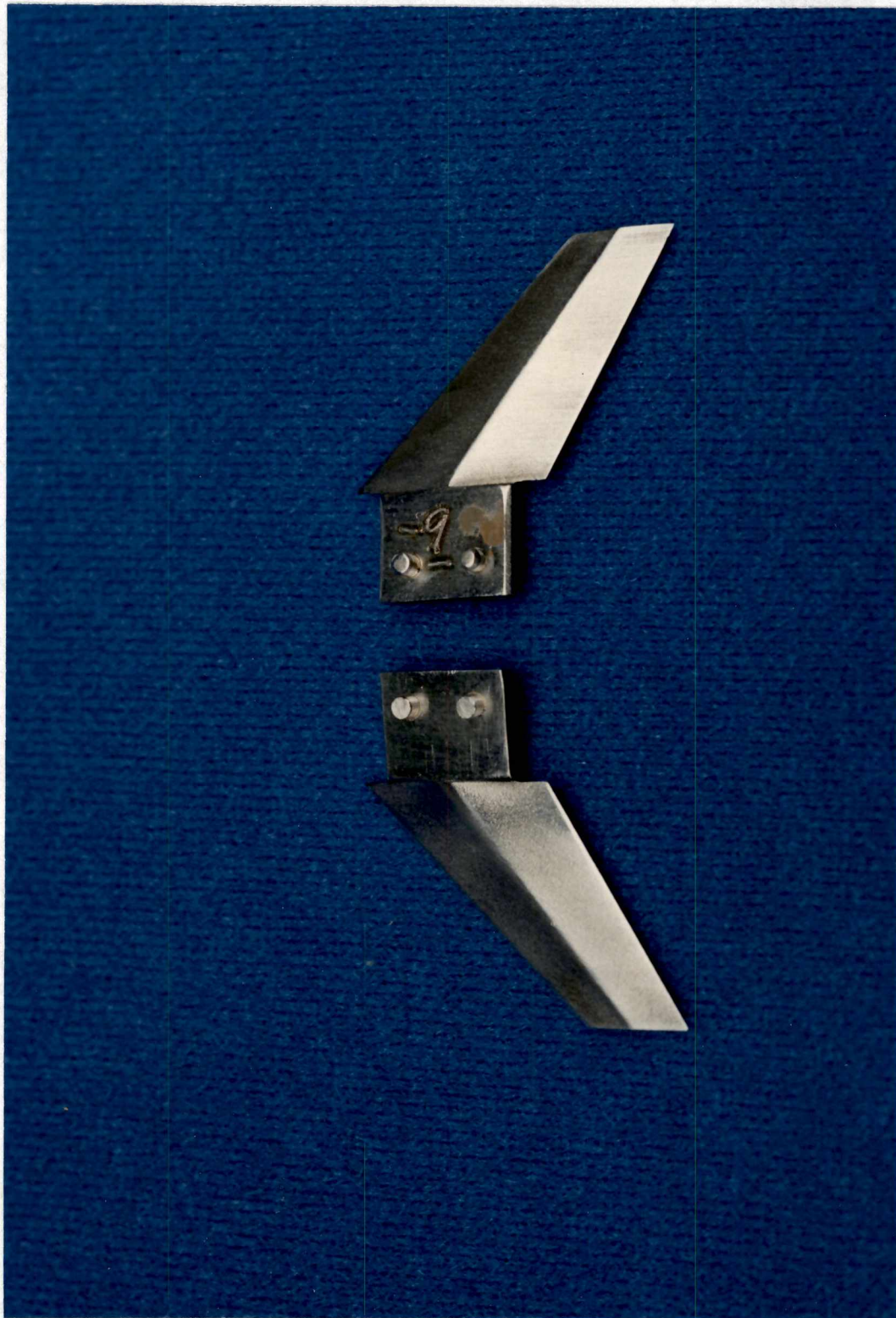


Figure B15. Horizontal fabricated FCA's #9.



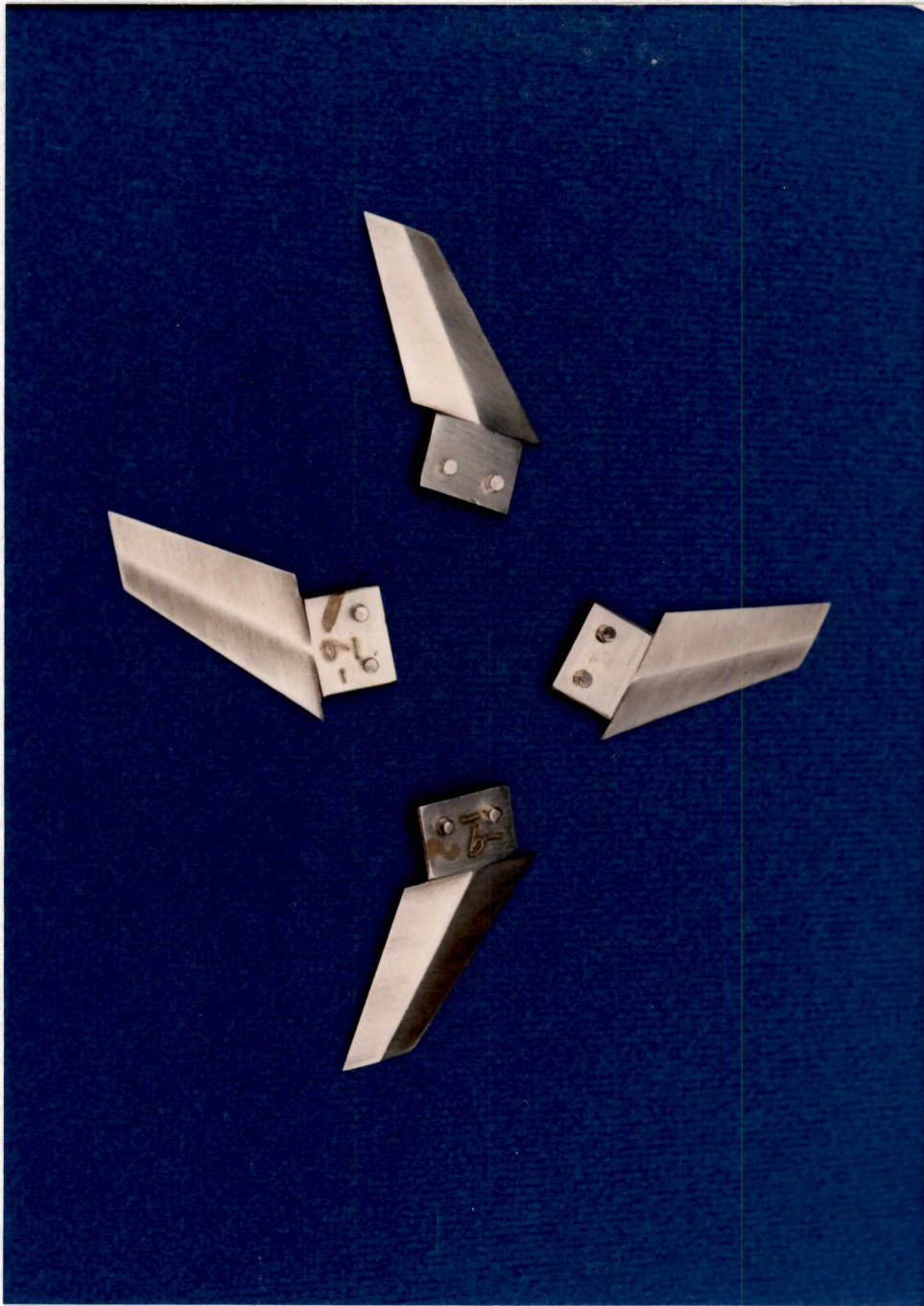


Figure B16. Horizontal and vertical fabricated FCA's #9.



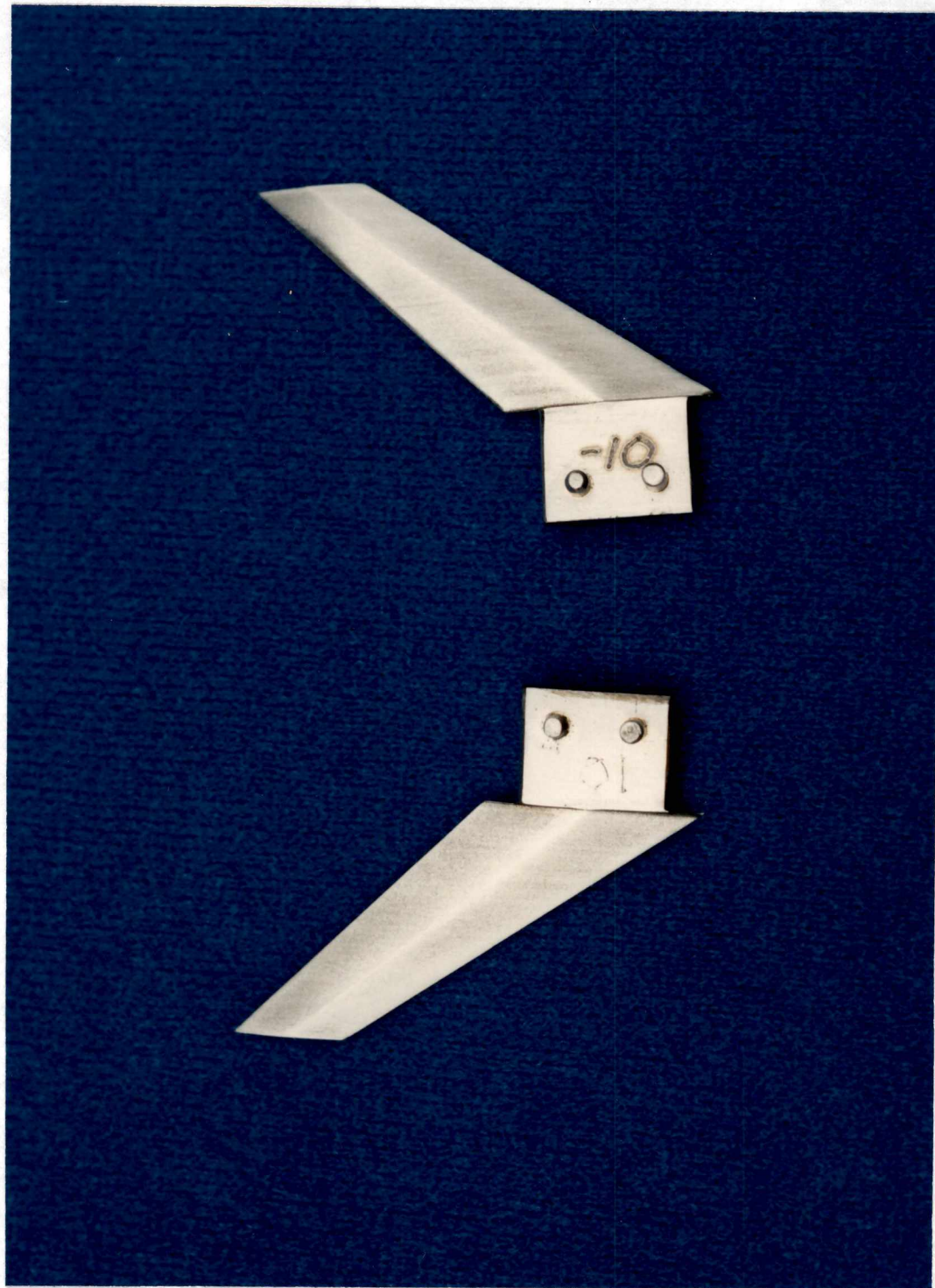


Figure B17. Fabricated FCA's #10.



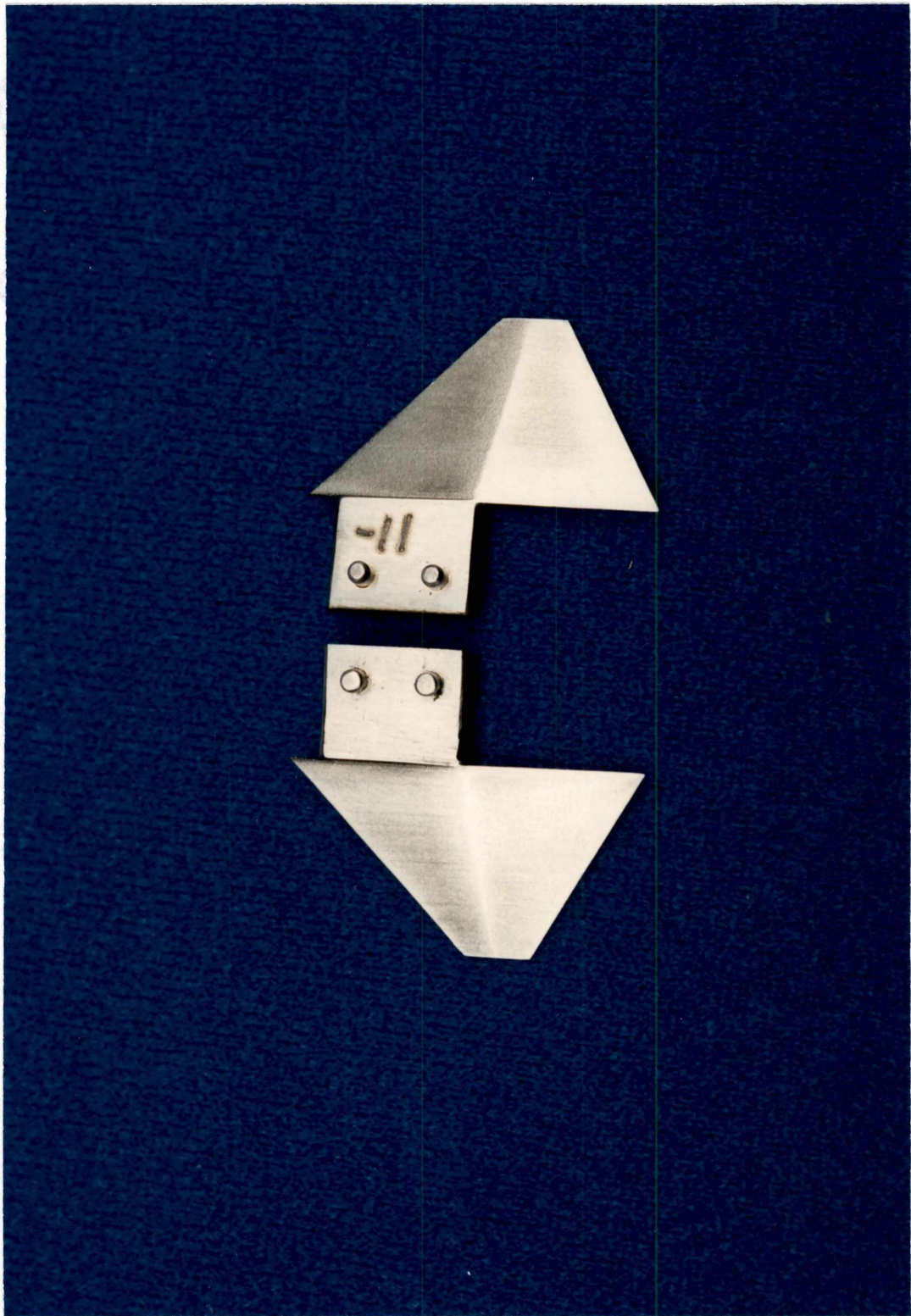


Figure B18. Fabricated FCA's #11.



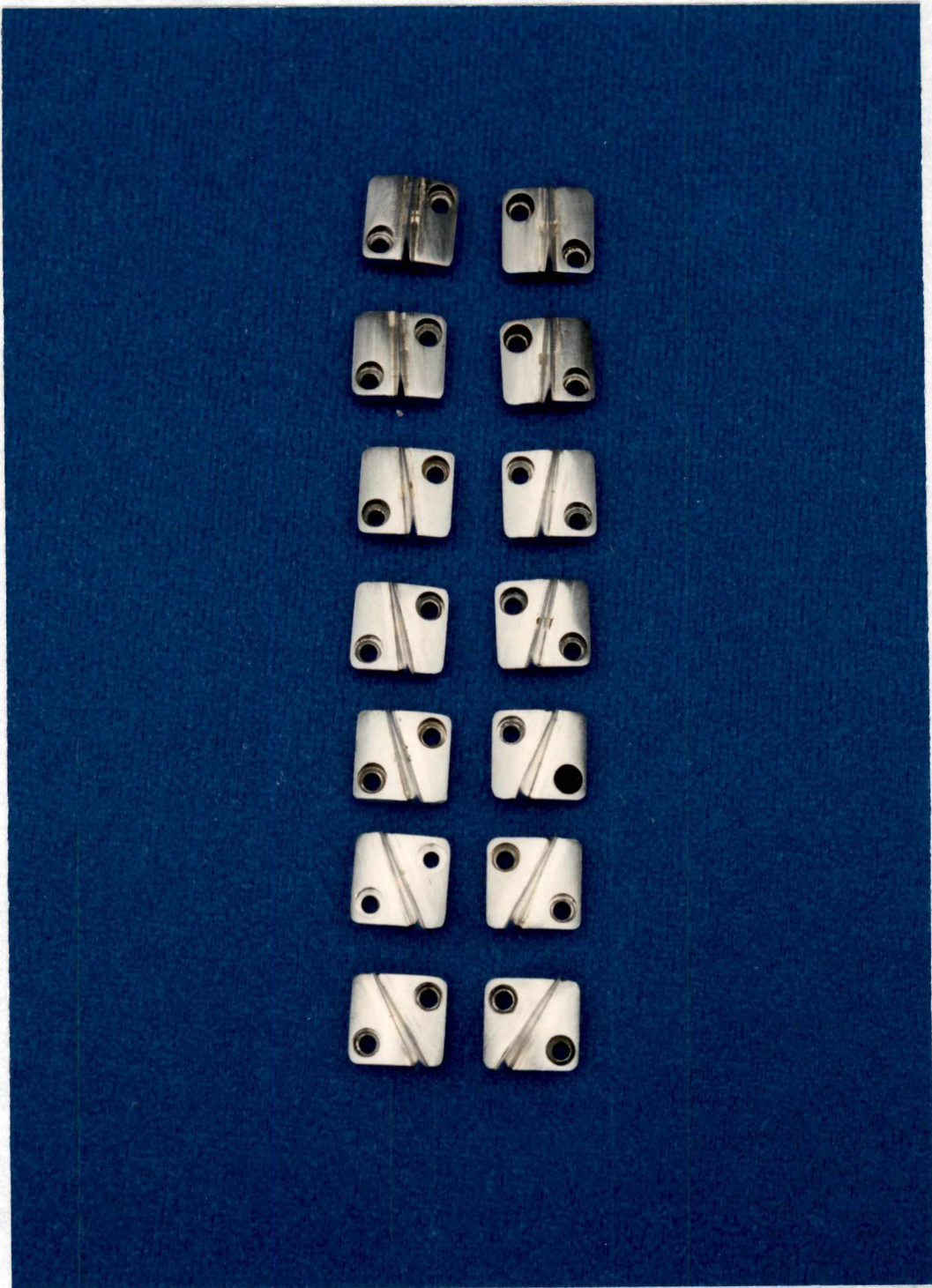


Figure B19. Fabricated canard FCA's top view.



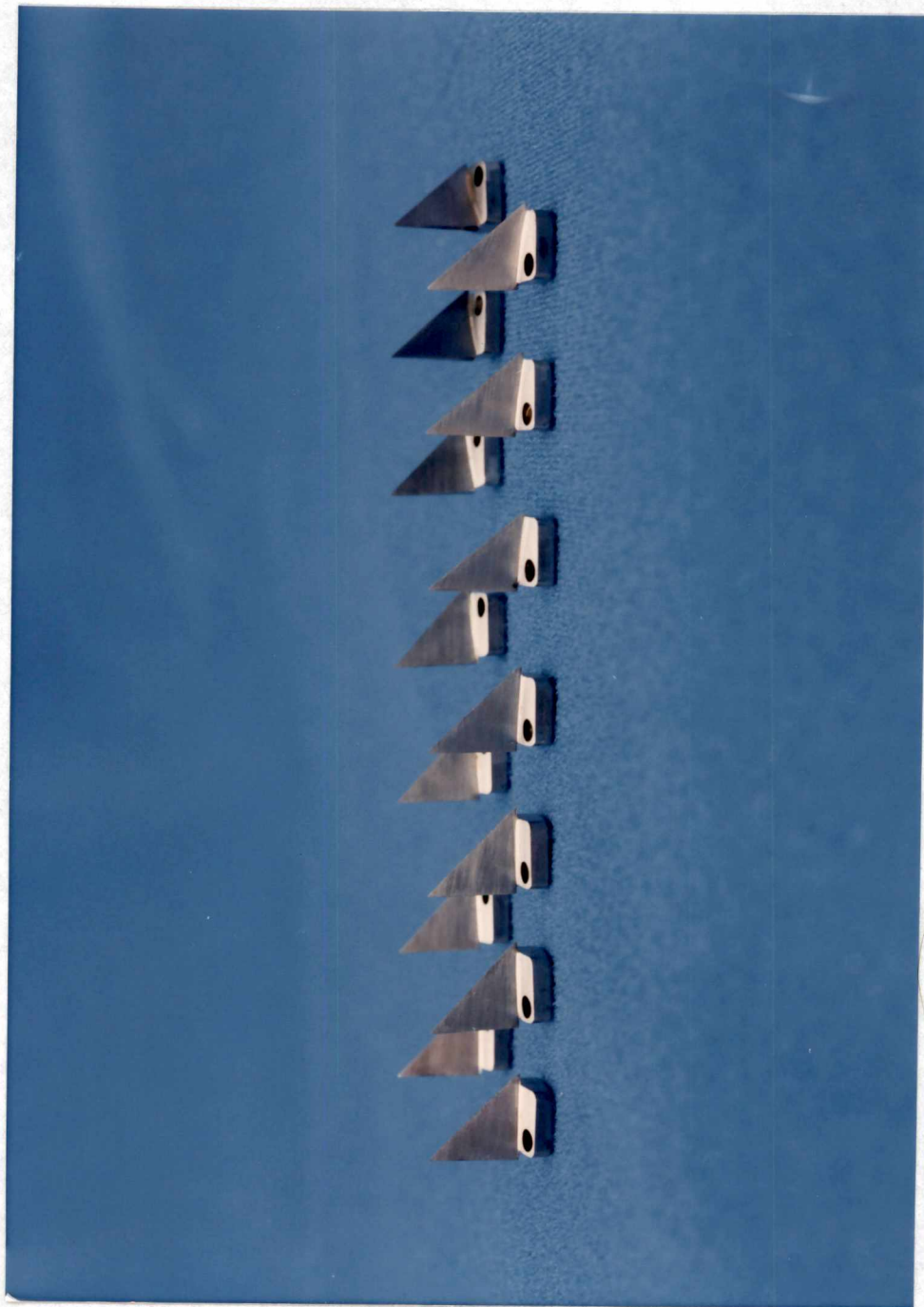


Figure B20. Fabricated canard FCA's side view.

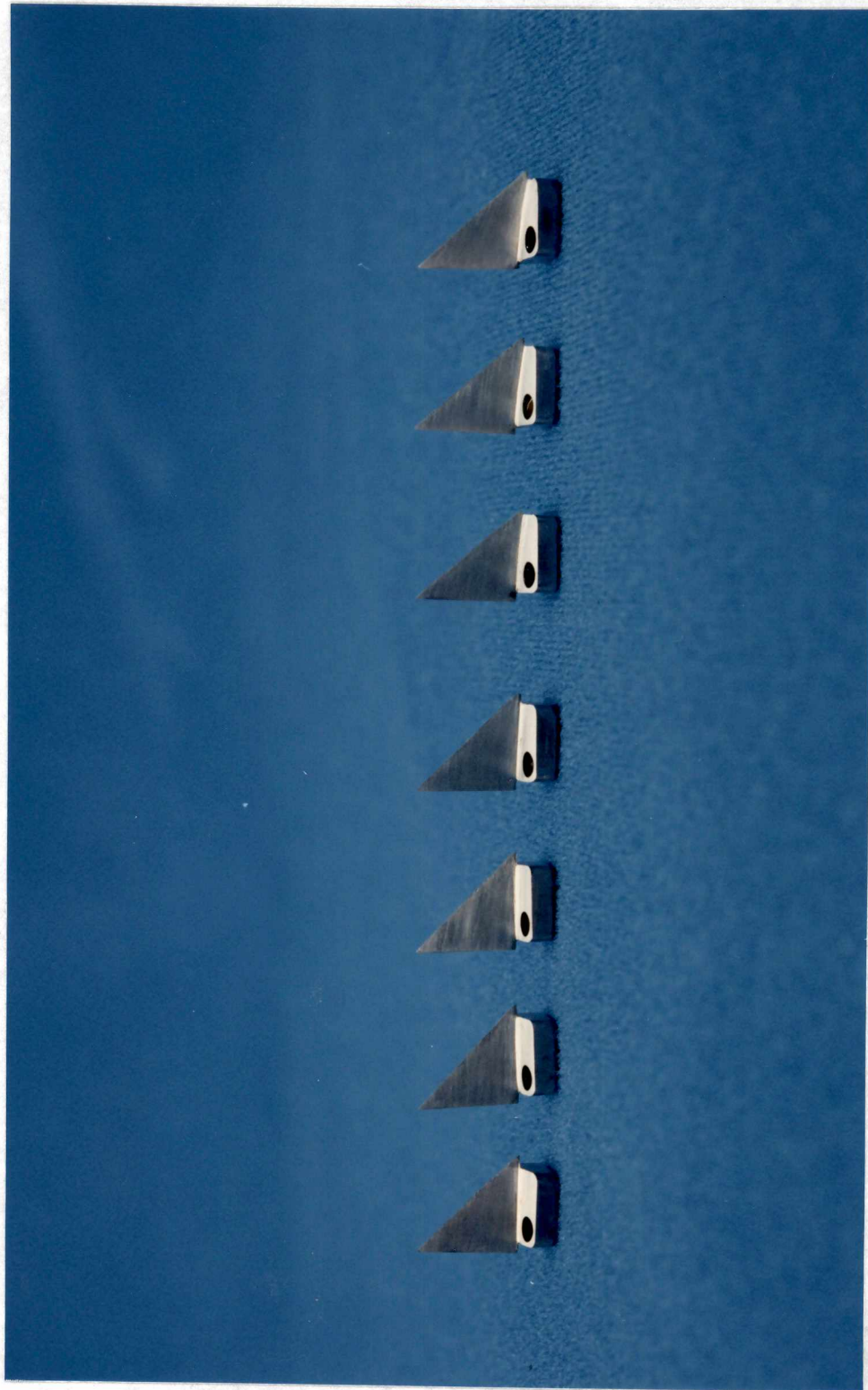


Figure B21. Close-up fabricated canard FCA's side view.

## **APPENDIX C**

### **STATIC STABILITY AND CONTROL DATA**

## **C.1 Raw Static Stability Wind Tunnel Test Data**

Table C1. Raw data for FCA #3;  $\delta = 0$ ,  $M = 0.2$ .

| CDDF Aero Control Study 10:50:30                      |                                                     |          |         |                |          |          |        |        |        |        |
|-------------------------------------------------------|-----------------------------------------------------|----------|---------|----------------|----------|----------|--------|--------|--------|--------|
| TWT0738                                               |                                                     | Run 66/0 |         | FLOW CONDITONS |          |          |        |        |        |        |
| Fr #                                                  | P-0                                                 | P-inf    | Q-inf   | M-inf          | T-0      | Rn/ft    | Sector |        |        |        |
| 1                                                     | 22.02                                               | 21.44    | .57     | .195           | 560      | 1.9E+06  | -9.46  |        |        |        |
| 2                                                     | 22.01                                               | 21.42    | .58     | .197           | 563      | 1.9E+06  | -7.38  |        |        |        |
| 3                                                     | 22.03                                               | 21.44    | .58     | .197           | 564      | 1.8E+06  | -5.42  |        |        |        |
| 4                                                     | 22.02                                               | 21.44    | .58     | .196           | 565      | 1.8E+06  | -3.44  |        |        |        |
| 5                                                     | 22.03                                               | 21.44    | .58     | .196           | 566      | 1.8E+06  | -1.44  |        |        |        |
| 6                                                     | 22.03                                               | 21.44    | .58     | .197           | 566      | 1.8E+06  | .58    |        |        |        |
| 7                                                     | 22.02                                               | 21.44    | .58     | .197           | 567      | 1.8E+06  | 2.54   |        |        |        |
| 8                                                     | 22.02                                               | 21.43    | .59     | .198           | 567      | 1.8E+06  | 4.60   |        |        |        |
| 9                                                     | 22.02                                               | 21.43    | .59     | .198           | 567      | 1.8E+06  | 6.56   |        |        |        |
| 10                                                    | 22.02                                               | 21.42    | .59     | .198           | 567      | 1.8E+06  | 8.60   |        |        |        |
| 11                                                    | 22.01                                               | 21.42    | .59     | .198           | 567      | 1.8E+06  | 9.68   |        |        |        |
| Fr #                                                  | T-inf                                               | Rho-inf  | V-inf   | W-dot          | a-0      | Time     |        |        |        |        |
| 1                                                     | 556.157                                             | .104     | 225.772 | 4.528          | 1156.109 | 10:47:58 |        |        |        |        |
| 2                                                     | 558.189                                             | .104     | 227.970 | 4.517          | 1158.219 | 10:48:02 |        |        |        |        |
| 3                                                     | 559.662                                             | .103     | 228.016 | 4.515          | 1159.746 | 10:48:05 |        |        |        |        |
| 4                                                     | 560.676                                             | .103     | 228.031 | 4.510          | 1160.796 | 10:48:08 |        |        |        |        |
| 5                                                     | 561.438                                             | .103     | 227.734 | 4.508          | 1161.584 | 10:48:11 |        |        |        |        |
| 6                                                     | 561.935                                             | .103     | 229.182 | 4.506          | 1162.098 | 10:48:14 |        |        |        |        |
| 7                                                     | 562.350                                             | .103     | 228.646 | 4.503          | 1162.527 | 10:48:17 |        |        |        |        |
| 8                                                     | 562.561                                             | .103     | 230.218 | 4.502          | 1162.746 | 10:48:20 |        |        |        |        |
| 9                                                     | 562.757                                             | .103     | 230.111 | 4.501          | 1162.948 | 10:48:23 |        |        |        |        |
| 10                                                    | 562.894                                             | .103     | 230.184 | 4.500          | 1163.089 | 10:48:26 |        |        |        |        |
| 11                                                    | 563.003                                             | .103     | 230.123 | 4.498          | 1163.202 | 10:48:30 |        |        |        |        |
| P-static offset, in percent of P-0 -.009              |                                                     |          |         |                |          |          |        |        |        |        |
| Header                                                |                                                     | Value    |         |                |          |          |        |        |        |        |
| 14 Mar 1994 CDDF Aero Control Study 10:50:38          |                                                     |          |         |                |          |          |        |        |        |        |
| Test name                                             | TWT0738                                             |          |         |                |          |          |        |        |        |        |
| Label Set                                             | LTWT0738A                                           |          |         |                |          |          |        |        |        |        |
| Engineer                                              | POKORA                                              |          |         |                |          |          |        |        |        |        |
| Contractor                                            | NONE                                                |          |         |                |          |          |        |        |        |        |
| ot frames                                             | 11                                                  | Fr #     | Alpha   | Beta           | CMF      | CMF      | CYF    | CYMF   | CBLF   | CAF    |
| Tot columns                                           | 32                                                  | 1        | -9.92   | 0.00           | -1.0380  | -1.5895  | .0000  | .0000  | .0000  | -.2075 |
| Tot headers                                           | 59                                                  | 2        | -7.84   | 0.00           | -.9325   | -1.5013  | .0000  | .0000  | .0000  | -.1463 |
| Time                                                  | 10:47:32                                            | 3        | -5.88   | 0.00           | -.7609   | -1.2063  | .0000  | .0000  | .0000  | -.1283 |
| Date                                                  | 14 Mar 1994                                         | 4        | -3.90   | 0.00           | -.4774   | -.6576   | .0000  | .0000  | .0000  | -.1405 |
| Basic Prog                                            | TWT0738A                                            | 5        | -1.90   | 0.00           | -.1739   | -.1410   | .0000  | .0000  | .0000  | -.1009 |
| Run                                                   | 66                                                  | 6        | .12     | 0.00           | .0047    | -.0146   | .0000  | .0000  | .0000  | -.1314 |
| Rerun                                                 | 0                                                   | 7        | 2.08    | 0.00           | .2371    | .2514    | .0000  | .0000  | .0000  | -.1154 |
| Config                                                | FIN 3 D=0                                           | 8        | 4.14    | 0.00           | .4465    | .5301    | .0000  | .0000  | .0000  | -.0626 |
| Pbar-mmHg                                             | 747.9                                               | 9        | 6.09    | -0.00          | .5308    | .4452    | .0000  | .0000  | .0000  | -1.341 |
| Sweep Angle                                           | ALPHA                                               | 10       | 8.14    | -0.00          | .9115    | 1.3036   | .0000  | .0000  | .0000  | -.0878 |
| Machset                                               | .2                                                  | 11       | 9.22    | -0.00          | 1.0312   | 1.5299   | .0000  | .0000  | .0000  | -.0610 |
| Alpha prime                                           | -.46                                                | Fr #     | Alpha   | Beta           | CPB01    | CPB02    | CPB03  | CPB04  | CPB    | CAB    |
| Beta prime                                            | 0                                                   | 1        | -9.92   | 0.00           | -.4768   | -.4929   | -.5426 | -.5177 | -.5075 | .8609  |
| Roll angle                                            | 0                                                   | 2        | -7.84   | 0.00           | -.4749   | -.5018   | -.5021 | -.4860 | -.4912 | .8332  |
| No. angles                                            | 11                                                  | 3        | -5.88   | 0.00           | -.4764   | -.4676   | -.4829 | -.4943 | -.4803 | .8147  |
| S,ref                                                 | 2.14                                                | 4        | -3.90   | 0.00           | -.4631   | -.4875   | -.4667 | -.5078 | -.4813 | .8163  |
| L,ref                                                 | 1.655                                               | 5        | -1.90   | 0.00           | -.4742   | -.4951   | -.4594 | -.5275 | -.4890 | .8295  |
| D,ref                                                 | 1.655                                               | 6        | .12     | 0.00           | -.4956   | -.4859   | -.4688 | -.5046 | -.4887 | .8290  |
| XTO                                                   | -4.757                                              | 7        | 2.08    | 0.00           | -.4673   | -.4904   | -.4689 | -.5246 | -.4878 | .8274  |
| Wt. tare ax                                           | 10.2                                                | 8        | 4.14    | 0.00           | -.4348   | -.5062   | -.4731 | -.5017 | -.4790 | .8125  |
| X,MRP                                                 | 3839.5                                              | 9        | 6.09    | -0.00          | -.4650   | -.4879   | -.4646 | -.5321 | -.4874 | .8267  |
| Y,MRP                                                 | 0                                                   | 10       | 8.14    | -0.00          | -.4691   | -.4775   | -.4772 | -.4849 | -.4772 | .8094  |
| Z,MRP                                                 | 0                                                   | 11       | 9.22    | -0.00          | -.5009   | -.4547   | -.4746 | -.4879 | -.4795 | .8133  |
| AB                                                    | 3.63                                                |          |         |                |          |          |        |        |        |        |
| Delta alpha/M .0317 Delta alpha/H .0131               |                                                     |          |         |                |          |          |        |        |        |        |
| Delta beta/S-.0307 Delta beta/Y-.0178 Delta roll/RH 0 |                                                     |          |         |                |          |          |        |        |        |        |
| S,ref                                                 | Ref. area, sq.in. 2.14                              |          |         |                |          |          |        |        |        |        |
| L,ref                                                 | Ref. length, in. 1.655                              |          |         |                |          |          |        |        |        |        |
| D,ref                                                 | Body diameter, in. 1.655                            |          |         |                |          |          |        |        |        |        |
| XTO                                                   | XFR dist., in. fwd. BMC = + -4.757                  |          |         |                |          |          |        |        |        |        |
| Wt. tare ax                                           | Weight tare axial, raw uv/deg 10.2                  |          |         |                |          |          |        |        |        |        |
| X,MRP                                                 | XMRP location, full scale inches, for record only   |          |         |                |          |          |        |        |        |        |
| 3839.5                                                |                                                     |          |         |                |          |          |        |        |        |        |
| Y,MRP                                                 | YMRP location, full scale inches, for record only 0 |          |         |                |          |          |        |        |        |        |
| Z,MRP                                                 | ZMRP location, full scale inches, for record only 0 |          |         |                |          |          |        |        |        |        |
| AB                                                    | base area sq. in. 3.63                              |          |         |                |          |          |        |        |        |        |

Table C2. Positive  $\alpha$  sweep raw data for FCA #4;  $\delta = 15^\circ$ ,  $M = 4.96$ .

CDDF AERO CONTROL STUDY 12:43:30

|         |           |                |       |       |     |         |        |
|---------|-----------|----------------|-------|-------|-----|---------|--------|
| TWT0738 | Run 570/1 | FLOW CONDITONS |       |       |     |         |        |
| Fr #    | P-0       | P-inf          | Q-inf | M-inf | T-0 | Rn/ft   | Sector |
| 1       | 80.08     | .16            | 2.73  | 4.960 | 588 | 4.3E+06 | .54    |
| 2       | 80.07     | .16            | 2.73  | 4.960 | 590 | 4.3E+06 | 2.56   |
| 3       | 80.10     | .16            | 2.73  | 4.960 | 593 | 4.2E+06 | 4.56   |
| 4       | 80.08     | .16            | 2.73  | 4.960 | 595 | 4.2E+06 | 6.56   |
| 5       | 80.07     | .16            | 2.73  | 4.960 | 597 | 4.2E+06 | 8.54   |
| 6       | 80.09     | .16            | 2.73  | 4.960 | 599 | 4.2E+06 | 9.86   |

|      |         |         |          |        |         |          |  |
|------|---------|---------|----------|--------|---------|----------|--|
| Fr # | T-inf   | Rho-inf | V-inf    | W-dot  | a-0     | Time     |  |
| 1    | 99.388  | .004    | 2424.091 | 16.070 | 488.728 | 12:41:59 |  |
| 2    | 99.731  | .004    | 2428.268 | 16.041 | 489.570 | 12:42:01 |  |
| 3    | 100.152 | .004    | 2433.384 | 16.012 | 490.602 | 12:42:03 |  |
| 4    | 100.480 | .004    | 2437.375 | 15.983 | 491.406 | 12:42:06 |  |
| 5    | 100.852 | .004    | 2441.881 | 15.951 | 492.315 | 12:42:08 |  |
| 6    | 101.192 | .004    | 2445.988 | 15.928 | 493.143 | 12:42:10 |  |

P-static offset, in percent of P-0 0

|             |            |
|-------------|------------|
| Header      | Value      |
| Test name   | TWT0738    |
| Label Set   | LWT0738A   |
| Engineer    | POKORA     |
| Contractor  | NONE       |
| Tot frames  | 6          |
| Tot columns | 32         |
| Tot headers | 59         |
| Time        | 12:41:21   |
| Date        | 7 Jul 1994 |
| Basic Prog  | TWT0738A   |
| Run         | 570        |
| Rerun       | 1          |
| Config      | FIN 4 D=15 |
| PCar-mmHg.  | 745.4      |
| Sweep Angle | ALPHA      |
| Machset     | 4.96       |
| Alpha prime | -0.54      |
| Beta prime  | 0          |
| Roll angle  | 0          |
| No. angles  | 6          |
| S,ref       | 2.14       |
| L,ref       | 1.655      |
| D,ref       | 1.655      |
| XTO         | -4.757     |
| Wt. tare ax | 10.2       |
| X,MRP       | 3839.5     |
| Y,MRP       | 0          |
| Z,MRP       | 0          |
| AB          | 3.63       |

|      |       |      |        |        |       |       |       |       |
|------|-------|------|--------|--------|-------|-------|-------|-------|
| Fr # | Alpha | Beta | CMF    | CMF    | CTF   | CTNF  | CLF   | CAF   |
| 1    | -0.03 | .01  | .0159  | -.1920 | .0000 | .0000 | .0000 | .3432 |
| 2    | 2.00  | .01  | .2087  | .2118  | .0000 | .0000 | .0000 | .3499 |
| 3    | 4.02  | 0.00 | .4012  | .6341  | .0000 | .0000 | .0000 | .3798 |
| 4    | 6.03  | 0.00 | .6188  | 1.0371 | .0000 | .0000 | .0000 | .3922 |
| 5    | 8.02  | .01  | .8883  | 1.5342 | .0000 | .0000 | .0000 | .4104 |
| 6    | 9.35  | 0.00 | 1.0738 | 1.8732 | .0000 | .0000 | .0000 | .4256 |

|      |       |      |        |        |        |        |        |       |
|------|-------|------|--------|--------|--------|--------|--------|-------|
| Fr # | Alpha | Beta | CP801  | CP802  | CP803  | CP804  | CP8    | CAS   |
| 1    | -0.03 | .01  | -.0598 | -.0641 | -.0646 | -.0634 | -.0630 | .1068 |
| 2    | 2.00  | .01  | -.0621 | -.0671 | -.0689 | -.0672 | -.0656 | .1113 |
| 3    | 4.02  | 0.00 | -.0638 | -.0678 | -.0696 | -.0659 | -.0668 | .1132 |
| 4    | 6.03  | 0.00 | -.0627 | -.0683 | -.0698 | -.0667 | -.0669 | .1134 |
| 5    | 8.02  | .01  | -.0652 | -.0681 | -.0696 | -.0668 | -.0671 | .1144 |
| 6    | 9.35  | 0.00 | -.0648 | -.0703 | -.0700 | -.0677 | -.0682 | .1156 |

Delta alpha/M .0339 Delta alpha/M .0157  
Delta beta/S-.0344 Delta beta/Y-.0195 Delta roll/RM 0

|             |                                                   |        |
|-------------|---------------------------------------------------|--------|
| S,ref       | Ref. area, sq.in.                                 | 2.14   |
| L,ref       | Ref. length, in.                                  | 1.655  |
| D,ref       | Body diameter, in.                                | 1.655  |
| XTO         | XFR dist., in. fwd. BMC = +                       | -4.757 |
| Wt. tare ax | Weight tare axial, raw uv/deg                     | 10.2   |
| X,MRP       | XMRP location, full scale inches, for record only |        |
| 3839.5      |                                                   |        |
| Y,MRP       | YMRP location, full scale inches, for record only | 0      |
| Z,MRP       | ZMRP location, full scale inches, for record only | 0      |
| AB          | base area sq. in.                                 | 3.63   |

Table C3. Negative  $\alpha$  sweep raw data for FCA #4;  $\delta = 15^\circ$ ,  $M = 4.96$ .

| COOF AERO CONTROL STUDY 12:29:10                      |         |                                                     |          |                 |         |          |        |
|-------------------------------------------------------|---------|-----------------------------------------------------|----------|-----------------|---------|----------|--------|
| TWT0738                                               |         | Run 570/0                                           |          | FLOW CONDITIONS |         |          |        |
| Fr #                                                  | P-0     | P-inf                                               | Q-inf    | M-inf           | T-0     | Rn/ft    | Sector |
| 1                                                     | 80.10   | .16                                                 | 2.73     | 4.960           | 589     | 4.3E+06  | -9.54  |
| 2                                                     | 80.07   | .16                                                 | 2.73     | 4.960           | 591     | 4.3E+06  | -7.46  |
| 3                                                     | 80.09   | .16                                                 | 2.73     | 4.960           | 593     | 4.2E+06  | -5.42  |
| 4                                                     | 80.08   | .16                                                 | 2.73     | 4.960           | 596     | 4.2E+06  | -3.48  |
| 5                                                     | 80.07   | .16                                                 | 2.73     | 4.960           | 597     | 4.2E+06  | -1.44  |
| 6                                                     | 80.07   | .16                                                 | 2.73     | 4.960           | 598     | 4.2E+06  | .52    |
| Fr #                                                  | T-inf   | Rho-inf                                             | V-inf    | W-dot           | a-0     | Time     |        |
| 1                                                     | 99.564  | .004                                                | 2426.229 | 16.061          | 489.159 | 12:27:45 |        |
| 2                                                     | 99.871  | .004                                                | 2429.972 | 16.029          | 489.914 | 12:27:47 |        |
| 3                                                     | 100.196 | .004                                                | 2433.921 | 16.008          | 490.710 | 12:27:49 |        |
| 4                                                     | 100.662 | .004                                                | 2439.578 | 15.968          | 491.850 | 12:27:52 |        |
| 5                                                     | 100.858 | .004                                                | 2441.946 | 15.951          | 492.328 | 12:27:54 |        |
| 6                                                     | 100.933 | .004                                                | 2442.855 | 15.946          | 492.511 | 12:27:56 |        |
| P-static offset, in percent of P-0 0                  |         |                                                     |          |                 |         |          |        |
| Header                                                |         | Value                                               |          |                 |         |          |        |
| Test name                                             |         | TWT0738                                             |          |                 |         |          |        |
| Label Set                                             |         | LTWT0738A                                           |          |                 |         |          |        |
| Engineer                                              |         | POKORA                                              |          |                 |         |          |        |
| Contractor                                            |         | NONE                                                |          |                 |         |          |        |
| Tot frames                                            |         | 6                                                   |          |                 |         |          |        |
| Tot columns                                           |         | 32                                                  |          |                 |         |          |        |
| Tot headers                                           |         | 59                                                  |          |                 |         |          |        |
| Time                                                  |         | 12:25:39                                            |          |                 |         |          |        |
| Date                                                  |         | 7 Jul 1994                                          |          |                 |         |          |        |
| Basic Prog                                            |         | TWT0738A                                            |          |                 |         |          |        |
| Run                                                   |         | 570                                                 |          |                 |         |          |        |
| Rerun                                                 |         | 0                                                   |          |                 |         |          |        |
| Config                                                |         | FIN 4 D=15                                          |          |                 |         |          |        |
| Pbar-mmHg.                                            |         | 745.4                                               |          |                 |         |          |        |
| Sweep Angle                                           |         | ALPHA                                               |          |                 |         |          |        |
| Machset                                               |         | 4.96                                                |          |                 |         |          |        |
| Alpha prime                                           |         | -1.54                                               |          |                 |         |          |        |
| Beta prime                                            |         | 0                                                   |          |                 |         |          |        |
| Roll angle                                            |         | 0                                                   |          |                 |         |          |        |
| No. angles                                            |         | 6                                                   |          |                 |         |          |        |
| S,ref                                                 |         | 2.14                                                |          |                 |         |          |        |
| L,ref                                                 |         | 1.655                                               |          |                 |         |          |        |
| D,ref                                                 |         | 1.655                                               |          |                 |         |          |        |
| XTO                                                   |         | -4.757                                              |          |                 |         |          |        |
| Wt. tare ax                                           |         | 10.2                                                |          |                 |         |          |        |
| X,MRP                                                 |         | 3839.5                                              |          |                 |         |          |        |
| Y,MRP                                                 |         | 0                                                   |          |                 |         |          |        |
| Z,MRP                                                 |         | 0                                                   |          |                 |         |          |        |
| AB                                                    |         | 3.63                                                |          |                 |         |          |        |
| Delta alpha/W .0339 Delta alpha/M .0157               |         |                                                     |          |                 |         |          |        |
| Delta beta/S-.0344 Delta beta/Y-.0195 Delta roll/RM 0 |         |                                                     |          |                 |         |          |        |
| S,ref                                                 |         | Ref. area, sq.in. 2.14                              |          |                 |         |          |        |
| L,ref                                                 |         | Ref. length, in. 1.655                              |          |                 |         |          |        |
| D,ref                                                 |         | Body diameter, in. 1.655                            |          |                 |         |          |        |
| XTO                                                   |         | XFR dist., in. fwd. BMC = + -4.757                  |          |                 |         |          |        |
| Wt. tare ax                                           |         | Weight tare axial, raw uv/deg 10.2                  |          |                 |         |          |        |
| X,MRP                                                 |         | XMRP location, full scale inches, for record only   |          |                 |         |          |        |
| 3839.5                                                |         |                                                     |          |                 |         |          |        |
| Y,MRP                                                 |         | YMRP location, full scale inches, for record only 0 |          |                 |         |          |        |
| Z,MRP                                                 |         | ZMRP location, full scale inches, for record only 0 |          |                 |         |          |        |
| AB                                                    |         | base area sq. in. 3.63                              |          |                 |         |          |        |

## **C.2 Stability Parameter Plots for Deflected Fore and Aft FCA's**



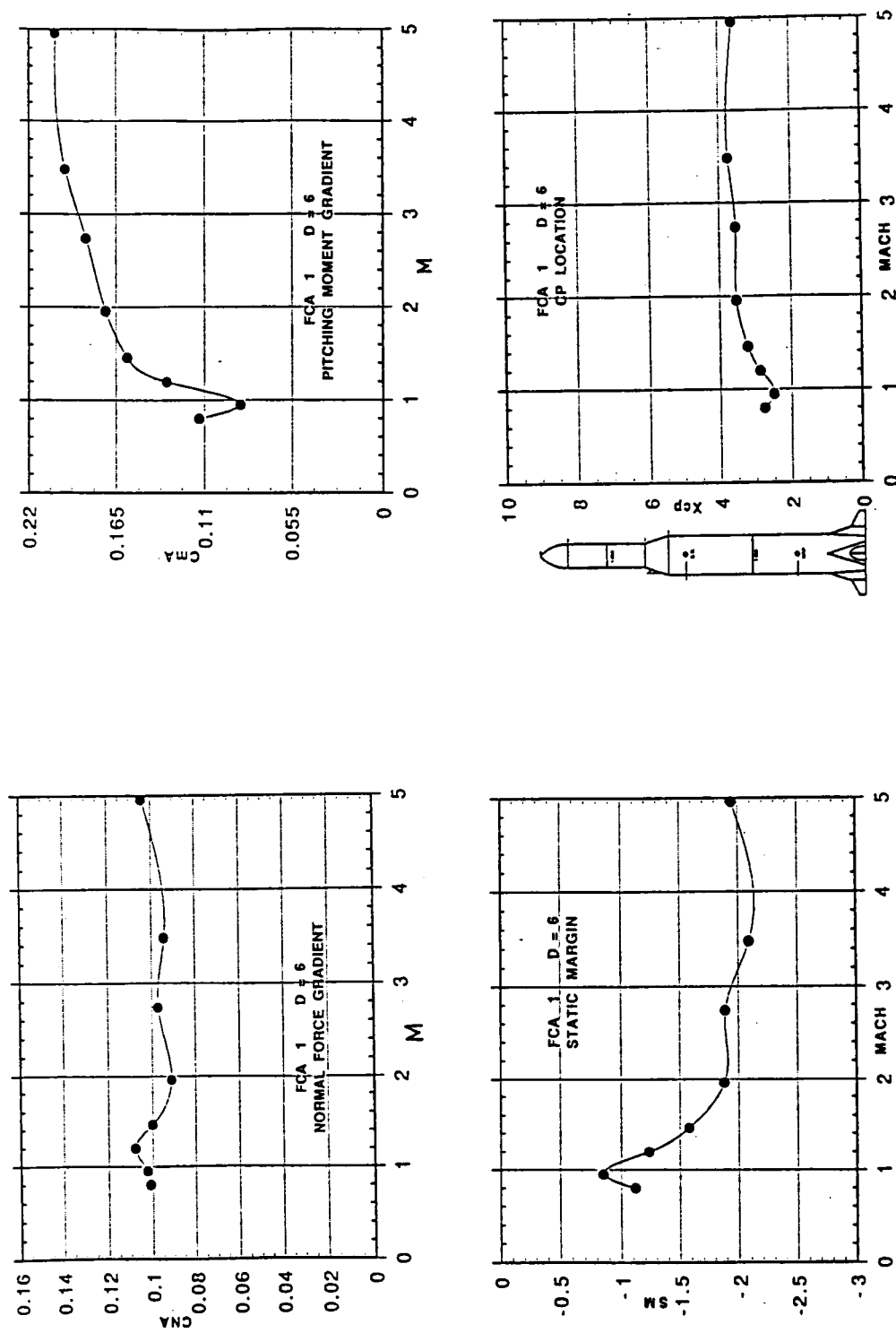


Figure C1 . Results for FCA 1;  $\delta = 6^\circ$ .

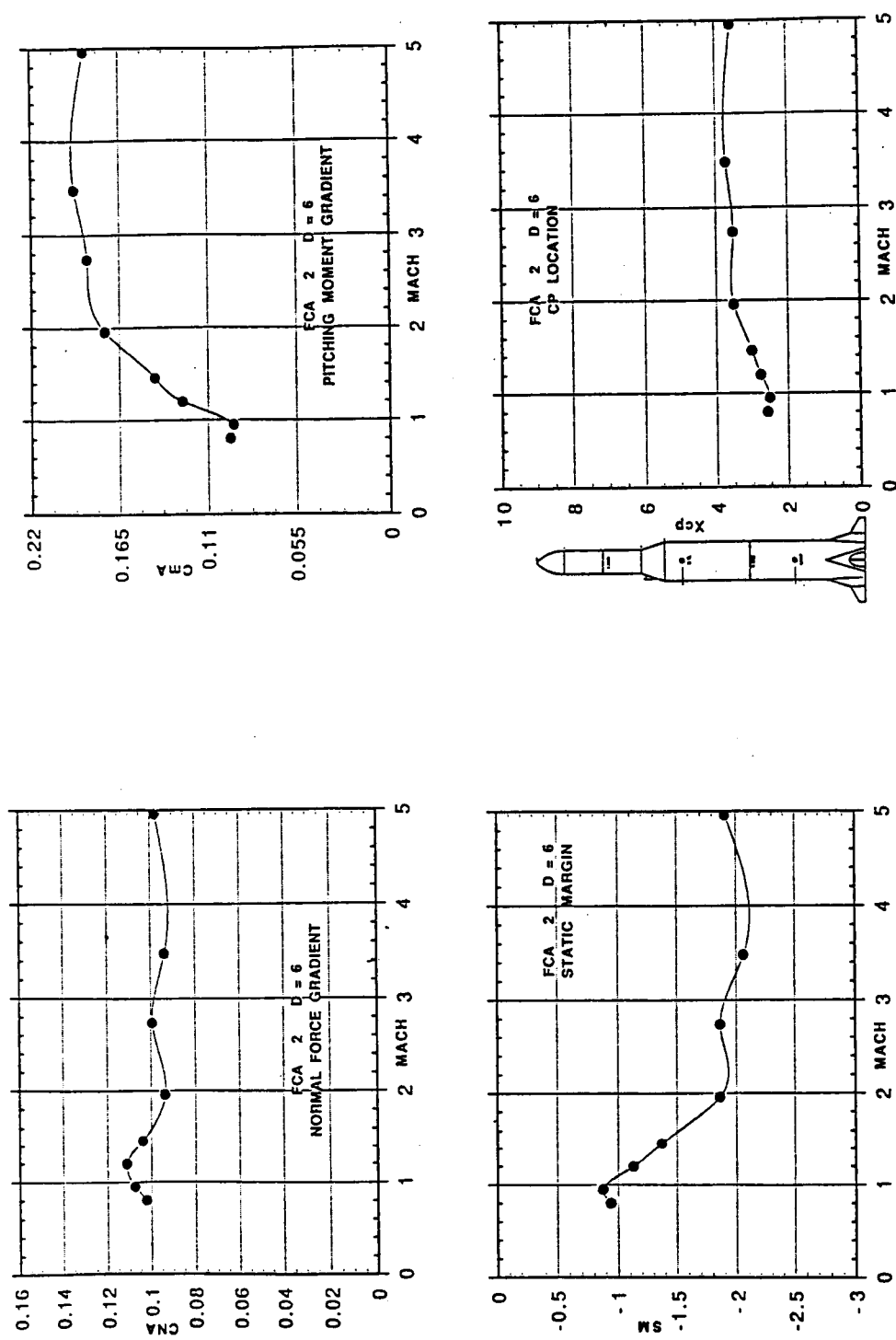


Figure C2. Results for FCA 2;  $\delta = 6^\circ$ .

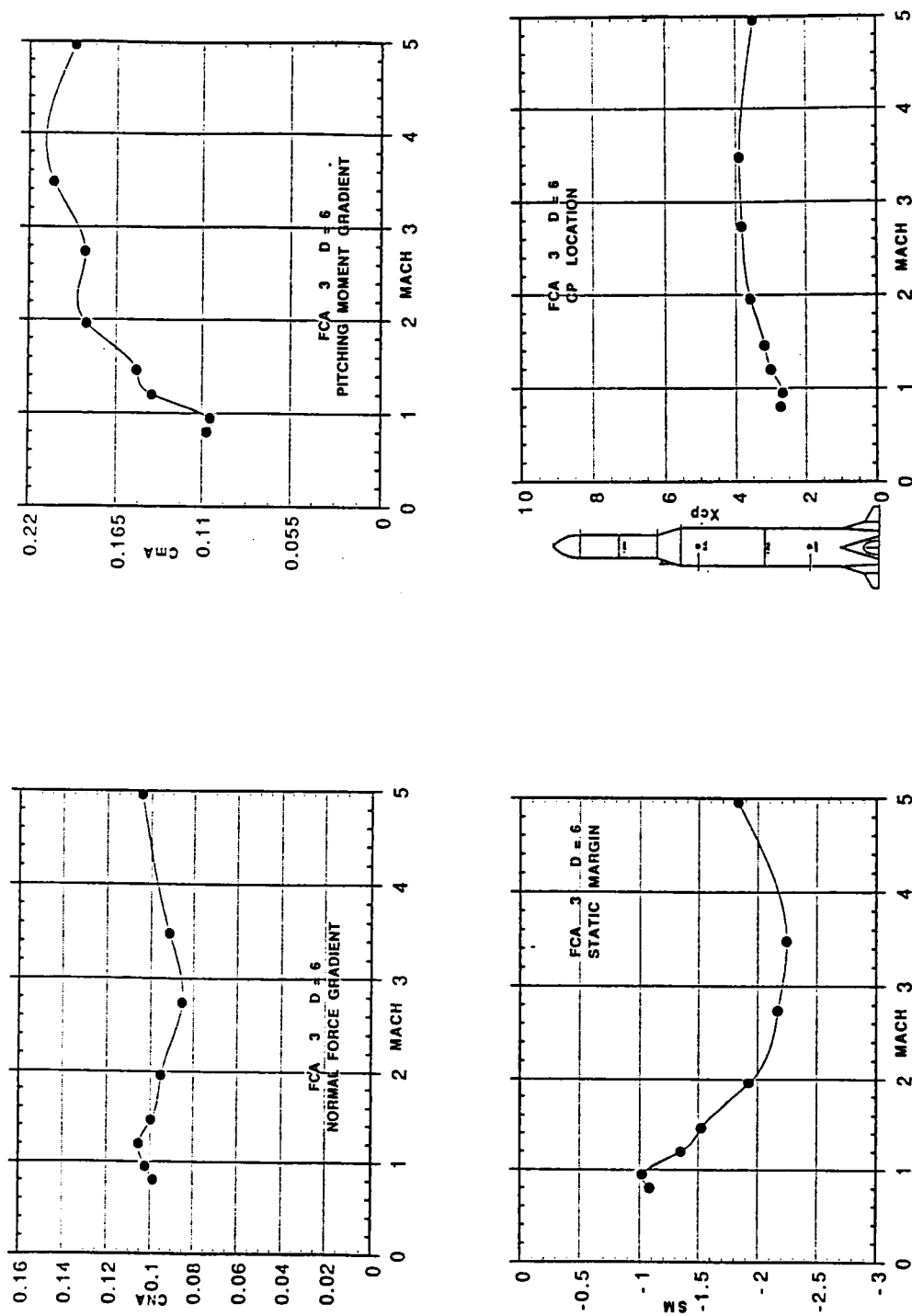


Figure C3. Results for FCA 3;  $\delta = 6^\circ$ .

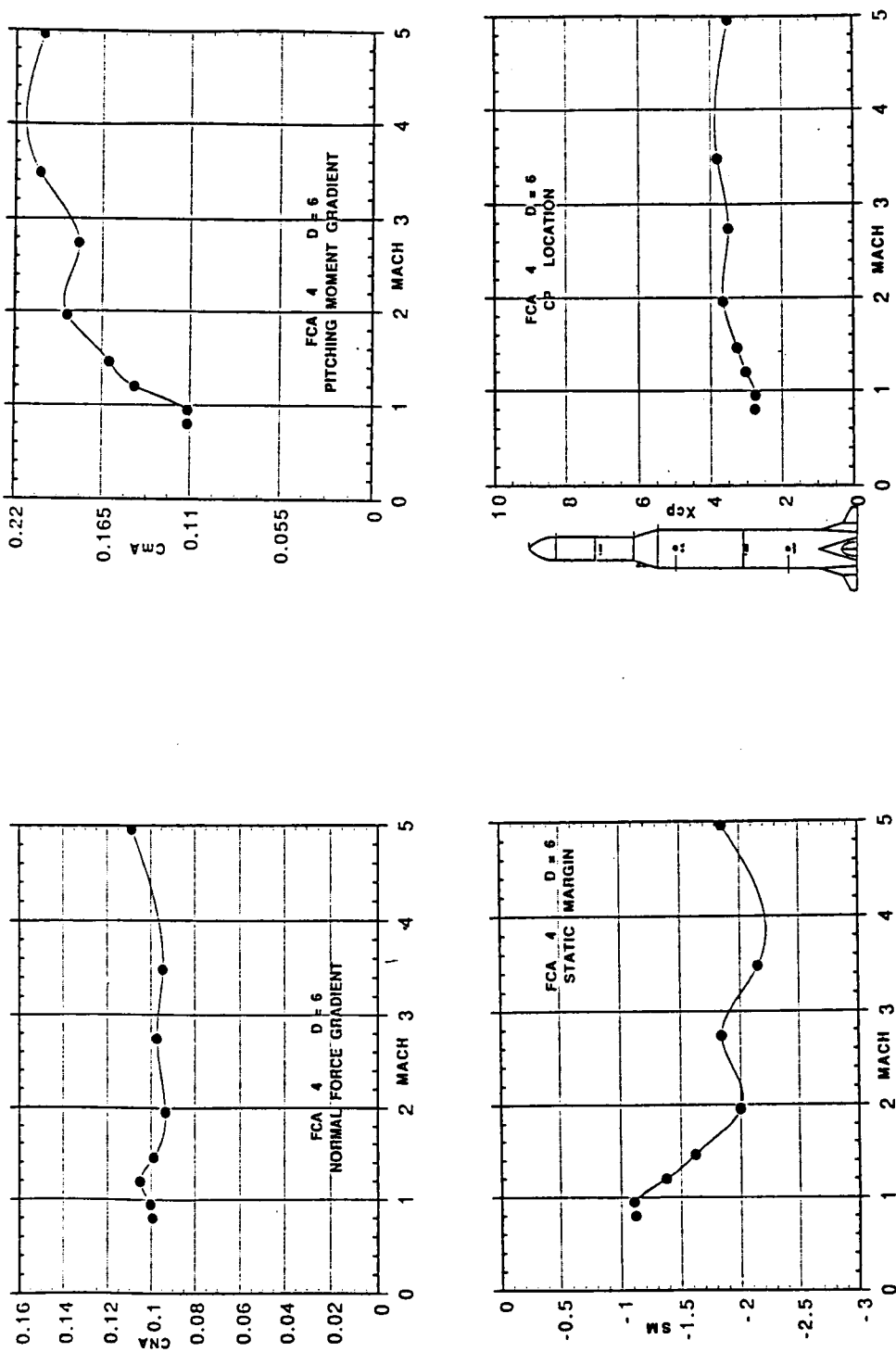


Figure C4. Results for FCA 4;  $\delta = 6^\circ$ .

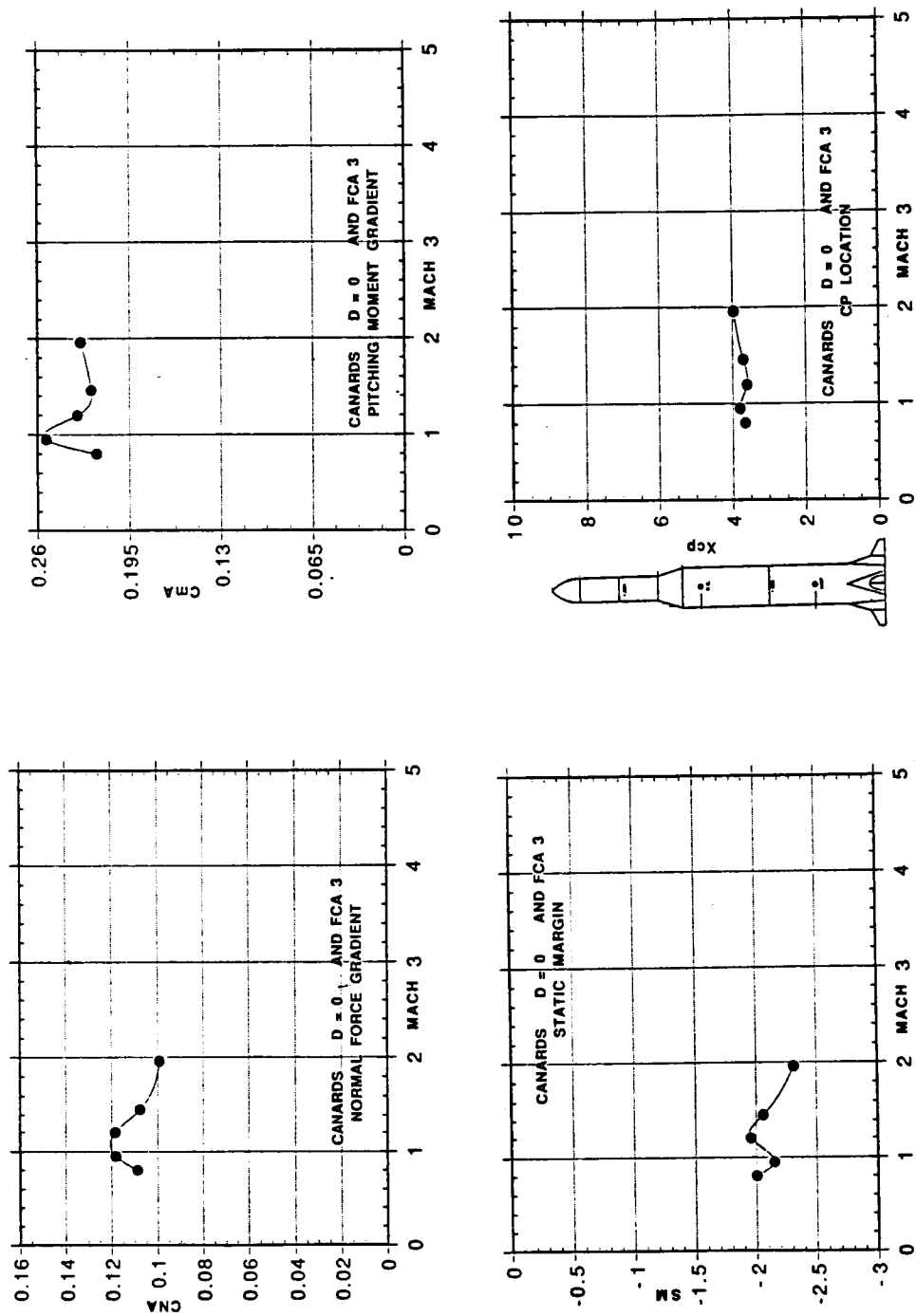


Figure C5. Results for canard;  $\delta = 0^\circ$ .

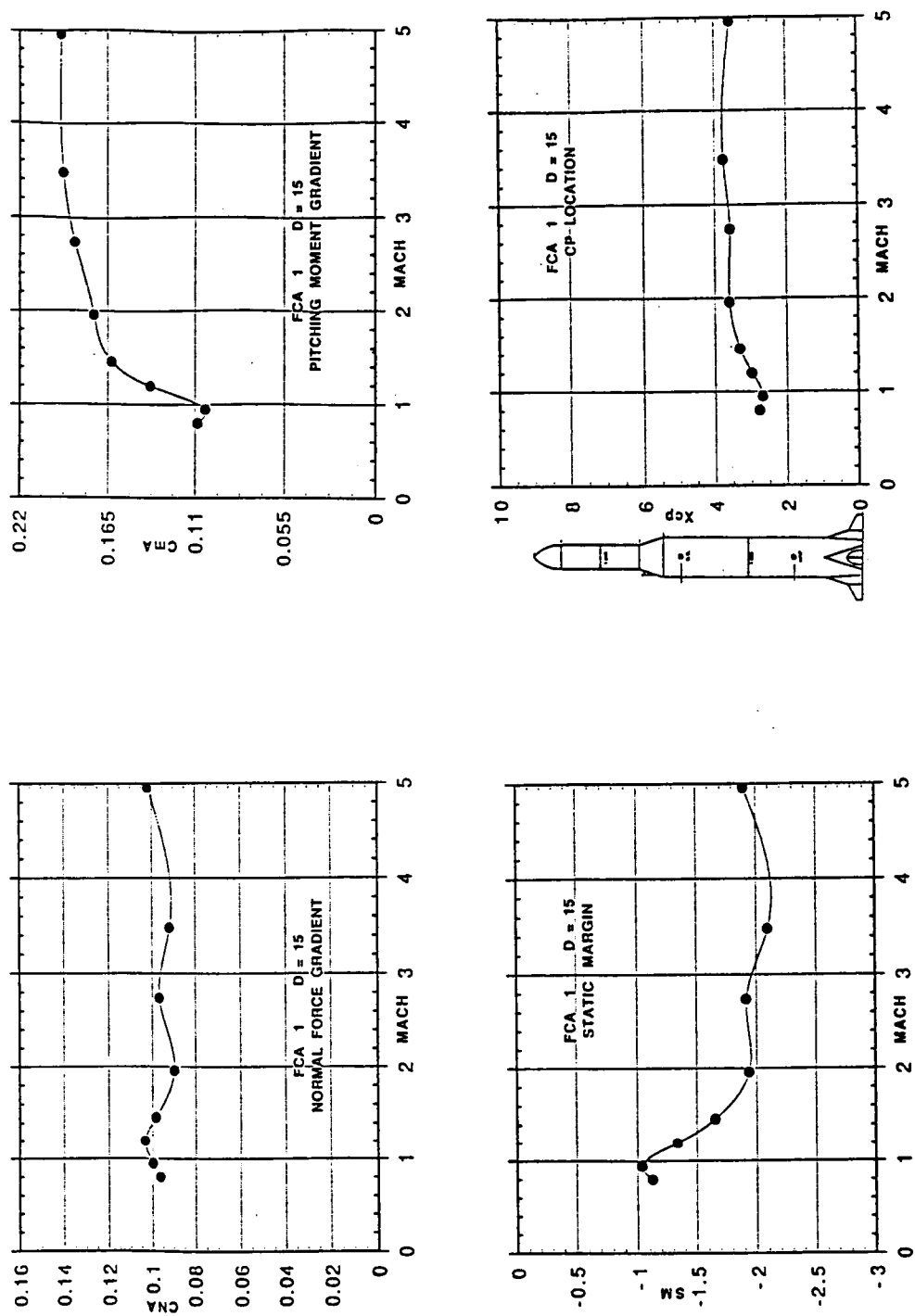


Figure C6. Results for FCA 1;  $\delta = 15^\circ$ .

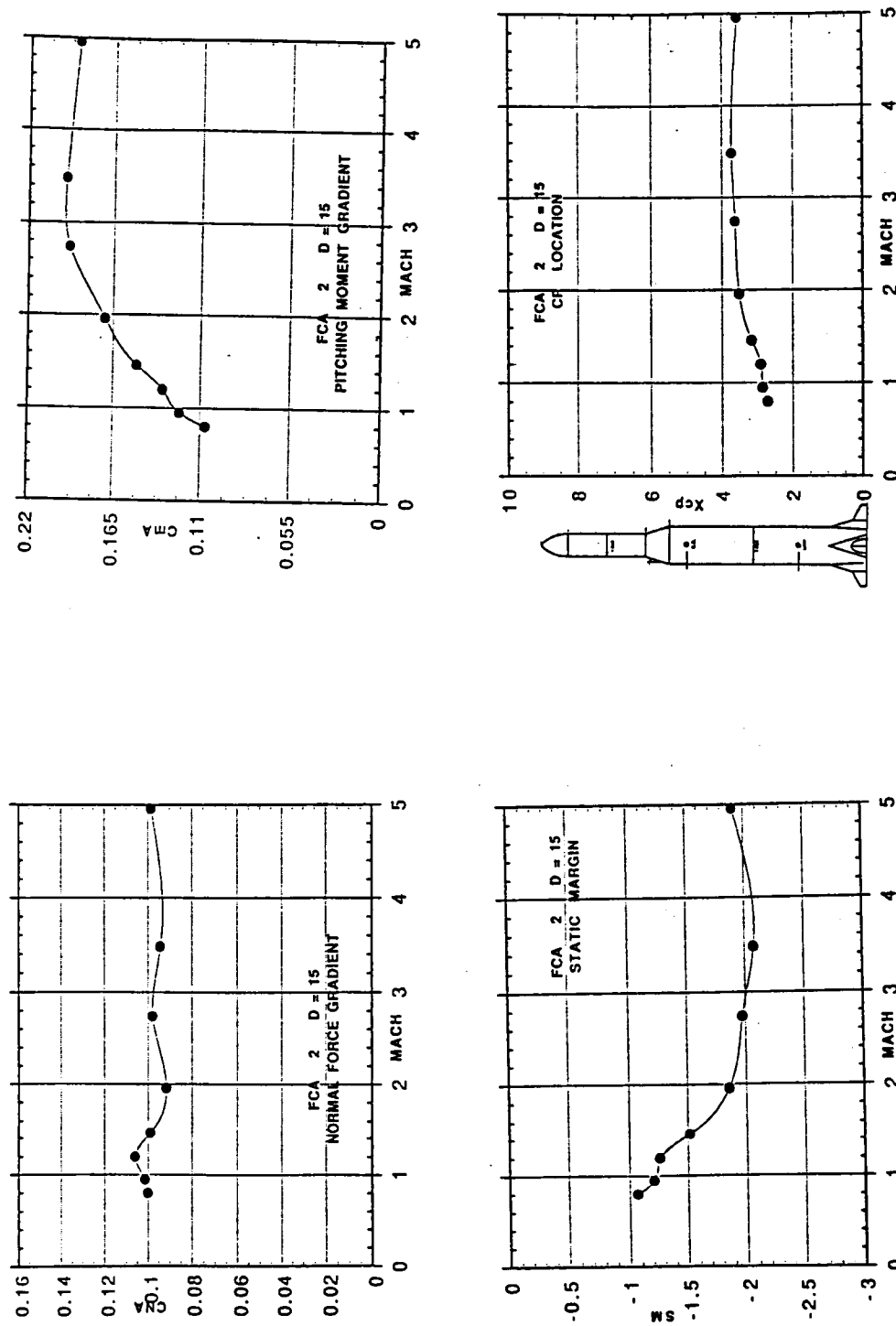


Figure C7. Results for FCA 2;  $\delta = 15^\circ$ .

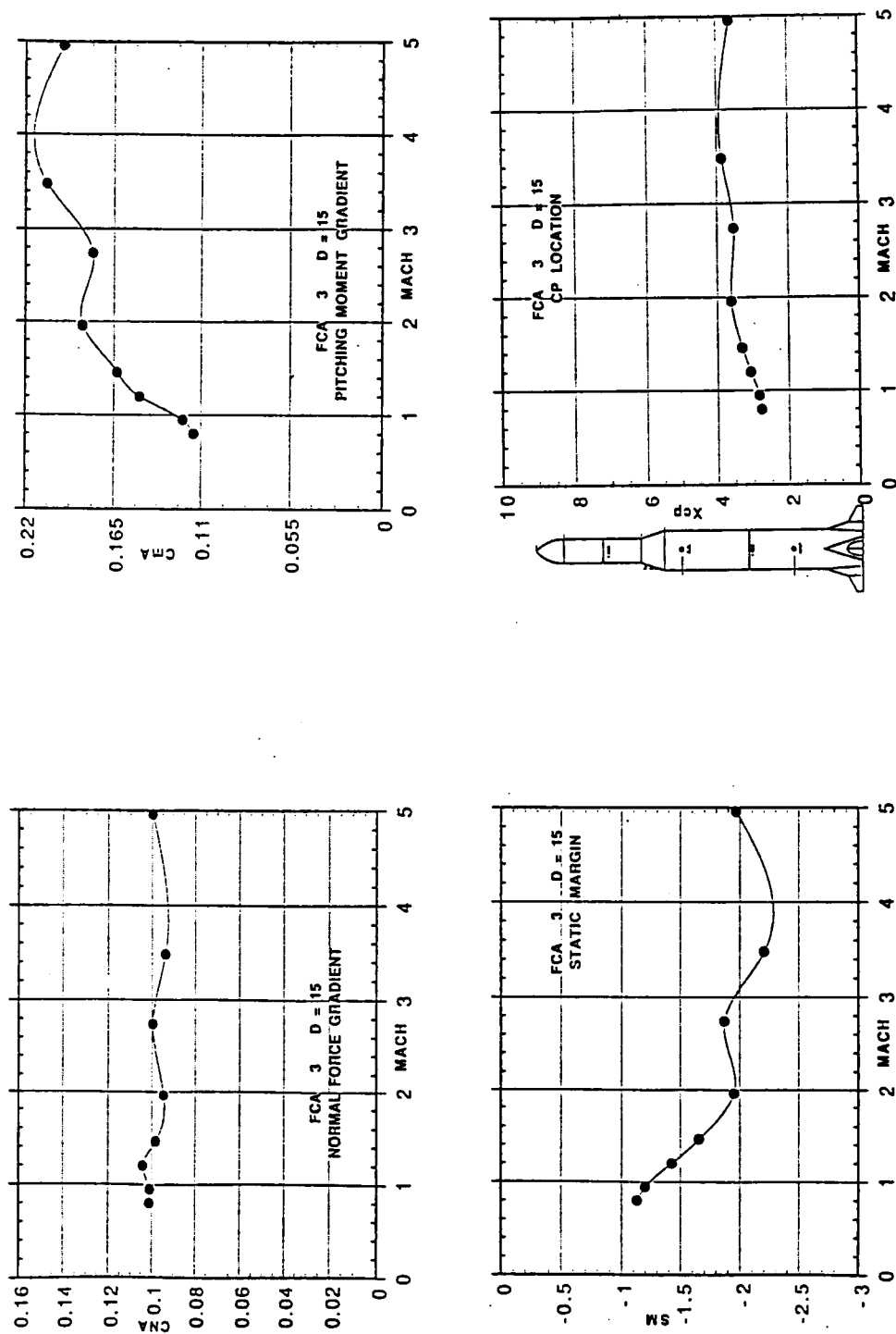


Figure C8. Results for FCA 3;  $\delta = 15^\circ$ .



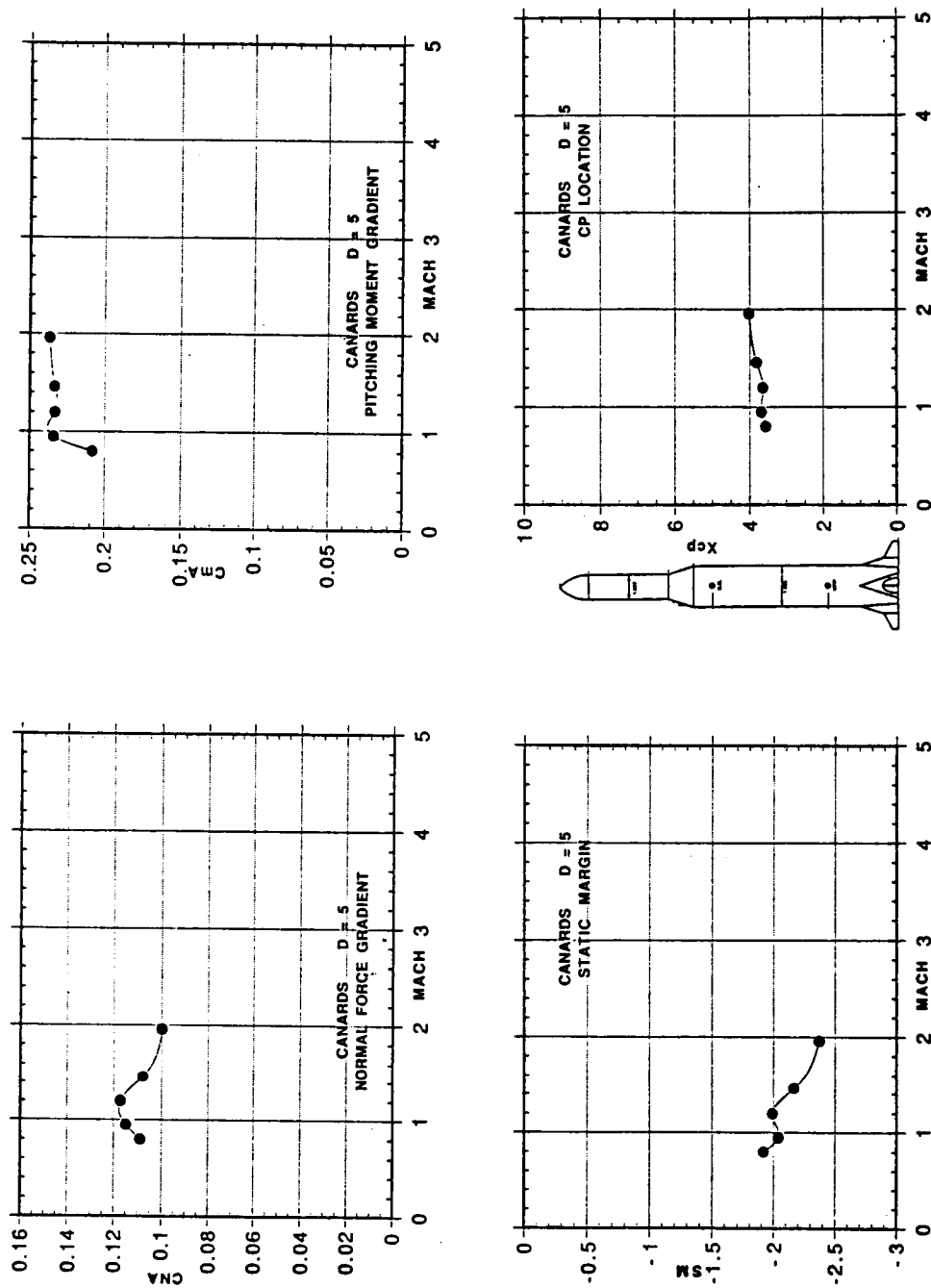


Figure C9. Results for canards;  $\delta = 5^\circ$ .

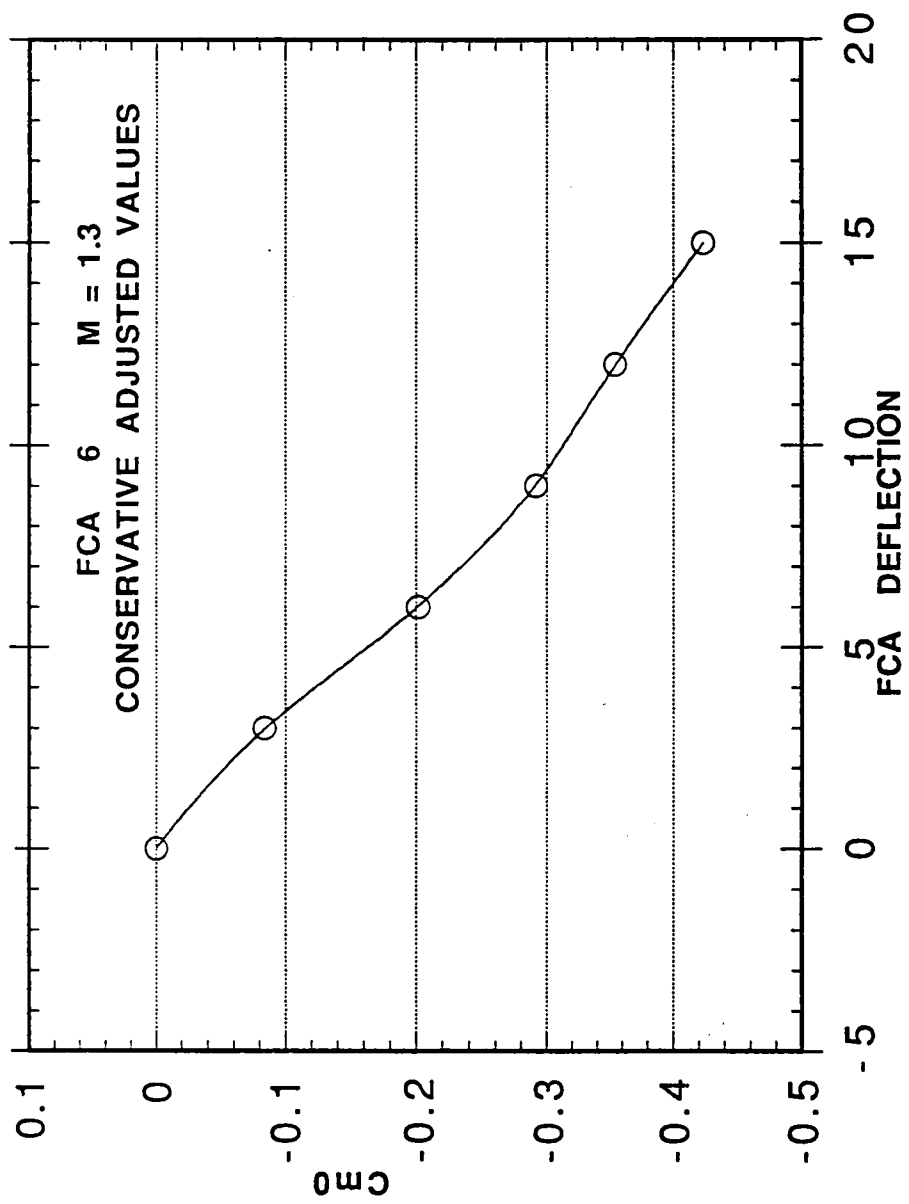


Figure C10.  $C_{m_0}$  vs. FCA deflection.

## **APPENDIX D**

### **DYNAMIC STABILITY DATA**

## **D.1 DATCOM Computer Simulations**

THE USAF AUTOMATED MISSILE DATUM \* REV 6/93 \*  
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS  
CONFER - INPUT ERROR CHECKING

0 ERROR CODES - N\* DENOTES THE NUMBER OF OCCURRENCES OF EACH ERROR

0 A - UNKNOWN VARIABLE NAME

0 B - MISSING EQUAL SIGN FOLLOWING VARIABLE NAME

0 C - NON-ARRAY VARIABLE HAS AN ARRAY ELEMENT DESIGNATION - (N)

0 D - NON-ARRAY VARIABLE HAS MULTIPLE VALUES ASSIGNED

0 E - ASSIGNED VALUES EXCEED ARRAY DIMENSION

0 F - SYNTAX ERROR

\*\*\*\*\* INPUT DATA CARDS \*\*\*\*\*

1 CASEID FCA #6 ; D = 0

2 SOSE

3 DIM FT

4 DERIV DEG

5 NO LAT

6 \$FLTCOM NMACH = 1.1,

7 MACH = 1.31,

8 ALT = 32165.,

9 NALPHA = 6., 2., 4., 6., 8., 10., \$

10 ALPHA = 0., 2., 4., 6., 8., 10., \$

11 XCG = 190.56

12 SREF = 597.58 \$

13 LREF = 27.58 \$

14 \$AXIBOD THOSE = OGIVE, LHOSE = 20., DNCOSE = 16.6657,

15 NX = 13.26, 40., 60., 80., 90., 97.43, 113., 170., 210., 234.72,

16 X = 0., 25., 40., 60., 80., 90., 97.43, 113., 170., 210., 234.72,

17 R = 0., 8.3333, 8.3333, 8.3333, 8.3333, 8.3333, 8.3333, 8.3333, 8.3333,

18 R = 0., 8.3333, 8.3333, 8.3333, 8.3333, 8.3333, 8.3333, 8.3333, 8.3333,

19 DISCON = 5.7, 11., 13., \$

20 \$FINSET1 SECTYP = HEL

21 SSPAN = 0.16

22 CHORD = 9.6635, 1.0032,

23 XLE = 247.4,

24 SWEEP = 0., STA = .,

25 NPANEL = 6.,

26 PHIF = 0.0, 90., 180., 270.,

27 LMAXU = .99,

28 LFLATU = 0.

29 ZUPPER = 0.0526, \$

30 \$DEFLCT

31 DELTA1 = 0., 0., 0., 0.,

32 XHNGE = 253.,

33 SKEN = C., \$

34 \*\*

35 \$TRIM

36 SET = 1.

37 ASYM = .TRUE.,

38 PNL1 = .FALSE.,

39 PNL2 = .TRUE.,

40 PNL3 = .FALSE.,

41 PNL4 = .TRUE.,

42 DELMIN = -15., \$

43 DELMAX = 15., \$

44 \*DAMP

45 PRINT AERO TRIM

46 PRINT AERO HINGE

47

\*\* SUBSTITUTING NUMERIC FOR NAME OGIVE

\*\* SUBSTITUTING NUMERIC FOR NAME HEX

48 \*PART  
49 \*PRESSURES  
50 \*PRINT GEOM BODY  
51 BUILD  
52 PLOT  
53 SAVE  
54 NEXT CASE  
1

CASE 1  
PAGE 1

THE USAF AUTOMATED MISSILE DATCOM \* REV 6/93 \*  
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS  
CASE INPUTS

OFOLLOWING ARE THE CARDS INPUT FOR THIS CASE

CASEID FCA #6 ; D = 0  
ROSE  
DIM FT  
DERIV DEG  
NO LAT  
\$FLTCN MMACH = 1.,  
MACH = 1.31,  
ALT = 32165.,  
NAPHA = 6.,  
ALPHA = 0., 2., 4., 6., 8., 10., \$  
\$REFQ XCG = 190.,  
SREF = 597.56,  
LREF = 27.58,  
\$AXIBOD TROSE = 1., LNOSE = 20., DRDSE = 16.6667,  
NK = 13.,  
X = 0., 26., 40., 60., 80., 90., 97.45, 135., 170., 210., 234.72,  
R = 0., 245., 257.133, 8.3333, 8.3333, 8.3333, 11.462, 13.792, 13.792,  
DISCON = 5.6, 7., 11., 13., \$  
\$FINSET1 SECTYP = 0.15,  
SSPAN = 0.6635, 1.0032,  
CHORD = 9.6635, 1.0032,  
XLE = 247.4,  
SWEEP = 0., STA = 1.,  
NAPANEL = 4.,  
PAIF = 0.0, 90., 180., 270.,  
LMAXU = .96,  
LFLATU = 0.0526, \$  
ZUPPER = 0.0526, \$  
\$DEFLCT DELTA1 = 0.0, 0., 0., 0.,  
XHING = 25.,  
SKEW = 0., \$  
\*\* (Run trim or damping and deflection)  
\$TRIM  
\* SET = 1.,  
ASYN = .TRUE.,  
PANL1 = .FALSE.,  
PANL2 = .TRUE.,  
PANL3 = .FALSE.,  
PANL4 = .TRUE.,  
DELMIN = -15.,  
DELMAX = 15., \$  
\*DAMP  
PRINT AERO TRIM  
PRINT AERO HINGE  
\*PART  
\*PRESSURES

\*PRINT GEOM BODY

BUILD

PLOT

SAVE

NEAT CASE

OTHE BOUNDARY LAYER IS ASSUMED TO BE TURBULENT OVER ALL COMPONENTS OF THE CONFIGURATION

OTHE INPUT UNITS ARE IN FEET, THE SCALE FACTOR IS 1.0000

THE USAF AUTOMATED MISSILE DATCOM \* REV 5/93 \*

AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS

FCA #6 : D = 0

UNTRIMMED STATIC AERODYNAMIC COEFFICIENTS

CASE 1  
PAGE 2

| MACH<br>NUMBER | ALTITUDE<br>FT | FLIGHT CONDITIONS  |                                | REYNOLDS<br>NUMBER | SIDESLIP<br>ANGLE<br>DEG | ROLL<br>ANGLE<br>DEG | REFERENCE DIMENSIONS            |                            | REF. CENTER<br>VERTICAL<br>FT |
|----------------|----------------|--------------------|--------------------------------|--------------------|--------------------------|----------------------|---------------------------------|----------------------------|-------------------------------|
|                |                | VELOCITY<br>FT/SEC | PRESSURE<br>LB/FT <sup>2</sup> |                    |                          |                      | REP.<br>AREA<br>FT <sup>2</sup> | REF. LENGTH<br>LONG.<br>FT |                               |
| 1.31           | 32165.00       | 1290.80            | 5.702E+02                      | 404.14             | 3.453E+06                | 0.00                 | 597.560                         | 27.580                     | 190.000                       |

# TABLE OF UNTRIMMED NORMAL FORCE COEFFICIENTS

PANEL DEFLECTION ANGLE, DEG.

| ALPHA | -15.0000 | -11.6667 | -8.3333 | -5.0000 | -1.6667 | 1.6667 | 5.0000 | 8.3333 | 11.6667 | 15.0000 |
|-------|----------|----------|---------|---------|---------|--------|--------|--------|---------|---------|
|       |          |          |         |         |         |        |        |        |         |         |
| 0.00  | -0.3556  | -0.2758  | -0.1942 | -0.1147 | -0.0372 | 0.0372 | 0.147  | 0.1949 | 0.2759  | 0.3556  |
| 2.00  | -0.3578  | -0.0786  | 0.0604  | 0.0773  | 0.1517  | 0.2294 | 0.305  | 0.3929 | 0.4747  | 0.5540  |
| 4.00  | 0.0498   | 0.1274   | 0.2035  | 0.2775  | 0.3552  | 0.4367 | 0.5203 | 0.6040 | 0.6857  | 0.7634  |
| 6.00  | 0.3505   | 0.4253   | 0.4987  | 0.5768  | 0.6591  | 0.7439 | 0.8292 | 0.9131 | 0.9935  | 1.0691  |
| 8.00  | 0.6276   | 0.7004   | 0.7775  | 0.8598  | 0.9452  | 1.0320 | 1.1180 | 1.2014 | 1.2808  | 1.3504  |
| 10.00 | 0.9166   | 0.9917   | 1.0728  | 1.1581  | 1.2458  | 1.3338 | 1.4202 | 1.5034 | 1.5797  | 1.6432  |

FIN SET FIN 1 FIN 2 FIN 3 FIN 4 FIN 5 FIN 6 FIN 7 FIN 8

1 0.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00

THE USAF AUTOMATED MISSILE DATCOM \* REV 6/93 \*

AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS

FCA #6 : D = 0

UNTRIMMED STATIC AERODYNAMIC COEFFICIENTS

CASE 1  
PAGE 3

| MACH<br>NUMBER | ALTITUDE<br>FT | FLIGHT CONDITIONS  |                                | REYNOLDS<br>NUMBER | SIDESLIP<br>ANGLE<br>DEG | ROLL<br>ANGLE<br>DEG | REP.<br>AREA<br>FT <sup>2</sup> | REFERENCE DIMENSIONS       |                           | REF. CENTER<br>VERTICAL<br>FT |
|----------------|----------------|--------------------|--------------------------------|--------------------|--------------------------|----------------------|---------------------------------|----------------------------|---------------------------|-------------------------------|
|                |                | VELOCITY<br>FT/SEC | PRESSURE<br>LB/FT <sup>2</sup> |                    |                          |                      |                                 | REF. LENGTH<br>LONG.<br>FT | REF. LENGTH<br>LAT.<br>FT |                               |
| 1.31           | 32165.00       | 1290.80            | 5.702E+02                      | 404.14             | 3.453E+06                | 0.00                 | 597.550                         | 27.580                     | 190.000                   | 0.000                         |

# TABLE OF UNTRIMMED PITCHING MOMENT COEFFICIENTS

PANEL DEFLECTION ANGLE, DEG.

| ALPHA | -15.0000 | -11.6667 | -8.3333 | -5.0000 | -1.6667 | 1.6667  | 5.0000  | 8.3333  | 11.6667 | 15.0000 |
|-------|----------|----------|---------|---------|---------|---------|---------|---------|---------|---------|
|       |          |          |         |         |         |         |         |         |         |         |
| 0.00  | 0.8221   | 0.6377   | 0.4505  | 0.2652  | 0.0361  | -0.0961 | -0.2652 | -0.4506 | -0.6377 | -0.8221 |
| 2.00  | 0.9166   | 0.7335   | 0.5507  | 0.3729  | 0.2010  | 0.0213  | -0.1651 | -0.3556 | -0.5453 | -0.7292 |
| 4.00  | 1.0329   | 0.8534   | 0.6774  | 0.5064  | 0.3268  | 0.1382  | -0.0551 | -0.2484 | -0.4373 | -0.6171 |
| 6.00  | 1.2740   | 1.1011   | 0.9213  | 0.7509  | 0.5606  | 0.3645  | 0.1673  | -0.0257 | -0.2136 | -0.3874 |
| 8.00  | 1.5530   | 1.3847   | 1.2063  | 1.0161  | 0.8186  | 0.5180  | 0.4190  | 0.2262  | 0.0427  | -0.1183 |
| 10.00 | 1.9135   | 1.7399   | 1.5524  | 1.3552  | 1.1526  | 0.9491  | 0.7493  | 0.5570  | 0.3804  | 0.2337  |

FIN SET FIN 1 FIN 2 FIN 3 FIN 4 FIN 5 FIN 6 FIN 7 FIN 8  
 1 0.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00  
 THE USAF AUTOMATED MISSILE DATCON \* REV 6/93 \*  
 AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS  
 FCA #6 ; D = 0  
 UNTRIMMED STATIC AERODYNAMIC COEFFICIENTS

1

CASE 1  
 PAGE 4

| MACH<br>NUMBER | FLIGHT CONDITIONS |                    |                                   | REF. DIMENSIONS              |                   |                                | REF. DIMENSIONS              |                   |                                |
|----------------|-------------------|--------------------|-----------------------------------|------------------------------|-------------------|--------------------------------|------------------------------|-------------------|--------------------------------|
|                | ALTITUDE<br>FT    | VELOCITY<br>FT/SEC | TEMPERATURE<br>LB/FT <sup>2</sup> | REF. AREA<br>FT <sup>2</sup> | REF. LENGTH<br>FT | REF. MOMENT<br>FT <sup>2</sup> | REF. AREA<br>FT <sup>2</sup> | REF. LENGTH<br>FT | REF. MOMENT<br>FT <sup>2</sup> |
| 1.31           | 32165.00          | 1290.80            | 5.702E+02                         | 404.14                       | 3.453E+06         | 0.00                           | 597.560                      | 27.580            | 190.000                        |

TABLE OF UNTRIMMED AXIAL FORCE COEFFICIENTS

PANEL DEFLECTION ANGLE, DEG.

| ALPHA | FLIGHT CONDITIONS |                    |                                   | REF. DIMENSIONS              |                   |                                | REF. DIMENSIONS              |                   |                                |
|-------|-------------------|--------------------|-----------------------------------|------------------------------|-------------------|--------------------------------|------------------------------|-------------------|--------------------------------|
|       | ALTITUDE<br>FT    | VELOCITY<br>FT/SEC | TEMPERATURE<br>LB/FT <sup>2</sup> | REF. AREA<br>FT <sup>2</sup> | REF. LENGTH<br>FT | REF. MOMENT<br>FT <sup>2</sup> | REF. AREA<br>FT <sup>2</sup> | REF. LENGTH<br>FT | REF. MOMENT<br>FT <sup>2</sup> |
| 0.00  | -15.0000          | -11.6667           | -8.3333                           | -5.0000                      | -1.6667           | 1.5667                         | 5.0000                       | 8.3333            | 11.5667                        |
| 2.00  | 0.6620            | 0.6278             | 0.6024                            | 0.5359                       | 0.5779            | 0.5779                         | 0.5859                       | 0.6024            | 0.6278                         |
| 4.00  | 0.6406            | 0.6110             | 0.5902                            | 0.5281                       | 0.5742            | 0.5742                         | 0.5902                       | 0.6110            | 0.6406                         |
| 6.00  | 0.6159            | 0.5910             | 0.5747                            | 0.5668                       | 0.5568            | 0.5568                         | 0.5910                       | 0.6159            | 0.6406                         |
| 8.00  | 0.5862            | 0.5654             | 0.5550                            | 0.5509                       | 0.5546            | 0.5546                         | 0.5862                       | 0.6159            | 0.6406                         |
| 10.00 | 0.5576            | 0.5418             | 0.5333                            | 0.5324                       | 0.5396            | 0.5396                         | 0.5862                       | 0.6159            | 0.6406                         |
|       | 0.5295            | 0.5161             | 0.5097                            | 0.5114                       | 0.5217            | 0.5217                         | 0.5862                       | 0.6159            | 0.6406                         |

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FIN SET FIN 1 FIN 2 FIN 3 FIN 4 FIN 5 FIN 6 FIN 7 FIN 8  
 1 0.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00  
 THE USAF AUTOMATED MISSILE DATCON \* REV 6/93 \*  
 AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS  
 FCA #6 ; D = 0  
 UNTRIMMED STATIC AERODYNAMIC COEFFICIENTS

1

CASE 1  
 PAGE 5

| MACH<br>NUMBER | FLIGHT CONDITIONS |                    |                                   | REF. DIMENSIONS              |                   |                                | REF. DIMENSIONS              |                   |                                |
|----------------|-------------------|--------------------|-----------------------------------|------------------------------|-------------------|--------------------------------|------------------------------|-------------------|--------------------------------|
|                | ALTITUDE<br>FT    | VELOCITY<br>FT/SEC | TEMPERATURE<br>LB/FT <sup>2</sup> | REF. AREA<br>FT <sup>2</sup> | REF. LENGTH<br>FT | REF. MOMENT<br>FT <sup>2</sup> | REF. AREA<br>FT <sup>2</sup> | REF. LENGTH<br>FT | REF. MOMENT<br>FT <sup>2</sup> |
| 1.31           | 32165.00          | 1290.80            | 5.702E+02                         | 404.14                       | 3.453E+06         | 0.00                           | 597.560                      | 27.580            | 190.000                        |

TABLE OF UNTRIMMED SIDE FORCE COEFFICIENTS

PANEL DEFLECTION ANGLE, DEG.

| ALPHA | FLIGHT CONDITIONS |                    |                                   | REF. DIMENSIONS              |                   |                                | REF. DIMENSIONS              |                   |                                |
|-------|-------------------|--------------------|-----------------------------------|------------------------------|-------------------|--------------------------------|------------------------------|-------------------|--------------------------------|
|       | ALTITUDE<br>FT    | VELOCITY<br>FT/SEC | TEMPERATURE<br>LB/FT <sup>2</sup> | REF. AREA<br>FT <sup>2</sup> | REF. LENGTH<br>FT | REF. MOMENT<br>FT <sup>2</sup> | REF. AREA<br>FT <sup>2</sup> | REF. LENGTH<br>FT | REF. MOMENT<br>FT <sup>2</sup> |
| 0.00  | -15.0000          | -11.6667           | -8.3333                           | -5.0000                      | -1.6667           | 1.6667                         | 5.0000                       | 8.3333            | 11.5667                        |
| 2.00  | 0.0000            | 0.0000             | 0.0000                            | 0.0000                       | 0.0000            | 0.0000                         | 0.0000                       | 0.0000            | 0.0000                         |
| 4.00  | 0.0000            | 0.0000             | 0.0000                            | 0.0000                       | 0.0000            | 0.0000                         | 0.0000                       | 0.0000            | 0.0000                         |
| 6.00  | 0.0000            | 0.0000             | 0.0000                            | 0.0000                       | 0.0000            | 0.0000                         | 0.0000                       | 0.0000            | 0.0000                         |
| 8.00  | 0.0000            | 0.0000             | 0.0000                            | 0.0000                       | 0.0000            | 0.0000                         | 0.0000                       | 0.0000            | 0.0000                         |
| 10.00 | 0.0000            | 0.0000             | 0.0000                            | 0.0000                       | 0.0000            | 0.0000                         | 0.0000                       | 0.0000            | 0.0000                         |

1

CASE 1  
 PAGE 6



# UNTRIMMED STATIC AERODYNAMIC COEFFICIENTS

| MACH NUMBER | ALTITUDE<br>FT | VELOCITY<br>FT/SEC | PRESSURE<br>LB/FT <sup>2</sup> | FLIGHT CONDITIONS    |                    |                          | REF. AREA<br>FT <sup>2</sup> | REFERENCE DIMENSIONS       |                      |                               |
|-------------|----------------|--------------------|--------------------------------|----------------------|--------------------|--------------------------|------------------------------|----------------------------|----------------------|-------------------------------|
|             |                |                    |                                | TEMPERATURE<br>DEG R | REYNOLDS<br>NUMBER | SIDESLIP<br>ANGLE<br>DEG |                              | REF. LENGTH<br>LONG.<br>FT | MOENT<br>LONG.<br>FT | CENTER<br>REF. VERTICAL<br>FT |
| 1.31        | 32165.00       | 1290.80            | 5.702E+02                      | 404.14               | 3.453E+06          | 0.00                     | 597.560                      | 27.580                     | 190.000              | 0.000                         |

## TABLE OF UNTRIMMED YAWING MOMENT COEFFICIENTS

PANEL DEFLECTION ANGLE, DEGS.

| ALPHA | -15.0000 | -11.6667 | -8.3333 | -5.0000 | -1.6667 | 1.6667 | 5.0000 | 8.3333 | 11.6667 | 15.0000 |
|-------|----------|----------|---------|---------|---------|--------|--------|--------|---------|---------|
|       |          |          |         |         |         |        |        |        |         |         |
| 0.00  | 0.0000   | 0.0000   | 0.0000  | 0.0000  | 0.0000  | 0.0000 | 0.0000 | 0.0000 | 0.0000  | 0.0000  |
| 2.00  | 0.0000   | 0.0000   | 0.0000  | 0.0000  | 0.0000  | 0.0000 | 0.0000 | 0.0000 | 0.0000  | 0.0000  |
| 4.00  | 0.0000   | 0.0000   | 0.0000  | 0.0000  | 0.0000  | 0.0000 | 0.0000 | 0.0000 | 0.0000  | 0.0000  |
| 6.00  | 0.0000   | 0.0000   | 0.0000  | 0.0000  | 0.0000  | 0.0000 | 0.0000 | 0.0000 | 0.0000  | 0.0000  |
| 8.00  | 0.0000   | 0.0000   | 0.0000  | 0.0000  | 0.0000  | 0.0000 | 0.0000 | 0.0000 | 0.0000  | 0.0000  |
| 10.00 | 0.0000   | 0.0000   | 0.0000  | 0.0000  | 0.0000  | 0.0000 | 0.0000 | 0.0000 | 0.0000  | 0.0000  |

FIN SET FIN 1 FIN 2 FIN 3 FIN 4 FIN 5 FIN 6 FIN 7 FIN 8

THE USAF AUTOMATED MISSILE DATCOM \* REV 6/93 \*

AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS

FCR #6 ; D = 0

## UNTRIMMED STATIC AERODYNAMIC COEFFICIENTS

| MACH NUMBER | ALTITUDE<br>FT | VELOCITY<br>FT/SEC | PRESSURE<br>LB/FT <sup>2</sup> | FLIGHT CONDITIONS    |                    |                          | REF. AREA<br>FT <sup>2</sup> | REFERENCE DIMENSIONS       |                      |                               |
|-------------|----------------|--------------------|--------------------------------|----------------------|--------------------|--------------------------|------------------------------|----------------------------|----------------------|-------------------------------|
|             |                |                    |                                | TEMPERATURE<br>DEG R | REYNOLDS<br>NUMBER | SIDESLIP<br>ANGLE<br>DEG |                              | REF. LENGTH<br>LONG.<br>FT | MOENT<br>LONG.<br>FT | CENTER<br>REF. VERTICAL<br>FT |
| 1.31        | 32165.00       | 1290.80            | 5.702E+02                      | 404.14               | 3.453E+06          | 0.00                     | 597.560                      | 27.580                     | 190.000              | 0.000                         |

## TABLE OF UNTRIMMED ROLLING MOMENT COEFFICIENTS

PANEL DEFLECTION ANGLE, DEGS.

| ALPHA | -15.0000 | -11.6667 | -8.3333 | -5.0000 | -1.6667 | 1.6667 | 5.0000 | 8.3333 | 11.6667 | 15.0000 |
|-------|----------|----------|---------|---------|---------|--------|--------|--------|---------|---------|
|       |          |          |         |         |         |        |        |        |         |         |
| 0.00  | 0.0000   | 0.0000   | 0.0000  | 0.0000  | 0.0000  | 0.0000 | 0.0000 | 0.0000 | 0.0000  | 0.0000  |
| 2.00  | 0.0000   | 0.0000   | 0.0000  | 0.0000  | 0.0000  | 0.0000 | 0.0000 | 0.0000 | 0.0000  | 0.0000  |
| 4.00  | 0.0000   | 0.0000   | 0.0000  | 0.0000  | 0.0000  | 0.0000 | 0.0000 | 0.0000 | 0.0000  | 0.0000  |
| 6.00  | 0.0000   | 0.0000   | 0.0000  | 0.0000  | 0.0000  | 0.0000 | 0.0000 | 0.0000 | 0.0000  | 0.0000  |
| 8.00  | 0.0000   | 0.0000   | 0.0000  | 0.0000  | 0.0000  | 0.0000 | 0.0000 | 0.0000 | 0.0000  | 0.0000  |
| 10.00 | 0.0000   | 0.0000   | 0.0000  | 0.0000  | 0.0000  | 0.0000 | 0.0000 | 0.0000 | 0.0000  | 0.0000  |

FIN SET FIN 1 FIN 2 FIN 3 FIN 4 FIN 5 FIN 6 FIN 7 FIN 8

THE USAF AUTOMATED MISSILE DATCOM \* REV 6/93 \*

AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS

FCR #6 ; D = 0

## TRIMMED STATIC AERODYNAMIC COEFFICIENTS

| MACH NUMBER | ALTITUDE<br>FT | VELOCITY<br>FT/SEC | PRESSURE<br>LB/FT <sup>2</sup> | FLIGHT CONDITIONS    |                    |                          | REF. AREA<br>FT <sup>2</sup> | REFERENCE DIMENSIONS       |                      |                               |
|-------------|----------------|--------------------|--------------------------------|----------------------|--------------------|--------------------------|------------------------------|----------------------------|----------------------|-------------------------------|
|             |                |                    |                                | TEMPERATURE<br>DEG R | REYNOLDS<br>NUMBER | SIDESLIP<br>ANGLE<br>DEG |                              | REF. LENGTH<br>LONG.<br>FT | MOENT<br>LONG.<br>FT | CENTER<br>REF. VERTICAL<br>FT |
| 1.31        | 32165.00       | 1290.80            | 5.702E+02                      | 404.14               | 3.453E+06          | 0.00                     | 597.560                      | 27.580                     | 190.000              | 0.000                         |

| NUMBER | FT    | 32165.00 | FT/SEC  | 1290.50 | LB/FT**2 | 5.702E+02 | DEG R  | 404.14 | NUMBER | 1/FT | 3.453E+06 | ANGLE | DEG    | 0.00 | ANGLE  | DEG | 0.00   | AREA | FT**2  | 597.560 | LONG.  | FT | 27.580 | LAT. | FT     | 27.580 | LONG.  | FT | 190.000 | VERTICAL | FT     | 0.000 |
|--------|-------|----------|---------|---------|----------|-----------|--------|--------|--------|------|-----------|-------|--------|------|--------|-----|--------|------|--------|---------|--------|----|--------|------|--------|--------|--------|----|---------|----------|--------|-------|
| 0      | ALPHA |          | DELTA   |         | CL       |           | CD     |        | CN     |      | CA        |       | CY     |      | CLN    |     | CLL    |      |        |         |        |    |        |      |        |        |        |    |         |          |        |       |
|        | 0.00  |          | 0.0000  |         | 0.0000   |           | 0.5779 |        | 0.0000 |      | 0.5779    |       | 0.0000 |      | 0.0000 |     | 0.0000 |      | 0.0000 |         | 0.0000 |    | 0.0000 |      | 0.0000 |        | 0.0000 |    | 0.0000  |          | 0.0000 |       |
|        | 2.00  |          | 2.0452  |         | 0.2133   |           | 0.5874 |        | 0.2355 |      | 0.5794    |       | 0.0000 |      | 0.0000 |     | 0.0000 |      | 0.0000 |         | 0.0000 |    | 0.0000 |      | 0.0000 |        | 0.0000 |    | 0.0000  |          | 0.0000 |       |
|        | 4.00  |          | 4.0501  |         | 0.4544   |           | 0.5194 |        | 0.4963 |      | 0.5862    |       | 0.0000 |      | 0.0000 |     | 0.0000 |      | 0.0000 |         | 0.0000 |    | 0.0000 |      | 0.0000 |        | 0.0000 |    | 0.0000  |          | 0.0000 |       |
|        | 6.00  |          | 7.8748  |         | 0.8323   |           | 0.7058 |        | 0.5015 |      | 0.6149    |       | 0.0000 |      | 0.0000 |     | 0.0000 |      | 0.0000 |         | 0.0000 |    | 0.0000 |      | 0.0000 |        | 0.0000 |    | 0.0000  |          | 0.0000 |       |
|        | 8.00  |          | 12.5514 |         | 1.1932   |           | 0.8460 |        | 1.2993 |      | 0.6717    |       | 0.0000 |      | 0.0000 |     | 0.0000 |      | 0.0000 |         | 0.0000 |    | 0.0000 |      | 0.0000 |        | 0.0000 |    | 0.0000  |          | 0.0000 |       |
|        | 10.00 |          |         |         |          |           |        |        |        |      |           |       |        |      |        |     |        |      |        |         |        |    |        |      |        |        |        |    |         |          |        |       |

OPANELS FROM FILM SET 1 WERE DEFLECTED OVER THE RANGE -15.0000 TO 15.0000 DEG.

PANEL 1 WAS FIXED  
PANEL 2 WAS VARIED  
PANEL 3 WAS FIXED  
PANEL 4 WAS VARIED

OMOTE - \*NT\* PRINTED WHEN NO TRIM POINT COULD BE FOUND  
1\*\*\* END OF JOB \*\*\*

THE USAF AUTOMATED MISSILE DATCOM \* REV 6/93 \*  
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS  
CONERR - INPUT ERROR CHECKING

0 ERROR CODES - N\* DENOTES THE NUMBER OF OCCURRENCES OF EACH ERROR  
0 A - UNKNOWN VARIABLE NAME  
0 B - MISSING EQUAL SIGN FOLLOWING VARIABLE NAME  
0 C - NON-ARRAY VARIABLE HAS AN ARRAY ELEMENT DESIGNATION - (N)  
0 D - NON-ARRAY VARIABLE HAS MULTIPLE VALUES ASSIGNED  
0 E - ASSIGNED VALUES EXCEED ARRAY DIMENSION  
0 F - SYNTAX ERROR

\*\*\*\*\* INPUT DATA CARDS \*\*\*\*\*

```

1 CASEID FCA #6 ; D = 0
2 SOSE
3 DIM FT
4 DERIV DEG
5 NO LAT
6 $FLATCON FNACH = 1.,
7 MACH = 1.31,
8 ALT = 32185.,
9 ALPHA = 6., 2., 4., 6., 8., 10., $
10 $REFQ XCG = 190.56,
11 SREF = 597.56,
12 LREF = 27.58, $
13 $AXIBOD TNOSE = OGIV3, LNOSE = 20., DNOSE = 16.6667,
14 NK = 13.
15 X 0., 20., 40., 60., 80., 90., 97.45, 135., 170., 210., 234.72,
16 R 245., 257.53, 8.3333, 8.3333, 8.3333, 8.3333, 11.462, 13.792, 13.792,
17 13.792, 13.792, 13.792, 15.298, 17.,
18 DISCON = 5., 7., 11., 13., $
19 $FINSET1 SECTYP = HEX
20 SSPAN = 0.15,
21 CHORD = 9.6635, 1.0032,
22 XLE = 247.4, STA = 1.,
23 SWEEP = 0.,
24 NPANEL = 4.,
25 PHIF = 0.0, 90., 180., 270.,
26 LMAXU = .99,
27 LFLATU = 0.
28 ZUPPER = 0.0526, $
29 $DEFLECT DELTA1 = 0., 0., 0., 0.,
30 XHINGE = 253.,
31 SKEW = 0., $
32 (Run trim or damping and deflection)
33 **
34 $TRIM
35 SET = 1.,
36 ASYM = .FALSE.,
37 PNL1 = .FALSE.,
38 PNL2 = .TRUE.,
39 PNL3 = .FALSE.,
40 PNL4 = .TRUE.,
41 DELMIN = -15.,
42 DELMAX = 15., $
43 DAMP
44 PRINT AERO TRIM
45 PRINT AERO HINGE
46

```

\*\* SUBSTITUTING NUMERIC FOR NAME OGIVE

\*\* SUBSTITUTING NUMERIC FOR NAME HEX

48 PART  
49 \*PRESSURES  
50 PRINT GEOM BODY  
51 BUILD  
52 PLOT  
53 SAVE  
54 NEXT CASE

1

THE USAF AUTOMATED MISSILE DATCOM \* REV 6/83 \*  
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS  
CASE INPUTS

CASE 1  
PAGE 1

OFOLLOWING ARE THE CARDS INPUT FOR THIS CASE

```

CASEID FCA #6 ; D = 0
ROSE
DIM FT
DERIV DEG
NO LAT
$FLTCN NMACH = 1.;
MACH = 1.31;
ALT = 32165.;
NAPHA = 6.;
ALPHA = 0.; 2.; 4.; 6.; 8.; 10.; $
$REFQ XCG = 190.56;
SREF = 597.56;
LREF = 27.58.;
$AXIBD TNOSE = 1.; LNOSE = 20.; DNOSE = 16.6667,
X = 0.; 20.; 40.; 60.; 80.; 90.; 97.45, 135.; 170.; 210.; 234.72,
245.; 257.4;
R = 0.; 8.3333, 8.3333, 8.3333, 11.462, 13.792, 13.792,
13.792, 13.792, 15.298, 17.;
DISCON = 5.67., 11., 13.; $
$FINSET1 SECTYP = 6.; 15.;
SSPAN = 0.6635, 1.0032,
CHORD = 9.6635, 1.0032,
XLE = 247.4,
SWEEP = 0.; STA = 1.,
NPANEL = 4.;
PHIF = 0.090., 180., 270.,
LMAXU = .99,
LFLATO = 0.;
ZUPPER = 0.6526, $
$DEFLCT DELTA1 = 0.; 0.; 0.; 0.;
XHINGE = 253.;
SKEW = 0.; $
** (Run trim or damping and deflection)
**
*$TRIM SET = 1.;
ASYN = .FALSE.;
PANL1 = .FALSE.;
PANL2 = .TRUE.;
PANL3 = .FALSE.;
PANL4 = .TRUE.;
DELHIN = -15.;
DELMAX = 15.; $
DAMP
PRINT AERO TRIM
PRINT AERO HINGE
PART
*PRESSURES

```

# PRINT GEOM BODY

BUILD  
PLOT  
SAVE

## NEXT CASE

OTHE BOUNDARY LAYER IS ASSUMED TO BE TURBULENT OVER ALL COMPONENTS OF THE CONFIGURATION  
OTHE INPUT UNITS ARE IN FEET, THE SCALE FACTOR IS 1.0000

THE USAF AUTOMATED MISSILE DATCOM \* REV 6/93 \*  
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS

FCA #6 ; D = 0  
AXISYMMETRIC BODY DEFINITION

CASE 1  
PAGE 2

LENGTH= 257.0000 FINENESS RATIO= 7.5588 PLANFORM AREA=6025.8340 AREA CENTROID= 146.8190 WETTED AREA=\*\*\*\*\*  
VOLUME=\*\*\*\*\* VOLUME CENTROID= 158.4596

UNITS ARE IN FT

## MOLD LINE CONTOUR

| LONGITUDINAL STATIONS | 0.0000    | 20.0000  | 40.0000   | 60.0000 | 80.0000 | 90.0000 | 97.4500* | 135.0000 | 170.0000 | 210.0000 |
|-----------------------|-----------|----------|-----------|---------|---------|---------|----------|----------|----------|----------|
| 0                     | 234.7200* | 245.0000 | 257.0000* | 8.3333  | 8.3333  | 8.3333* | 11.4620  | 13.7920* | 13.7920  | 13.7920  |
| BODY RADII            | 0.0000    | 8.3333   | 8.3333    | 8.3333  | 8.3333  | 8.3333* | 11.4620  | 13.7920* | 13.7920  | 13.7920  |
|                       | 13.7920*  | 15.2980  | 17.0000*  |         |         |         |          |          |          |          |

## NOTE - \* INDICATES SLOPE DISCONTINUOUS POINTS

THE USAF AUTOMATED MISSILE DATCOM \* REV 6/93 \*  
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS

FCA #6 ; D = 0

## FIN SET NUMBER 1 AIRFOIL SECTION

CASE 1  
PAGE 3

| NACA 8-1-81.1-08.5-00.0 |          | LOWER ORIGINATE |          | X-FRACTION CHORD |          | MEAN LINE |          | THICKNESS |          |
|-------------------------|----------|-----------------|----------|------------------|----------|-----------|----------|-----------|----------|
| UPPER                   | ABSCISSA | UPPER           | ORDINATE | LOWER            | ABSCISSA | LOWER     | ORDINATE | UPPER     | ABSCISSA |
| 0                       | 0.00000  | 0.00000         | 0.00000  | 0.00000          | 0.00000  | 0.00000   | 0.00000  | 0.00000   | 0.00000  |
|                         | 0.00100  | 0.00005         | -0.00005 | 0.00100          | 0.00000  | 0.00000   | 0.00000  | 0.00100   | 0.00000  |
|                         | 0.00200  | 0.00010         | -0.00010 | 0.00200          | 0.00000  | 0.00000   | 0.00000  | 0.00200   | 0.00000  |
|                         | 0.00300  | 0.00016         | -0.00016 | 0.00300          | 0.00000  | 0.00000   | 0.00000  | 0.00300   | 0.00000  |
|                         | 0.00400  | 0.00021         | -0.00021 | 0.00400          | 0.00000  | 0.00000   | 0.00000  | 0.00400   | 0.00000  |
|                         | 0.00500  | 0.00026         | -0.00026 | 0.00500          | 0.00000  | 0.00000   | 0.00000  | 0.00500   | 0.00000  |
|                         | 0.00600  | 0.00031         | -0.00031 | 0.00600          | 0.00000  | 0.00000   | 0.00000  | 0.00600   | 0.00000  |
|                         | 0.00800  | 0.00042         | -0.00042 | 0.00800          | 0.00000  | 0.00000   | 0.00000  | 0.00800   | 0.00000  |
|                         | 0.01000  | 0.00052         | -0.00052 | 0.01000          | 0.00000  | 0.00000   | 0.00000  | 0.01000   | 0.00000  |
|                         | 0.02000  | 0.00105         | -0.00105 | 0.02000          | 0.00000  | 0.00000   | 0.00000  | 0.02000   | 0.00000  |
|                         | 0.03000  | 0.00157         | -0.00157 | 0.03000          | 0.00000  | 0.00000   | 0.00000  | 0.03000   | 0.00000  |
|                         | 0.04000  | 0.00210         | -0.00210 | 0.04000          | 0.00000  | 0.00000   | 0.00000  | 0.04000   | 0.00000  |
|                         | 0.05000  | 0.00262         | -0.00262 | 0.05000          | 0.00000  | 0.00000   | 0.00000  | 0.05000   | 0.00000  |
|                         | 0.06000  | 0.00314         | -0.00314 | 0.06000          | 0.00000  | 0.00000   | 0.00000  | 0.06000   | 0.00000  |
|                         | 0.08000  | 0.00419         | -0.00419 | 0.08000          | 0.00000  | 0.00000   | 0.00000  | 0.08000   | 0.00000  |
|                         | 0.10000  | 0.00524         | -0.00524 | 0.10000          | 0.00000  | 0.00000   | 0.00000  | 0.10000   | 0.00000  |
|                         | 0.12000  | 0.00629         | -0.00629 | 0.12000          | 0.00000  | 0.00000   | 0.00000  | 0.12000   | 0.00000  |
|                         | 0.14000  | 0.00734         | -0.00734 | 0.14000          | 0.00000  | 0.00000   | 0.00000  | 0.14000   | 0.00000  |
|                         | 0.16000  | 0.00838         | -0.00838 | 0.16000          | 0.00000  | 0.00000   | 0.00000  | 0.16000   | 0.00000  |
|                         | 0.18000  | 0.00943         | -0.00943 | 0.18000          | 0.00000  | 0.00000   | 0.00000  | 0.18000   | 0.00000  |
|                         | 0.20000  | 0.01048         | -0.01048 | 0.20000          | 0.00000  | 0.00000   | 0.00000  | 0.20000   | 0.00000  |
|                         | 0.22000  | 0.01153         | -0.01153 | 0.22000          | 0.00000  | 0.00000   | 0.00000  | 0.22000   | 0.00000  |
|                         | 0.24000  | 0.01258         | -0.01258 | 0.24000          | 0.00000  | 0.00000   | 0.00000  | 0.24000   | 0.00000  |
|                         | 0.26000  | 0.01363         | -0.01363 | 0.26000          | 0.00000  | 0.00000   | 0.00000  | 0.26000   | 0.00000  |
|                         | 0.28000  | 0.01467         | -0.01467 | 0.28000          | 0.00000  | 0.00000   | 0.00000  | 0.28000   | 0.00000  |
|                         | 0.30000  | 0.01572         | -0.01572 | 0.30000          | 0.00000  | 0.00000   | 0.00000  | 0.30000   | 0.00000  |
|                         | 0.32000  | 0.01677         | -0.01677 | 0.32000          | 0.00000  | 0.00000   | 0.00000  | 0.32000   | 0.00000  |

| SEGMENT NUMBER | PLATFORM AREA | TAPER RATIO | ASPECT RATIO | LEADING EDGE SWEEP | TRAILING EDGE SWEEP | MEAN AERODYNAMIC CHORD | LEADING M.A.C. POSITION | LATERAL M.A.C. POSITION | THICKNESS TO CHORD RATIO |
|----------------|---------------|-------------|--------------|--------------------|---------------------|------------------------|-------------------------|-------------------------|--------------------------|
| 1              | 80.00024      | 0.1038      | 2.81249      | 30.00013           | 0.00000             | 6.5052                 | 3.1583                  | 5.4702                  | 0.08500                  |
| TOTAL          | 80.00024      | 0.1038      | 2.81249      | 30.00013           | 0.00000             | 6.5052                 | 3.1583                  | 5.4702                  | 0.08500                  |

CASE 1  
PAGE 4

THE USAF AUTOMATED MISSILE DATCOM \* REV 6/93 \*  
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS  
FCA #6 ; D = 0

# GEOMETRIC RESULTS FOR FIN SETS

FIN SET NUMBER 1  
(DATA FOR ONE PANEL ONLY)

| SEGMENT NUMBER | PLATFORM AREA | TAPER RATIO | ASPECT RATIO | LEADING EDGE SWEEP | TRAILING EDGE SWEEP | MEAN AERODYNAMIC CHORD | LEADING M.A.C. POSITION | LATERAL M.A.C. POSITION | THICKNESS TO CHORD RATIO |
|----------------|---------------|-------------|--------------|--------------------|---------------------|------------------------|-------------------------|-------------------------|--------------------------|
| 1              | 80.00024      | 0.1038      | 2.81249      | 30.00013           | 0.00000             | 6.5052                 | 3.1583                  | 5.4702                  | 0.08500                  |
| TOTAL          | 80.00024      | 0.1038      | 2.81249      | 30.00013           | 0.00000             | 6.5052                 | 3.1583                  | 5.4702                  | 0.08500                  |

THE USAF AUTOMATED MISSILE DATCOM \* REV 6/93 \*  
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS  
FCA #6 ; D = 0

# FIN SET 1 SECTION AERODYNAMICS

IDEAL ANGLE OF ATTACK = 0.00000 DEG.  
ZERO LIFT ANGLE OF ATTACK = 0.00000 DEG.  
IDEAL LIFT COEFFICIENT = 0.00000  
ZERO LIFT PITCHING MOMENT COEFFICIENT = 0.00000  
MACH ZERO LIFT-CURVE-SLOPE = 0.07768 /DEG.  
LEADING EDGE RADIUS = 0.00000 FRACTION CHORD  
MAXIMUM AIRFOIL THICKNESS = 0.08500 FRACTION CHORD  
DELTA-Y = 0.30657 PERCENT CHORD

CASE 1  
PAGE 5

\*\*\* CREST CRITICAL MACH NUMBER EXCEEDED \*\*\*

CREST CRITICAL MACH = 0.70138

LOCATION = 0.78757 FRACTION CHORD

LIFT-CURVE-SLOPE = 0.07351 /DEG.

THE USAF AUTOMATED MISSILE DATCOM \* REV 6/93 \*

AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS

FCA #6 ; D = 0

BODY ALONE PARTIAL OUTPUT

CASE 1  
PAGE 6

| MACH<br>NUMBER | ALTITUDE<br>FT | FLIGHT CONDITIONS  |                      |                      |                    | REFERENCE DIMENSIONS     |                      |                       |                         |                         |                          |
|----------------|----------------|--------------------|----------------------|----------------------|--------------------|--------------------------|----------------------|-----------------------|-------------------------|-------------------------|--------------------------|
|                |                | VELOCITY<br>FT/SEC | PRESSURE<br>LB/FT**2 | TEMPERATURE<br>DEG R | REYNOLDS<br>NUMBER | SIDESLIP<br>ANGLE<br>DEG | ROLL<br>ANGLE<br>DEG | REF.<br>AREA<br>FT**2 | REF. LENGTH<br>LONG. FT | MOMENT REF.<br>-ONG. FT | CENTER<br>VERTICAL<br>FT |
| 1.31           | 32165.00       | 1290.80            | 5.702E+02            | 404.14               | 3.453E+06          | 0.00                     | 0.00                 | 597.550               | 27.580                  | 190.000                 | 0.000                    |

0 ALPHA CA-FRICTION CA-PRESSURE/WAVE CA-BASE CA-ALPHA

|        |         |         |         |         |
|--------|---------|---------|---------|---------|
| 0.000  | 0.04485 | 0.47067 | 0.34006 | 0.00000 |
| 2.000  | 0.04486 | 0.46881 | 0.34006 | 0.00000 |
| 4.000  | 0.04439 | 0.46324 | 0.34006 | 0.00000 |
| 6.000  | 0.04494 | 0.45398 | 0.34006 | 0.00000 |
| 8.000  | 0.04500 | 0.44108 | 0.34006 | 0.00000 |
| 10.000 | 0.04509 | 0.42460 | 0.34006 | 0.00000 |

0 NOTE - THE BASE DRAG INCREMENT IS NOT INCLUDED IN THE AXIAL FORCE CALCULATIONS

0 CROSS FLOW DRAG PROPORTIONALITY FACTOR = 1.00000

| MACH<br>NUMBER | ALTITUDE<br>FT | FLIGHT CONDITIONS  |                      |                      |                    | REFERENCE DIMENSIONS     |                      |                       |                         |                         |                          |
|----------------|----------------|--------------------|----------------------|----------------------|--------------------|--------------------------|----------------------|-----------------------|-------------------------|-------------------------|--------------------------|
|                |                | VELOCITY<br>FT/SEC | PRESSURE<br>LB/FT**2 | TEMPERATURE<br>DEG R | REYNOLDS<br>NUMBER | SIDESLIP<br>ANGLE<br>DEG | ROLL<br>ANGLE<br>DEG | REF.<br>AREA<br>FT**2 | REF. LENGTH<br>LONG. FT | MOMENT REF.<br>-ONG. FT | CENTER<br>VERTICAL<br>FT |
| 0              | 0.000          | 0.0000             | 0.0000               | 0.0000               | 0.0000             | 0.0000                   | 0.0000               | 1.2000                | 0.0000                  | 0.0000                  | 0.0000                   |
| 2.000          | 0.1093         | 0.0073             | 0.2708               | 0.0114               | 0.0114             | 0.0114                   | 0.0114               | 0.5911                | 0.0114                  | 0.0114                  | 0.0114                   |
| 4.000          | 0.2180         | 0.0268             | 0.5401               | 0.0419               | 0.0419             | 0.0419                   | 0.0419               | 0.5460                | 0.0419                  | 0.0419                  | 0.0419                   |
| 6.000          | 0.3254         | 0.1356             | 0.8063               | 0.2123               | 0.2123             | 0.2123                   | 0.2123               | 1.2308                | 0.2123                  | 0.2123                  | 0.2123                   |
| 8.000          | 0.4309         | 0.2424             | 1.0678               | 0.3795               | 0.3795             | 0.3795                   | 0.3795               | 1.2410                | 0.3795                  | 0.3795                  | 0.3795                   |
| 10.000         | 0.5339         | 0.3804             | 1.3231               | 0.5957               | 0.5957             | 0.5957                   | 0.5957               | 1.2512                | 0.5957                  | 0.5957                  | 0.5957                   |

THE USAF AUTOMATED MISSILE DATCOM \* REV 6/93 \*

AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS

FCA #6 ; D = 0

FIN SET 1 CN PARTIAL OUTPUT

CASE 1  
PAGE 7

| MACH<br>NUMBER | ALTITUDE<br>FT | FLIGHT CONDITIONS  |                      |                      |                    | REFERENCE DIMENSIONS     |                      |                       |                         |                         |                          |
|----------------|----------------|--------------------|----------------------|----------------------|--------------------|--------------------------|----------------------|-----------------------|-------------------------|-------------------------|--------------------------|
|                |                | VELOCITY<br>FT/SEC | PRESSURE<br>LB/FT**2 | TEMPERATURE<br>DEG R | REYNOLDS<br>NUMBER | SIDESLIP<br>ANGLE<br>DEG | ROLL<br>ANGLE<br>DEG | REF.<br>AREA<br>FT**2 | REF. LENGTH<br>LONG. FT | MOMENT REF.<br>-ONG. FT | CENTER<br>VERTICAL<br>FT |
| 1.31           | 32165.00       | 1290.80            | 5.702E+02            | 404.14               | 3.453E+06          | 0.00                     | 0.00                 | 597.560               | 27.580                  | 190.000                 | 0.000                    |

0 POTENTIAL NORMAL FORCE SLOPE AT ALPHA ZERO (1 PANEL), CNA = 0.01100/DEG

0 SECTION CLA = 0.07351/DEG

| PANEL<br>NO. | ALPHA<br>TOTAL<br>DEG | PHI<br>FIN<br>DEG | ALPHA<br>EQUIV<br>DEG | CMAA    | CN        |          | CN<br>TOTAL |
|--------------|-----------------------|-------------------|-----------------------|---------|-----------|----------|-------------|
|              |                       |                   |                       |         | POTENTIAL | VISCIOUS |             |
| 1            | 0.000                 | 0.000             | 0.000                 | 0.45350 | 0.00000   | 0.00000  | 0.00000     |
| 2            | 0.000                 | 90.000            | 0.000                 | 0.45350 | 0.00000   | 0.00000  | 0.00000     |

|     |        |         |        |         |         |         |         |         |
|-----|--------|---------|--------|---------|---------|---------|---------|---------|
| 3   | 0.000  | 180.000 | 0.000  | 0.45350 | 0.00000 | 0.00000 | 0.00000 | 0.00000 |
| 4   | 0.000  | 270.000 | 0.000  | 0.45350 | 0.00000 | 0.00000 | 0.00000 | 0.00000 |
| SET |        |         |        |         |         |         |         |         |
| 1   | 2.000  | 0.000   | 0.000  | 0.45350 | 0.00000 | 0.00000 | 0.00000 | 0.00000 |
| 2   | 2.000  | 90.000  | 2.000  | 0.43595 | 0.02199 | 0.00053 | 0.02252 | 0.00000 |
| 3   | 2.000  | 180.000 | 0.000  | 0.45350 | 0.00000 | 0.00000 | 0.00000 | 0.00000 |
| 4   | 2.000  | 270.000 | 2.000  | 0.43595 | 0.02199 | 0.00053 | 0.02252 | 0.00000 |
| SET |        |         |        |         |         |         |         |         |
| 1   | 4.000  | 0.000   | 0.000  | 0.45350 | 0.00000 | 0.00000 | 0.00000 | 0.00000 |
| 2   | 4.000  | 90.000  | 4.000  | 0.43368 | 0.04387 | 0.00211 | 0.04398 | 0.00000 |
| 3   | 4.000  | 180.000 | 0.000  | 0.45350 | 0.00000 | 0.00000 | 0.00000 | 0.00000 |
| 4   | 4.000  | 270.000 | 4.000  | 0.43368 | 0.04387 | 0.00211 | 0.04598 | 0.00000 |
| SET |        |         |        |         |         |         |         |         |
| 1   | 6.000  | 0.000   | 0.000  | 0.45350 | 0.00000 | 0.00000 | 0.00000 | 0.00000 |
| 2   | 6.000  | 90.000  | 6.000  | 0.43141 | 0.08553 | 0.00871 | 0.07325 | 0.00000 |
| 3   | 6.000  | 180.000 | 0.000  | 0.45350 | 0.00000 | 0.00000 | 0.00000 | 0.00000 |
| 4   | 6.000  | 270.000 | 6.000  | 0.43141 | 0.08553 | 0.00871 | 0.07325 | 0.00000 |
| SET |        |         |        |         |         |         |         |         |
| 1   | 8.000  | 0.000   | 0.000  | 0.45350 | 0.00000 | 0.00000 | 0.00000 | 0.00000 |
| 2   | 8.000  | 90.000  | 8.000  | 0.42914 | 0.08688 | 0.00831 | 0.09519 | 0.00000 |
| 3   | 8.000  | 180.000 | 0.000  | 0.45350 | 0.00000 | 0.00000 | 0.00000 | 0.00000 |
| 4   | 8.000  | 270.000 | 8.000  | 0.42914 | 0.08688 | 0.00831 | 0.09519 | 0.00000 |
| SET |        |         |        |         |         |         |         |         |
| 1   | 10.000 | 0.000   | 0.000  | 0.45350 | 0.00000 | 0.00000 | 0.00000 | 0.00000 |
| 2   | 10.000 | 90.000  | 10.000 | 0.42687 | 0.10780 | 0.01287 | 0.12368 | 0.00000 |
| 3   | 10.000 | 180.000 | 0.000  | 0.45350 | 0.00000 | 0.00000 | 0.00000 | 0.00000 |
| 4   | 10.000 | 270.000 | 10.000 | 0.42687 | 0.10780 | 0.01287 | 0.12368 | 0.00000 |
| SET |        |         |        |         |         |         |         |         |

1

THE USAF AUTOMATED MISSILE DATUM \* REV 5/53 \*  
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS  
FCA #6 ; D = 0  
FIN SET 1 CA PARTIAL OUTPUT

CASE 1  
PAGE 8

| MACH NUMBER | ALTITUDE<br>FT | VELOCITY<br>FT/SEC | PRESSURE<br>LB/FT <sup>2</sup> | TEMPERATURE<br>DEG R | REYNOLDS<br>NUMBER<br>1/FT | SIDESLIP<br>ANGLE<br>DEG | ROLL<br>ANGLE<br>DEG | REFERENCE DIMENSIONS         |                   |              |                               |       |
|-------------|----------------|--------------------|--------------------------------|----------------------|----------------------------|--------------------------|----------------------|------------------------------|-------------------|--------------|-------------------------------|-------|
|             |                |                    |                                |                      |                            |                          |                      | REP. AREA<br>FT <sup>2</sup> | REP. LENGTH<br>FT | MOMENT<br>FT | REF. CENTER<br>VERTICAL<br>FT |       |
| 1.31        | 32165.00       | 1290.80            | 5.702E+02                      | 404.14               | 3.453E+06                  | 0.00                     | 0.00                 | 597.550                      | 27.580            | 27.580       | 190.000                       | 0.000 |

# SINGLE FIN PANEL ZERO-LIFT AXIAL FORCE COMPONENTS

SKIN FRICTION 0.00062  
SUBSONIC PRESSURE 0.00000  
TRANSONIC WAVE 0.00000  
SUPERSONIC WAVE 0.01474  
LEADING EDGE 0.00000  
TRAILING EDGE 0.00000  
TOTAL CNO 0.01536

# FIN AXIAL FORCE DUE TO ANGLE OF ATTACK

ALPHA DEG CA DUE TO ALPHA CA-TOTAL (4 FIN)



0.0000 0.00000 0.05144  
 -0.00032 0.05142  
 4.0000 0.00007 0.05137  
 -0.00017 0.05127  
 8.0000 0.05114  
 -0.00030 0.05098  
 10.0000 -0.00047 0.05098  
 1 THE USAF AUTOMATED MISSILE DATCOM \* REV 6/93 \*  
 AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS  
 FCA #6 ; D = 0  
 FIN SET 1 CM PARTIAL OUTPUT

CASE 1  
PAGE 9

| MACH NUMBER | FLIGHT CONDITIONS |                    |                      |                      | REFERENCE DIMENSIONS |                          |                      |                    |                           |                            |                          |
|-------------|-------------------|--------------------|----------------------|----------------------|----------------------|--------------------------|----------------------|--------------------|---------------------------|----------------------------|--------------------------|
|             | ALTITUDE<br>FT    | VELOCITY<br>FT/SEC | PRESSURE<br>LB/FT**2 | TEMPERATURE<br>DEG R | REYNOLDS<br>NUMBER   | SIDESLIP<br>ANGLE<br>DEG | ROLL<br>ANGLE<br>DEG | REF. AREA<br>FT**2 | REF. LENGTH<br>LAI.<br>FT | MOMENT RES.<br>LONG.<br>FT | CENTER<br>VERTICAL<br>FT |
| 1.31        | 32165.00          | 1250.80            | 5.702E+02            | 404.14               | 3.453E+06            | 0.00                     | 0.00                 | 597.560            | 27.580                    | 190.000                    | 0.000                    |

0 CENTER OF PRESSURE FOR LINEAR CM = -2.31184 (CALIBERS FROM C.G.)  
 CENTER OF PRESSURE FOR NON-LINEAR CM = -2.31367 (CALIBERS FROM C.G.)

| ALPHA<br>DEG. | CM       |            | TOTAL    |
|---------------|----------|------------|----------|
|               | LINEAR   | NON-LINEAR |          |
| 0.00000       | 0.00000  | 0.00000    | 0.00000  |
| 2.00000       | -0.10166 | -0.00246   | -0.10412 |
| 4.00000       | -0.20283 | -0.00977   | -0.21259 |
| 6.00000       | -0.30300 | -0.02181   | -0.32482 |
| 8.00000       | -0.40170 | -0.03846   | -0.44017 |
| 10.00000      | -0.49845 | -0.05956   | -0.55801 |

THE USAF AUTOMATED MISSILE DATCOM \* REV 6/93 \*  
 AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS  
 FCA #6 ; D = 0  
 AERODYNAMIC FORCE AND MOMENT SYNTHESIS

CASE 1  
PAGE 10

| MACH NUMBER | FLIGHT CONDITIONS |                    |                      |                      | REFERENCE DIMENSIONS |                          |                      |                    |                           |                            |                          |
|-------------|-------------------|--------------------|----------------------|----------------------|----------------------|--------------------------|----------------------|--------------------|---------------------------|----------------------------|--------------------------|
|             | ALTITUDE<br>FT    | VELOCITY<br>FT/SEC | PRESSURE<br>LB/FT**2 | TEMPERATURE<br>DEG R | REYNOLDS<br>NUMBER   | SIDESLIP<br>ANGLE<br>DEG | ROLL<br>ANGLE<br>DEG | REF. AREA<br>FT**2 | REF. LENGTH<br>LAI.<br>FT | MOMENT RES.<br>LONG.<br>FT | CENTER<br>VERTICAL<br>FT |
| 1.31        | 32165.00          | 1250.80            | 5.702E+02            | 404.14               | 3.453E+06            | 0.00                     | 0.00                 | 597.560            | 27.580                    | 190.000                    | 0.000                    |

FIN SET 1 IN PRESENCE OF THE BODY

PANEL CHARACTERISTICS  
 PANEL NUMBER  
 AEO (IN PANEL AXIS SYS.)  
 CN

| ANGLE OF<br>ATTACK, DEG. | FIN SET 1 IN PRESENCE OF THE BODY |         |        |        | PANEL CHARACTERISTICS |        |                 |                                   |
|--------------------------|-----------------------------------|---------|--------|--------|-----------------------|--------|-----------------|-----------------------------------|
|                          | CN                                | CH      | CA     | CY     | CLN                   | CLL    | PANEL<br>NUMBER | AEO (IN PANEL AXIS<br>SYS.)<br>CN |
| 0.0000                   | 0.0000                            | 0.0000  | 0.0614 | 0.0000 | 0.0000                | 0.0000 | 1               | 0.0000                            |
| 2.0000                   | 0.0670                            | -0.1549 | 0.0614 | 0.0000 | 0.0000                | 0.0000 | 2               | 0.0000                            |
| 4.0000                   | 0.1376                            | -0.3182 | 0.0613 | 0.0000 | 0.0000                | 0.0000 | 3               | 0.0000                            |
| 6.0000                   | 0.2190                            | -0.5063 | 0.0611 | 0.0000 | 0.0000                | 0.0000 | 4               | 0.0000                            |

|   |          |         |
|---|----------|---------|
| 2 | 9.1281   | 0.1095  |
| 3 | 0.0000   | 0.0000  |
| 4 | -9.1281  | -0.1095 |
| 1 | 0.0000   | 0.0000  |
| 2 | 11.7303  | 0.1430  |
| 3 | 0.0000   | 0.0000  |
| 4 | -11.7303 | -0.1430 |
| 1 | 0.0000   | 0.0000  |
| 2 | 13.7341  | 0.1692  |
| 3 | 0.0000   | 0.0000  |
| 4 | -13.7341 | -0.1692 |

CASE 1  
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THE USAF AUTOMATED MISSILE DATCOM \* REV 6/93 \*  
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS  
FCA #6 ; D = 0  
AERODYNAMIC FORCE AND MOMENT SYNTHESIS

| MACH NUMBER | FLIGHT CONDITIONS |                    |                      | REFERENCE DIMENSIONS |                    |                                                          |
|-------------|-------------------|--------------------|----------------------|----------------------|--------------------|----------------------------------------------------------|
|             | ALTITUDE<br>FT    | VELOCITY<br>FT/SEC | PRESSURE<br>LB/FT**2 | TEMPERATURE<br>DEG R | REF. AREA<br>FT**2 | MOMENT REF. CENTER<br>LONG. FT<br>LAT. FT<br>VERTICAL FT |
| 1.31        | 32165.00          | 1290.80            | 5.702E+02            | 404.14               | 597.560            | 27.580 190.000 0.000                                     |

| ANGLE OF<br>ATTACK, DEG. | SYNTHESIS AERODYNAMICS FOR BODY-FIN SET 1 |        |        | REFERENCE DIMENSIONS |                    |                                                          |
|--------------------------|-------------------------------------------|--------|--------|----------------------|--------------------|----------------------------------------------------------|
|                          | CM                                        | CA     | CY     | CIM                  | REF. AREA<br>FT**2 | MOMENT REF. CENTER<br>LONG. FT<br>LAT. FT<br>VERTICAL FT |
| 0.0000                   | 0.0000                                    | 0.5770 | 0.0000 | 0.0000               | 597.560            | 27.580 190.000 0.000                                     |
| 2.0000                   | 0.1900                                    | 0.1125 | 0.0000 | 0.0000               | 597.560            | 27.580 190.000 0.000                                     |
| 4.0000                   | 0.3956                                    | 0.2333 | 0.0000 | 0.0000               | 597.560            | 27.580 190.000 0.000                                     |
| 6.0000                   | 0.7013                                    | 0.4630 | 0.0000 | 0.0000               | 597.560            | 27.580 190.000 0.000                                     |
| 8.0000                   | 0.9885                                    | 0.7184 | 0.0000 | 0.0000               | 597.560            | 27.580 190.000 0.000                                     |
| 10.0000                  | 1.2899                                    | 1.0506 | 0.0000 | 0.0000               | 597.560            | 27.580 190.000 0.000                                     |

CASE 1  
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THE USAF AUTOMATED MISSILE DATCOM \* REV 6/93 \*  
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS  
FCA #6 ; D = 0  
AERODYNAMIC FORCE AND MOMENT SYNTHESIS

| MACH NUMBER | FLIGHT CONDITIONS |                    |                      | REFERENCE DIMENSIONS |                    |                                                          |
|-------------|-------------------|--------------------|----------------------|----------------------|--------------------|----------------------------------------------------------|
|             | ALTITUDE<br>FT    | VELOCITY<br>FT/SEC | PRESSURE<br>LB/FT**2 | TEMPERATURE<br>DEG R | REF. AREA<br>FT**2 | MOMENT REF. CENTER<br>LONG. FT<br>LAT. FT<br>VERTICAL FT |
| 1.31        | 32165.00          | 1290.90            | 5.702E+02            | 404.14               | 597.560            | 27.580 190.000 0.000                                     |

| ALPHA   | CARRYOVER INTERFERENCE FACTORS - FIN SET 1 |        |         | REFERENCE DIMENSIONS |                    |                                                          |
|---------|--------------------------------------------|--------|---------|----------------------|--------------------|----------------------------------------------------------|
|         | K-W(B)                                     | K-B(W) | KK-W(B) | KK-B(W)              | REF. AREA<br>FT**2 | MOMENT REF. CENTER<br>LONG. FT<br>LAT. FT<br>VERTICAL FT |
| 0.0000  | 1.4738                                     | 0.1412 | 0.9413  | 0.5326               | 2.3133             | 0.4137                                                   |
| 2.0000  | 1.3719                                     | 0.1412 | 0.9413  | 0.5326               | 2.3133             | 0.4100                                                   |
| 4.0000  | 1.3092                                     | 0.1412 | 0.9413  | 0.5326               | 2.3133             | 0.4061                                                   |
| 6.0000  | 1.2896                                     | 0.1412 | 0.9413  | 0.5326               | 2.3133             | 0.4015                                                   |
| 8.0000  | 1.2896                                     | 0.1412 | 0.9413  | 0.5326               | 2.3133             | 0.3979                                                   |
| 10.0000 | 1.2896                                     | 0.1412 | 0.9413  | 0.5326               | 2.3133             | 0.3950                                                   |

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THE USAF AUTOMATED MISSILE DATCOM \* REV 6/93 \*  
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS  
FCA #6 ; D = 0  
FIN SET 1 PANEL BENDING MOMENTS (ABOUT EXPOSED ROOT CHORD)

| MACH<br>NUMBER                                   | ALTITUDE<br>FT | FLIGHT CONDITIONS  |                                |                      |                            | REFERENCE DIMENSIONS     |                      |                                 |                       |
|--------------------------------------------------|----------------|--------------------|--------------------------------|----------------------|----------------------------|--------------------------|----------------------|---------------------------------|-----------------------|
|                                                  |                | VELOCITY<br>FT/SEC | PRESSURE<br>LB/FT <sup>2</sup> | TEMPERATURE<br>DEG R | REYNOLDS<br>NUMBER<br>1/FT | SIDESLIP<br>ANGLE<br>DEG | ROLL<br>ANGLE<br>DEG | REF.<br>AREA<br>FT <sup>2</sup> | MOMENT<br>LONG.<br>FT |
| 1.31                                             | 32165.00       | 1290.80            | 5.702E+02                      | 404.14               | 3.453E+06                  | 0.00                     | 0.00                 | 597.560                         | 190.000               |
| ANGLE OF ATTACK, DEG.                            |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| PANEL 1                                          |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| PANEL 2                                          |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| PANEL 3                                          |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| PANEL 4                                          |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| 0.0000                                           |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| 2.0000                                           |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| 4.0000                                           |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| 6.0000                                           |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| 8.0000                                           |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| 10.0000                                          |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| THE USAF AUTOMATED MISSILE DATCON * REV 6/83 *   |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS   |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| FCA #6, D = 0                                    |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| FIN SET 1 PANEL HINGE MOMENTS (ABOUT HINGE LINE) |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| CASE 1                                           |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| PAGE 1A                                          |                |                    |                                |                      |                            |                          |                      |                                 |                       |

| MACH<br>NUMBER                                 | ALTITUDE<br>FT | FLIGHT CONDITIONS  |                                |                      |                            | REFERENCE DIMENSIONS     |                      |                                 |                       |
|------------------------------------------------|----------------|--------------------|--------------------------------|----------------------|----------------------------|--------------------------|----------------------|---------------------------------|-----------------------|
|                                                |                | VELOCITY<br>FT/SEC | PRESSURE<br>LB/FT <sup>2</sup> | TEMPERATURE<br>DEG R | REYNOLDS<br>NUMBER<br>1/FT | SIDESLIP<br>ANGLE<br>DEG | ROLL<br>ANGLE<br>DEG | REF.<br>AREA<br>FT <sup>2</sup> | MOMENT<br>LONG.<br>FT |
| 1.31                                           | 32165.00       | 1290.80            | 5.702E+02                      | 404.14               | 3.453E+06                  | 0.00                     | 0.00                 | 597.560                         | 190.000               |
| ANGLE OF ATTACK, DEG.                          |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| PANEL 1                                        |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| PANEL 2                                        |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| PANEL 3                                        |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| PANEL 4                                        |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| 0.0000                                         |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| 2.0000                                         |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| 4.0000                                         |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| 6.0000                                         |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| 8.0000                                         |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| 10.0000                                        |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| THE USAF AUTOMATED MISSILE DATCON * REV 6/83 * |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| FCA #6, D = 0                                  |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| BODY ALONE STATIC AERODYNAMIC CHARACTERISTIC   |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| CASE 1                                         |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| PAGE 15                                        |                |                    |                                |                      |                            |                          |                      |                                 |                       |

| MACH<br>NUMBER                                 | ALTITUDE<br>FT | FLIGHT CONDITIONS  |                                |                      |                            | REFERENCE DIMENSIONS     |                      |                                 |                       |
|------------------------------------------------|----------------|--------------------|--------------------------------|----------------------|----------------------------|--------------------------|----------------------|---------------------------------|-----------------------|
|                                                |                | VELOCITY<br>FT/SEC | PRESSURE<br>LB/FT <sup>2</sup> | TEMPERATURE<br>DEG R | REYNOLDS<br>NUMBER<br>1/FT | SIDESLIP<br>ANGLE<br>DEG | ROLL<br>ANGLE<br>DEG | REF.<br>AREA<br>FT <sup>2</sup> | MOMENT<br>LONG.<br>FT |
| 1.31                                           | 32165.00       | 1290.80            | 5.702E+02                      | 404.14               | 3.453E+06                  | 0.00                     | 0.00                 | 597.560                         | 190.000               |
| ANGLE OF ATTACK, DEG.                          |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| PANEL 1                                        |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| PANEL 2                                        |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| PANEL 3                                        |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| PANEL 4                                        |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| 0.0000                                         |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| 2.0000                                         |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| 4.0000                                         |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| 6.0000                                         |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| 8.0000                                         |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| 10.0000                                        |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| THE USAF AUTOMATED MISSILE DATCON * REV 6/83 * |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| FCA #6, D = 0                                  |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| BODY ALONE STATIC AERODYNAMIC CHARACTERISTIC   |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| CASE 1                                         |                |                    |                                |                      |                            |                          |                      |                                 |                       |
| PAGE 15                                        |                |                    |                                |                      |                            |                          |                      |                                 |                       |

| ALPHA | CN    | LONGITUDINAL |       |       |       | DERIVATIVES (PER DEGREE) |           |            |           |
|-------|-------|--------------|-------|-------|-------|--------------------------|-----------|------------|-----------|
|       |       | CK           | CA    | CY    | CLM   | CNA                      | CYR       | CLNE       | CLLE      |
| 0.00  | 0.000 | 0.000        | 0.516 | 0.000 | 0.000 | 5.537E-02                | 1.357E-01 | -1.389E-01 | 0.000E+00 |
| 2.00  | 0.117 | 0.282        | 0.514 | 0.000 | 0.000 | 6.119E-03                | 1.455E-01 | -1.412E-01 | 0.000E+00 |
| 4.00  | 0.245 | 0.582        | 0.508 | 0.000 | 0.000 | 8.506E-03                | 1.839E-01 | -1.465E-01 | 0.000E+00 |
| 6.00  | 0.461 | 1.019        | 0.499 | 0.000 | 0.000 | 1.071E-01                | 2.153E-01 | -1.698E-01 | 0.000E+00 |

8.00 0.673 1.447 0.486 0.000 0.000 0.000 1.133E-01 2.250E-01 -8.383E-02 -1.801E-01 0.000E+00  
 10.00 0.914 1.919 0.470 0.000 0.000 0.000 1.277E-01 2.465E-01 -9.068E-02 -1.902E-01 0.000E+00

0 LINEAR DATA FOR BODY ALONE WAS GENERATED USING THE SECOND-ORDER SHOCK EXPANSION METHOD  
 1 THE USAF AUTOMATED MISSILE DATCOM \* REV 6/93 \*  
 AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS  
 FCA #6 ; D = 0  
 FIN 1 ALONE STATIC AERODYNAMIC CHARACTERISTICS

CASE 1  
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| MACH NUMBER | ALTITUDE FT | VELOCITY FT/SEC | PRESSURE LB/FT**2 | FLIGHT CONDITIONS |           | TEMPERATURE DEG R | REYNOLDS NUMBER 1/FT | SIDESLIP ANGLE DEG | ROLL ANGLE DEG | REF. AREA FT**2 | REFERENCE DIMENSIONS |                |         |                 |
|-------------|-------------|-----------------|-------------------|-------------------|-----------|-------------------|----------------------|--------------------|----------------|-----------------|----------------------|----------------|---------|-----------------|
|             |             |                 |                   | CY                | CLN       |                   |                      |                    |                |                 | CLL                  | REF. LENGTH FT | LAT. FT | MOMENT LONG. FT |
| 1.31        | 32165.00    | 1290.80         | 5.702E+02         | 404.14            | 3.453E+06 | 0.00              | 0.00                 | 0.00               | 0.00           | 597.560         | 27.580               | 27.580         | 190.000 | 0.000           |

| ALPHA | CN    | LONGITUDINAL CN | CA    | LATERAL DIRECTIONAL |       | CY    | CLN   | CLL   | CNA       | CMA        | DERIVATIVES (PER DEGREE) |      |
|-------|-------|-----------------|-------|---------------------|-------|-------|-------|-------|-----------|------------|--------------------------|------|
|       |       |                 |       | CLN                 | CLL   |       |       |       |           |            | LATERAL CLNE             | CLLB |
| 0.00  | 0.000 | 0.000           | 0.061 | 0.000               | 0.061 | 0.000 | 0.000 | 0.000 | 2.205E-02 | -5.097E-02 |                          |      |
| 2.00  | 0.045 | -0.104          | 0.061 | 0.043               | 0.063 | 0.000 | 0.000 | 0.000 | 2.299E-02 | -5.315E-02 |                          |      |
| 4.00  | 0.092 | -0.213          | 0.061 | 0.087               | 0.068 | 0.000 | 0.000 | 0.000 | 2.386E-02 | -5.517E-02 |                          |      |
| 6.00  | 0.140 | -0.325          | 0.061 | 0.133               | 0.076 | 0.000 | 0.000 | 0.000 | 2.461E-02 | -5.689E-02 |                          |      |
| 8.00  | 0.190 | -0.440          | 0.061 | 0.180               | 0.087 | 0.000 | 0.000 | 0.000 | 2.521E-02 | -5.830E-02 |                          |      |
| 10.00 | 0.241 | -0.558          | 0.061 | 0.227               | 0.102 | 0.000 | 0.000 | 0.000 | 2.575E-02 | -5.954E-02 |                          |      |

CASE 1  
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1 THE USAF AUTOMATED MISSILE DATCOM \* REV 6/93 \*  
 AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS  
 FCA #6 ; D = 0  
 STATIC AERODYNAMICS FOR BODY-FIN SET 1

| MACH NUMBER | ALTITUDE FT | VELOCITY FT/SEC | PRESSURE LB/FT**2 | FLIGHT CONDITIONS |           | TEMPERATURE DEG R | REYNOLDS NUMBER 1/FT | SIDESLIP ANGLE DEG | ROLL ANGLE DEG | REF. AREA FT**2 | REFERENCE DIMENSIONS |                |         |                 |
|-------------|-------------|-----------------|-------------------|-------------------|-----------|-------------------|----------------------|--------------------|----------------|-----------------|----------------------|----------------|---------|-----------------|
|             |             |                 |                   | CY                | CLN       |                   |                      |                    |                |                 | CLL                  | REF. LENGTH FT | LAT. FT | MOMENT LONG. FT |
| 1.31        | 32165.00    | 1290.80         | 5.702E+02         | 404.14            | 3.453E+06 | 0.00              | 0.00                 | 0.00               | 0.00           | 597.560         | 27.580               | 27.580         | 190.000 | 0.000           |

| ALPHA | CN    | LONGITUDINAL CN | CA    | LATERAL DIRECTIONAL |       | CY    | CLN   | CLL   | CNA       | CMA        | DERIVATIVES (PER DEGREE) |      |
|-------|-------|-----------------|-------|---------------------|-------|-------|-------|-------|-----------|------------|--------------------------|------|
|       |       |                 |       | CLN                 | CLL   |       |       |       |           |            | LATERAL CLNE             | CLLB |
| 0.00  | 0.000 | 0.000           | 0.061 | 0.000               | 0.061 | 0.000 | 0.000 | 0.000 | 2.205E-02 | -5.097E-02 |                          |      |
| 2.00  | 0.045 | -0.104          | 0.061 | 0.043               | 0.063 | 0.000 | 0.000 | 0.000 | 2.299E-02 | -5.315E-02 |                          |      |
| 4.00  | 0.092 | -0.213          | 0.061 | 0.087               | 0.068 | 0.000 | 0.000 | 0.000 | 2.386E-02 | -5.517E-02 |                          |      |
| 6.00  | 0.140 | -0.325          | 0.061 | 0.133               | 0.076 | 0.000 | 0.000 | 0.000 | 2.461E-02 | -5.689E-02 |                          |      |
| 8.00  | 0.190 | -0.440          | 0.061 | 0.180               | 0.087 | 0.000 | 0.000 | 0.000 | 2.521E-02 | -5.830E-02 |                          |      |
| 10.00 | 0.241 | -0.558          | 0.061 | 0.227               | 0.102 | 0.000 | 0.000 | 0.000 | 2.575E-02 | -5.954E-02 |                          |      |

|       |       |       |       |       |       |       |       |           |           |
|-------|-------|-------|-------|-------|-------|-------|-------|-----------|-----------|
| 0.00  | 0.000 | 0.000 | 0.577 | 0.000 | 0.000 | 0.000 | 0.030 | 9.107E-02 | 5.415E-02 |
| 2.00  | 0.190 | 0.112 | 0.575 | 0.000 | 0.000 | 0.000 | 0.000 | 9.39E-01  | 5.83E-02  |
| 4.00  | 0.396 | 0.233 | 0.569 | 0.000 | 0.000 | 0.000 | 0.000 | 1.278E-01 | 8.755E-02 |
| 6.00  | 0.701 | 0.463 | 0.560 | 0.000 | 0.000 | 0.000 | 0.000 | 2.482E-01 | 1.213E-01 |
| 8.00  | 0.989 | 0.718 | 0.547 | 0.000 | 0.000 | 0.000 | 0.000 | 4.71E-01  | 1.469E-01 |
| 10.00 | 1.290 | 1.051 | 0.530 | 0.000 | 0.000 | 0.000 | 0.000 | 1.542E-01 | 1.854E-01 |

ALPHA CL CD CL/CD X-C.F.

|       |       |       |       |       |
|-------|-------|-------|-------|-------|
| 0.00  | 0.000 | 0.577 | 0.000 | 0.595 |
| 2.00  | 0.170 | 0.583 | 0.212 | 0.592 |
| 4.00  | 0.355 | 0.596 | 0.536 | 0.590 |
| 6.00  | 0.639 | 0.630 | 1.014 | 0.660 |
| 8.00  | 0.903 | 0.679 | 1.323 | 0.737 |
| 10.00 | 1.178 | 0.746 | 1.579 | 0.815 |

FIN SET FIN 1 FIN 2 FIN 3 FIN 4  
1 0.00 0.00 0.00 0.00

0 LINEAR DATA FOR BODY ALONE WAS GENERATED USING THE SECOND-ORDER SHOCK EXPANSION METHOD  
1 THE USAF AUTOMATED MISSILE DATCOM \* REV 6/93 \*  
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS  
FCA #6 ; D = 0

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BODY ALONE DYNAMIC DERIVATIVES

| MACH<br>NUMBER | ALTITUDE<br>FT | VELOCITY<br>FT/SEC | PRESSURE<br>LB/FT**2 | FLIGHT CONDITIONS    |           | REYNOLDS<br>NUMBER<br>1/FT | SIDESLIP<br>ANGLE<br>DEG | ROLL<br>ANGLE<br>DEG | REFERENCE DIMENSIONS  |                   |            |                       |                               |
|----------------|----------------|--------------------|----------------------|----------------------|-----------|----------------------------|--------------------------|----------------------|-----------------------|-------------------|------------|-----------------------|-------------------------------|
|                |                |                    |                      | TEMPERATURE<br>DEG R | DEG R     |                            |                          |                      | REF.<br>AREA<br>FT**2 | REF. LENGTH<br>FT | LAT.<br>FT | MOMENT<br>LONG.<br>FT | REF. CENTER<br>VERTICAL<br>FT |
| 1.31           | 32165.00       | 1290.80            | 5.702E+02            | 404.14               | 3.453E+06 | 3.00                       | 0.00                     | 0.00                 | 597.560               | 27.580            | 27.580     | 190.000               | 0.000                         |

ALPHA DYNAMIC DERIVATIVES (PER DEGREE) CNQ+CNAD

|      |           |           |              |
|------|-----------|-----------|--------------|
| 0.0  | 1.345E-01 | 2.656E-01 | -3.19300E-01 |
| 2.0  | 1.486E-01 | 2.935E-01 | -3.52859E-01 |
| 4.0  | 2.091E-01 | 4.128E-01 | -4.96316E-01 |
| 6.0  | 2.603E-01 | 5.138E-01 | -6.17822E-01 |
| 8.0  | 2.753E-01 | 5.436E-01 | -6.53543E-01 |
| 10.0 | 3.103E-01 | 6.127E-01 | -7.36703E-01 |

THE USAF AUTOMATED MISSILE DATCOM \* REV 6/93 \*  
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS  
FCA #6 ; D = 0

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FIN SET 1 DYNAMIC DERIVATIVES

| MACH<br>NUMBER | ALTITUDE<br>FT | VELOCITY<br>FT/SEC | PRESSURE<br>LB/FT**2 | TEMPERATURE<br>DEG R | REYNOLDS<br>NUMBER<br>1/FT | SIDESLIP<br>ANGLE<br>DEG | ROLL<br>ANGLE<br>DEG | REFERENCE DIMENSIONS |                   |            |                       |                               |       |
|----------------|----------------|--------------------|----------------------|----------------------|----------------------------|--------------------------|----------------------|----------------------|-------------------|------------|-----------------------|-------------------------------|-------|
|                |                |                    |                      |                      |                            |                          |                      | REF. AREA<br>FT**2   | REF. LENGTH<br>FT | LAT.<br>FT | MOMENT<br>LONG.<br>FT | REF. CENTER<br>VERTICAL<br>FT |       |
| 1.31           | 32165.00       | 1290.80            | 5.702E+02            | 404.14               | 3.453E+06                  | 0.00                     | 0.00                 | 0.00                 | 597.560           | 27.580     | 27.580                | 190.000                       | 0.000 |

ALPHA DYNAMIC DERIVATIVES (PER DEGREE) CNQ

|     |           |            |             |           |
|-----|-----------|------------|-------------|-----------|
| 0.0 | 2.812E-02 | -3.601E-03 | -6.5651E-02 | 8.452E-03 |
|-----|-----------|------------|-------------|-----------|

2.0 2.932E-02 -3.754E-03 -6.845E-02 8.813E-03  
 4.0 3.044E-02 -3.897E-03 -7.106E-02 9.149E-03  
 6.0 3.139E-02 -4.019E-03 -7.327E-02 9.434E-03  
 8.0 3.216E-02 -4.118E-03 -7.508E-02 9.667E-03  
 10.0 3.285E-02 -4.206E-03 -7.658E-02 9.873E-03

THE USAF AUTOMATED MISSILE DATCOM + REV 5/93 +  
 AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS  
 FCA #6 ; D = 0  
 BODY + 1 FIN SET DYNAMIC DERIVATIVES

CASE 1  
 PAGE 20

| MACH<br>NUMBER | ALTITUDE<br>FT | VELOCITY<br>FT/SEC | FLIGHT CONDITIONS    |                      | REYNOLDS<br>NUMBER | SIDESLIP     |              | ROLL<br>ANGLE<br>DEG | REFERENCE DIMENSIONS  |                            |                       |                               |
|----------------|----------------|--------------------|----------------------|----------------------|--------------------|--------------|--------------|----------------------|-----------------------|----------------------------|-----------------------|-------------------------------|
|                |                |                    | PRESSURE<br>LB/FT**2 | TEMPERATURE<br>DEG R |                    | ANGLE<br>DEG | ANGLE<br>DEG |                      | REF.<br>AREA<br>FT**2 | REF. LENGTH<br>LONG.<br>FT | MOMENT<br>LONG.<br>FT | CENTER<br>REF. VERTICAL<br>FT |
| 1.31           | 32165.00       | 1290.80            | 5.702E+02            | 404.14               | 3.453E+06          | 0.00         | 0.00         | 0.00                 | 597.550               | 27.580                     | 190.000               | 0.000                         |

| DYNAMIC DERIVATIVES (PER DEGREE) |           |           |           | CINQ+CHAD    |  |
|----------------------------------|-----------|-----------|-----------|--------------|--|
| CINQ                             |           |           |           | CINQ+CHAD    |  |
| ALPHA                            |           |           |           |              |  |
| 0.0                              | 1.747E-01 | 2.604E-01 | 2.831E-01 | -4.01141E-01 |  |
| 2.0                              | 1.906E-01 | 2.831E-01 | 2.831E-01 | -4.38194E-01 |  |
| 4.0                              | 2.526E-01 | 4.072E-01 | 4.072E-01 | -5.84902E-01 |  |
| 6.0                              | 3.052E-01 | 5.081E-01 | 5.081E-01 | -7.09166E-01 |  |
| 8.0                              | 3.214E-01 | 5.377E-01 | 5.377E-01 | -7.47240E-01 |  |
| 10.0                             | 3.573E-01 | 6.067E-01 | 6.067E-01 | -8.32297E-01 |  |

1\*\*\* END OF JOB \*\*\*

## **D.2 Raw Dynamic Stability Wind Tunnel Test Data**

11 Dec 1995 CDDF AERO CONTROL TEST 10:20:53

TWT0747 Run 6/1

FLOW CONDITONS

| Fr # | P-0   | P-inf | Q-inf | M-inf | T-0 | Rn/ft   | BLANK |
|------|-------|-------|-------|-------|-----|---------|-------|
| 1    | 22.05 | 9.07  | 9.17  | 1.202 | 552 | 6.8E+06 | 0.00  |
| 2    | 22.03 | 9.08  | 9.16  | 1.200 | 552 | 6.7E+06 | 0.00  |
| 3    | 22.04 | 9.09  | 9.16  | 1.200 | 552 | 6.7E+06 | 0.00  |
| 4    | 22.04 | 9.09  | 9.16  | 1.200 | 552 | 6.7E+06 | 0.00  |

| Fr # | T-inf   | Rho-inf | V-inf    | W-dot | a-0      | Time     |
|------|---------|---------|----------|-------|----------|----------|
| 1    | 428.240 | .057    | 1218.918 | 4.569 | 1014.480 | 10:20:36 |
| 2    | 428.652 | .057    | 1218.338 | 4.565 | 1014.968 | 10:20:39 |
| 3    | 428.731 | .057    | 1217.920 | 4.567 | 1015.061 | 10:20:42 |
| 4    | 428.638 | .057    | 1217.772 | 4.566 | 1014.951 | 10:20:45 |

P-static offset, in percent of P-0 -1.59

|             |             |
|-------------|-------------|
| Test name   | TWT0747     |
| Label Set   | LTWT0747A   |
| Engineer    | FROST       |
| Contractor  | NONE        |
| Tot frames  | 4           |
| Tot columns | 12          |
| Tot headers | 35          |
| Time        | 10:19:20    |
| Date        | 11 Dec 1995 |
| Basic Prog  | TWT0747A    |
| Run         | 6           |
| Rerun       | 1           |
| Config      | No Fins     |
| Pbar-mmHg.  | 758.0       |
| BLANK       | 0           |
| Machset     | 1.20        |
| No. angles  | 4           |



13 Dec 1995 CDDF AERO CONTROL TEST 11:46:30

TWT0747 Run 51/0

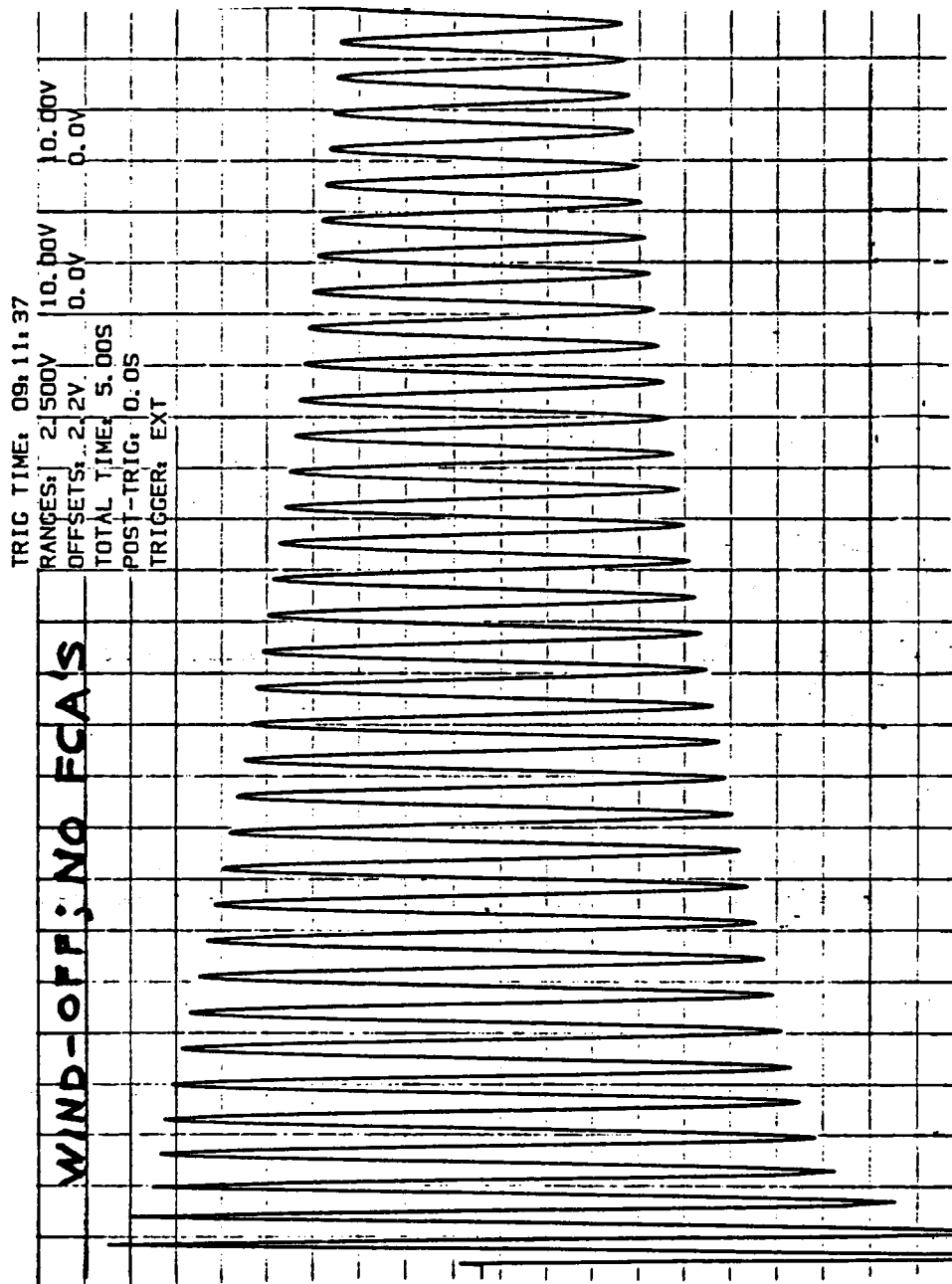
# FLOW CONDITONS

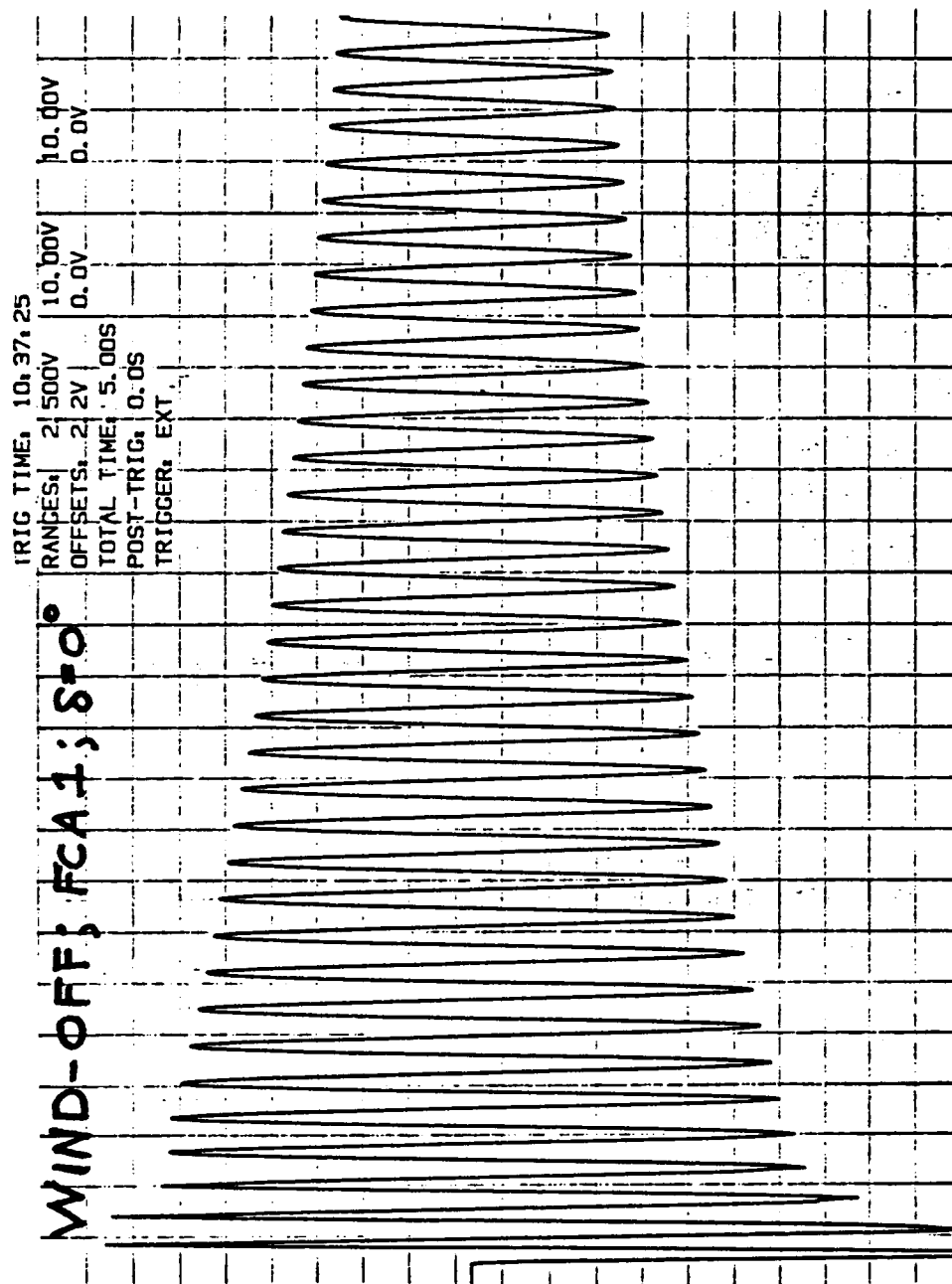
| Fr # | P-0   | P-inf | Q-inf | M-inf | T-0 | Rn/ft   | BLANK |
|------|-------|-------|-------|-------|-----|---------|-------|
| 1    | 22.03 | 6.34  | 9.48  | 1.462 | 562 | 6.4E+06 | 0.00  |
| 2    | 22.04 | 6.35  | 9.49  | 1.461 | 562 | 6.4E+06 | 0.00  |
| 3    | 22.02 | 6.35  | 9.48  | 1.461 | 562 | 6.4E+06 | 0.00  |

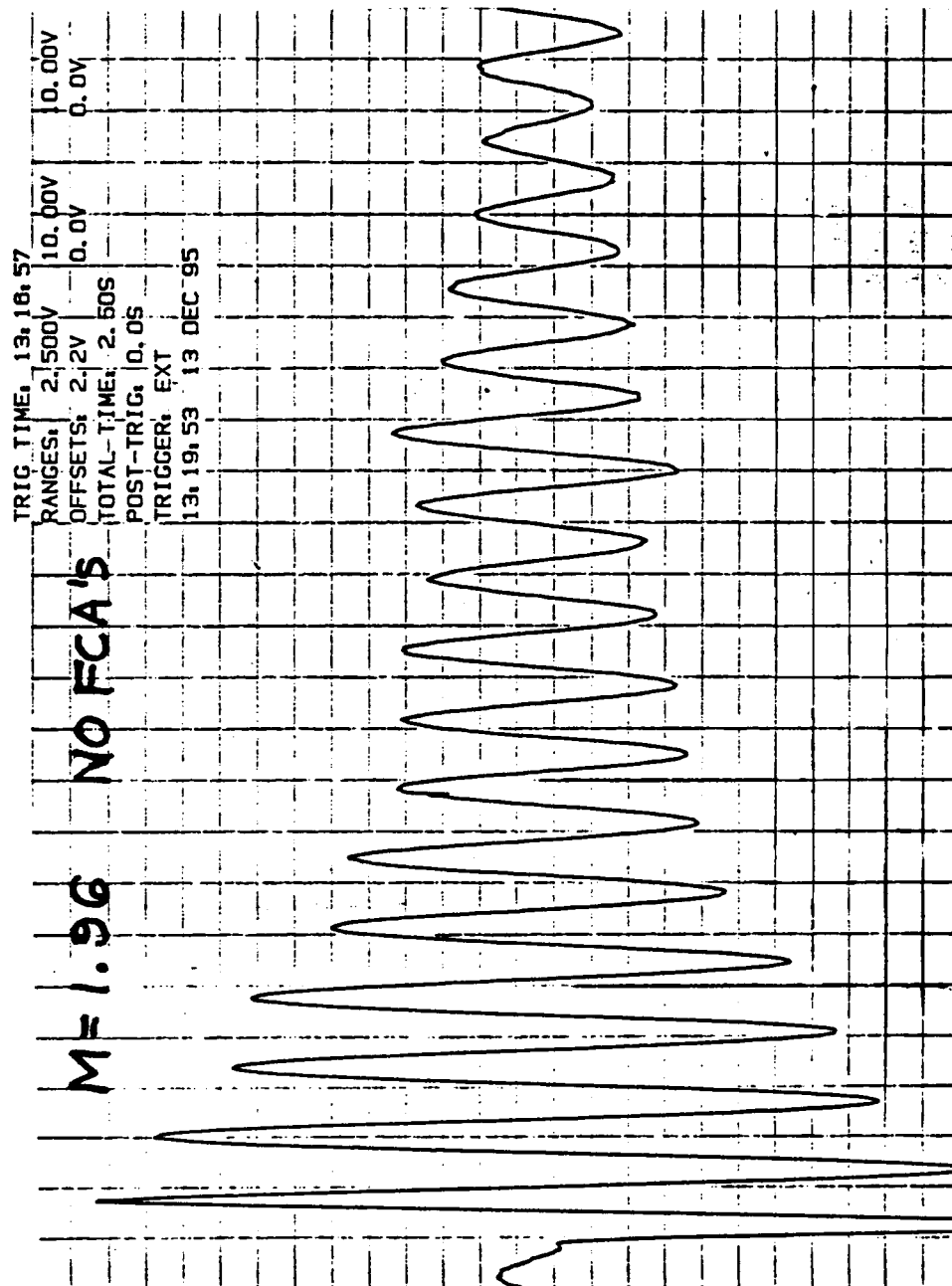
| Fr # | T-inf   | Rho-inf | V-inf    | W-dot | a-0     | Time     |
|------|---------|---------|----------|-------|---------|----------|
| 1    | 393.717 | .043    | 1421.694 | 4.524 | 972.730 | 11:46:12 |
| 2    | 393.920 | .043    | 1421.918 | 4.524 | 972.980 | 11:46:15 |
| 3    | 394.058 | .043    | 1421.624 | 4.520 | 973.151 | 11:46:18 |

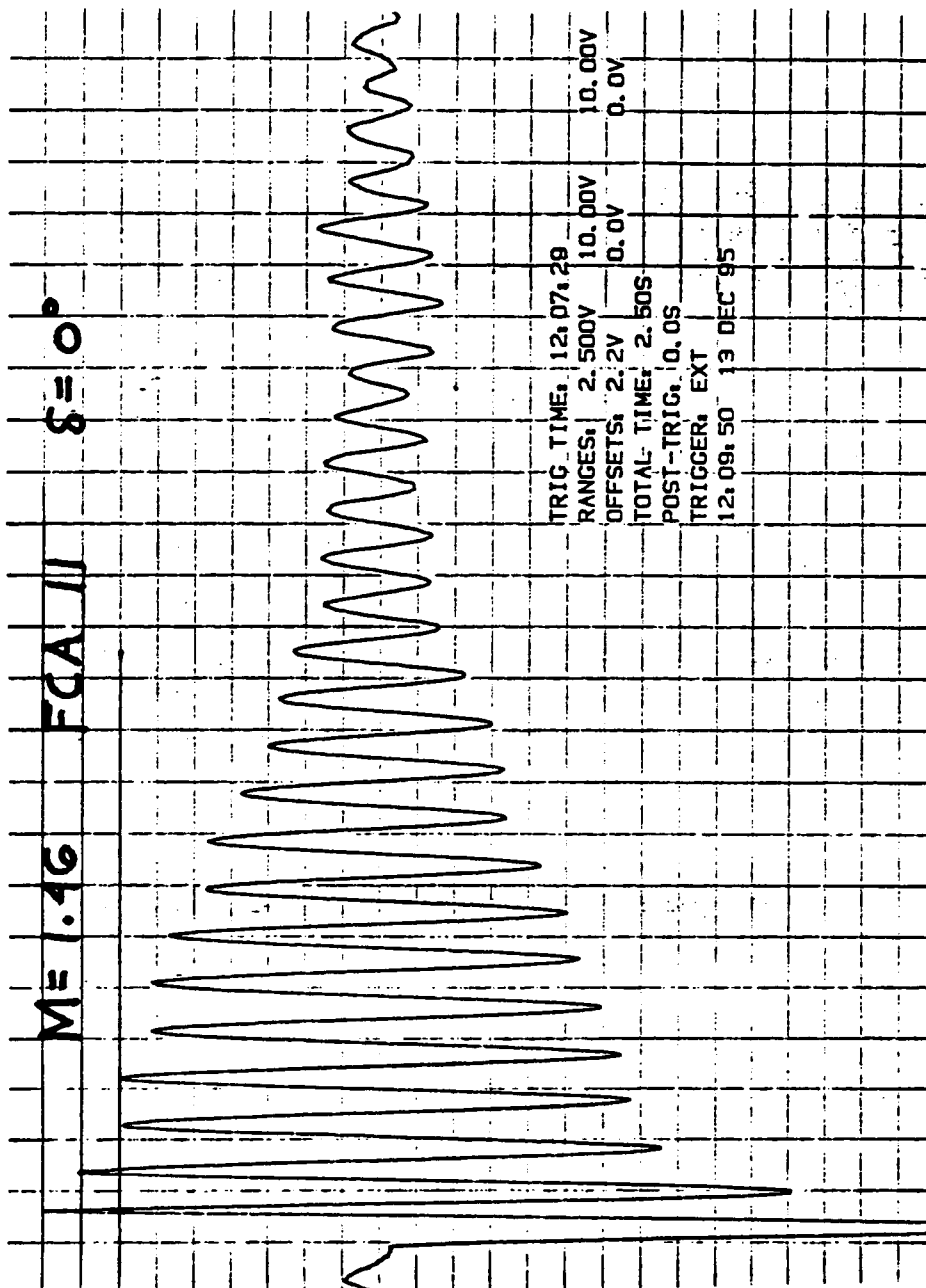
P-static offset, in percent of P-0 .05

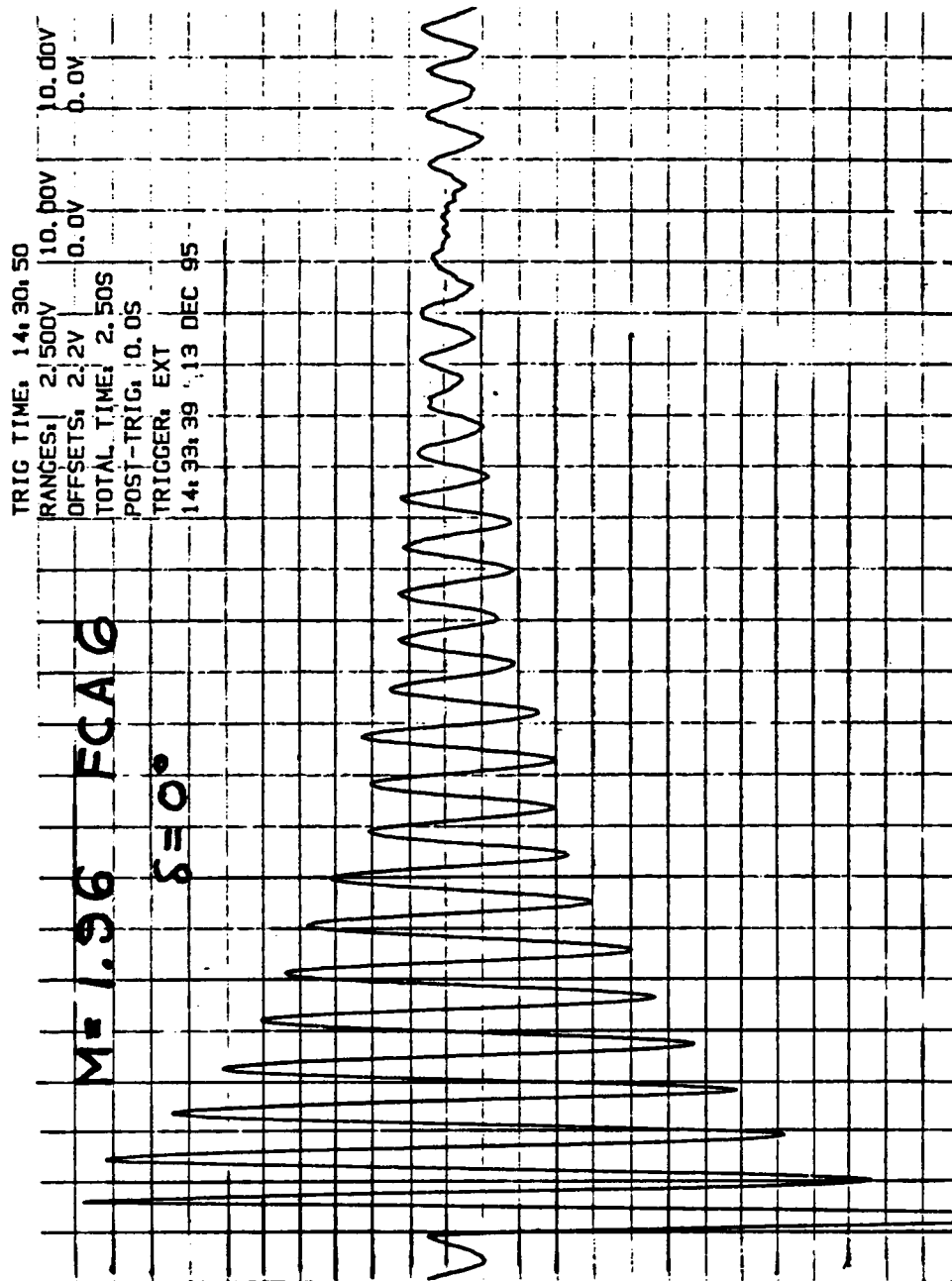
|             |             |
|-------------|-------------|
| Test name   | TWT0747     |
| Label Set   | LTWT0747A   |
| Engineer    | FROST       |
| Contractor  | NONE        |
| Tot frames  | 3           |
| Tot columns | 12          |
| Tot headers | 35          |
| Time        | 11:45:34    |
| Date        | 13 Dec 1995 |
| Basic Prog  | TWT0747A    |
| Run         | 51          |
| Rerun       | 0           |
| Config      | Fin 6 d=15  |
| Pbar-mmHg.  | 751.5       |
| BLANK       | 0           |
| Machset     | 1.46        |
| No. angles  | 3           |

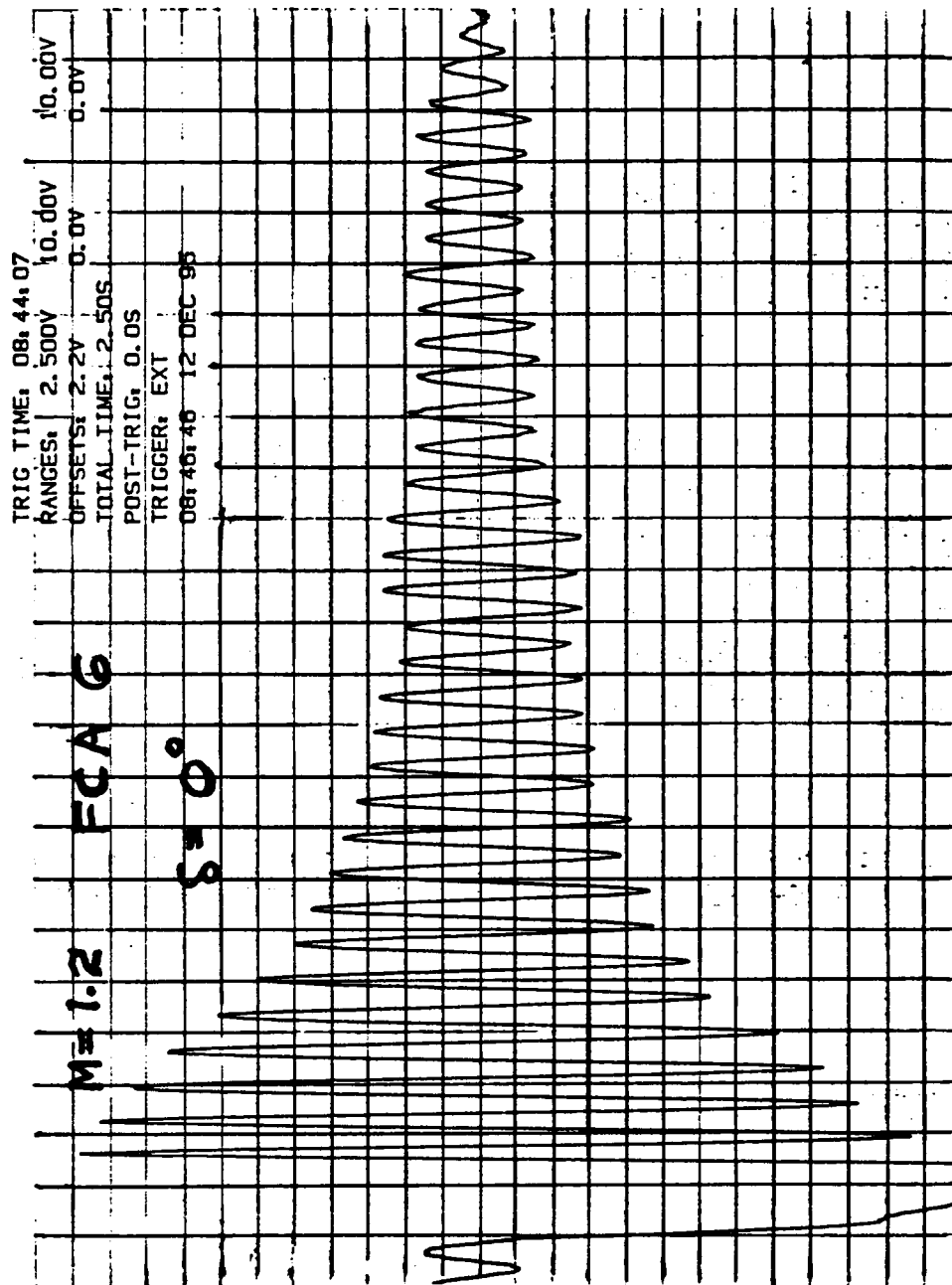


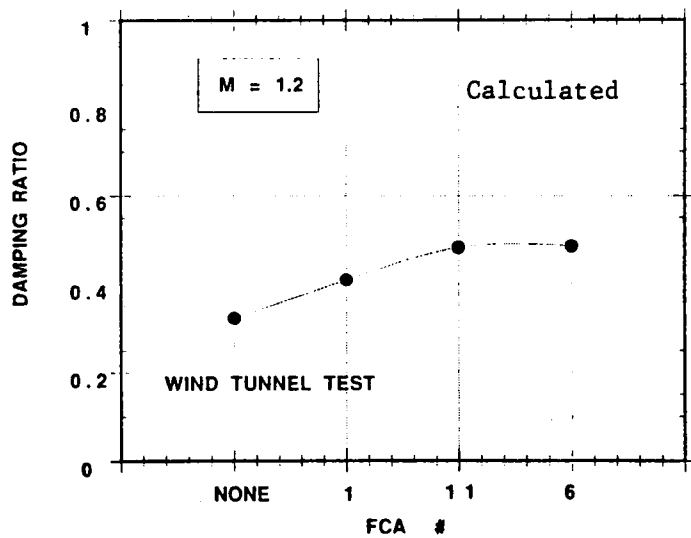
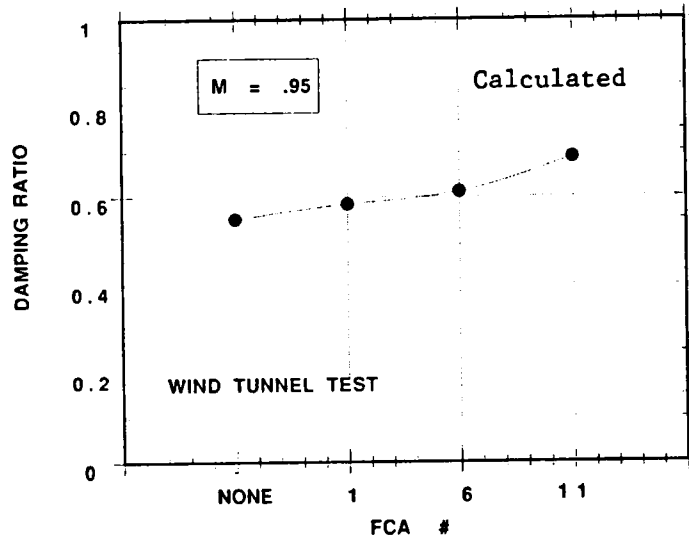
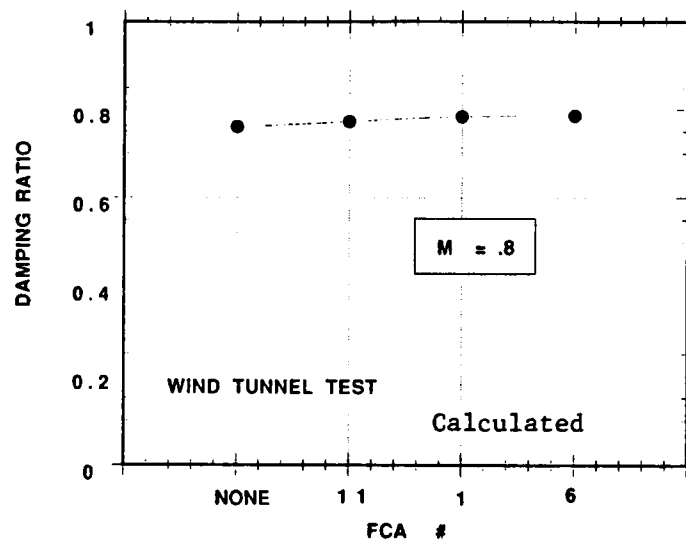




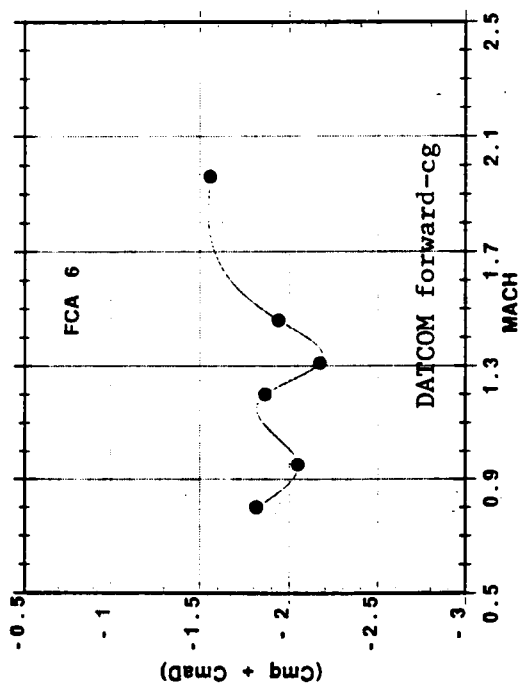
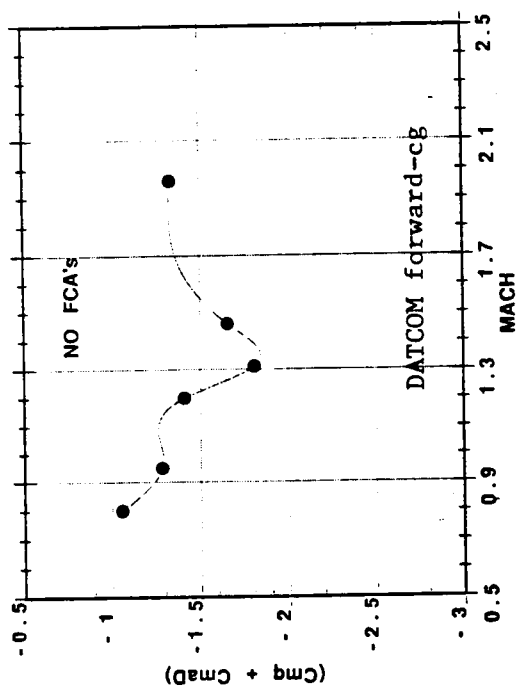
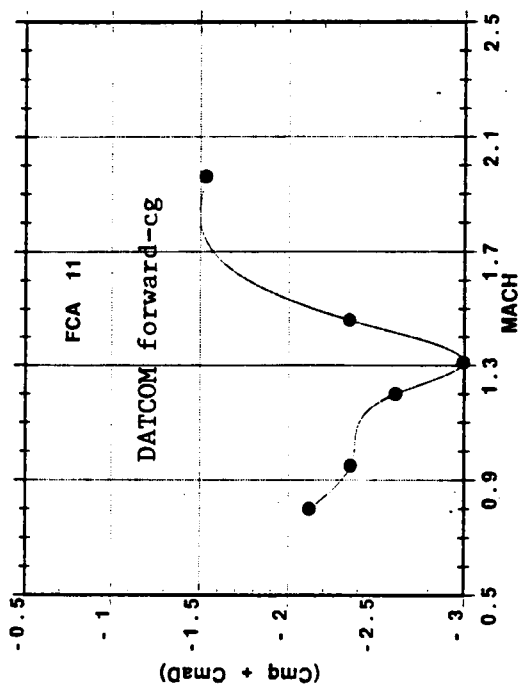
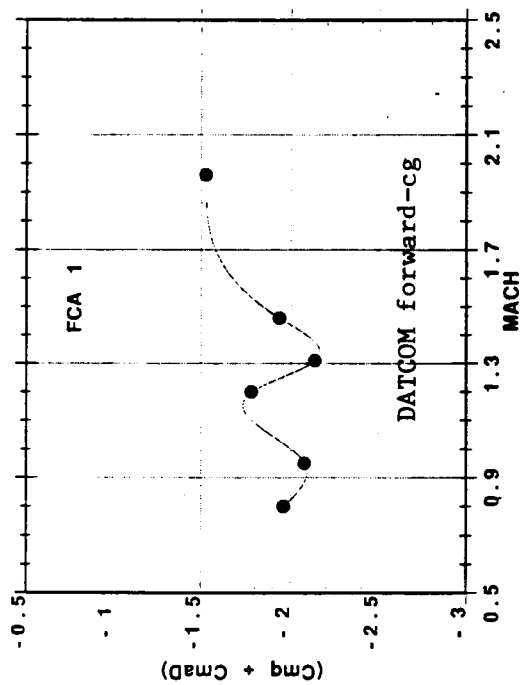












## VITA

Chris Barret was born and raised in Cleveland, Ohio, lived most of her adult life in Los Angeles, and then transitioned to a wilderness retreat on a private mountain in Tennessee. Chris has earned a B.S. in Aeronautical and Astronautical Engineering from the Ohio State University (3.5/4.0), an M.S. in Aerospace Engineering from the University of Tennessee Space Institute (UTSI) (3.6/4.0), and a Ph.D. in Aerospace Engineering from UTSI. She was awarded a 1994 Engineer of the Year Award, serves on the AIAA Atmospheric Flight Mechanics Technical Committee, and is a member of Tau Beta Pi the National Engineering Honor Society, Sigma Gamma Tau the National Honor Society in Aerospace Engineering, the American Institute of Aeronautics and Astronautics, the Society for Women Engineers, the Experimental Aircraft Association, the Soaring Society of America, and the American Helicopter Society.

Chris is a world adventure traveler with Africa her favorite destination, and has been a pilot for 21 years. She has organized a Civil Engineering Study Tour from the Ohio State University to Egypt, which studied the Pyramids, the Aswan High Dam, the Irrigation Systems of the Nile Delta, the Hydrodynamic Oil Tanker Facilities on the Suez Canal, and the ancient Temples and Tombs in the Valley of the Kings and Queens. In North Africa, she has piloted her own plane over the Sahara Desert, climbed Cheops Pyramid, and sailed the Nile. In East Africa she has been to Mt. Kilimanjaro, on grand safaris, and ballooned over the Serengeti Plains. In South Africa she has studied Mining Engineering on tour in the diamond and gold mines of South Africa at 11,000 ft below the surface, and sailed around the tip of the African Continent, the Cape of Good Hope. She has just returned from an Alaskan Adventure where she interacted with the Kodiak bears on Kodiak Island, crossed the Arctic Circle, and went to the top of the North American Continent to Prudhoe Bay, the oil field of the Alaskan Pipeline, and Barrow on

the Beaufort Sea of the Arctic Ocean. Her next trip will be to West Africa including Dakar, the Ivory Coast, Gold Coast, and Timbuktu! After that the Amazon and a circumnavigation of the Antarctica await.

Chris is employed as an Aerospace Engineer at the Marshall Space Flight Center where she has worked for the past ten years in the Flight Mechanics and Vehicle Control Systems Branch of the Structures and Dynamics Laboratory. She has worked on the flight dynamics of such launch vehicle programs as the Space Shuttle, the Shuttle-C, the National Launch Systems (NLS), and Saturn V derivative heavy lift launch vehicles (HLLV's). Most recently she has been the Principle Investigator on a special research project for the Center Director and funded from the Center Director's Discretionary Fund, to provide flight control augmentation for aft cg launch vehicles. This research involves the design and test of innovative aerodynamic flight control surfaces to provide flight control augmentation on aft cg launch vehicles allowing them to fly with increased payloads. She also works on the flight dynamics of other flight vehicles such as the X-2000, the High Speed Civil Transport, Single-Stage-to-Orbit, Reusable Launch Vehicle Programs, X-33, and X-34.