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FOREWORD

The research reported herein was conducted by Giannini Controls Corporation, Astromechanics Research Division, Malvern, Pennsylvania under Contract AF 33(616)-8260, Projects Number 1469 and 8219, Tasks Number 146901 and 821901. The contract with this numerical designation has been termed the MODEL FLY program. The work was administered by Messrs. H.A. Wood and W.G. Williams of the Air Force Flight Dynamics Laboratory, Research and Technology Division, who served as Air Force Project Engineers. Mr. James A. Hill served as Project Engineer for the Contractor.

The MODEL FLY program is a multi-fiscal-year effort. The studies reported herein are for the time period 1 September 1962 to 31 August 1963 and as such, constitute the first MODEL FLY Annual Technical Report. This report concludes the work on Contract AF 33(616)-8260. Further development of MODEL FLY will be carried on under Contract AF 33(615)-1102.

The analysis referenced in Section 6 was performed by Messrs. W.G. Williams and H.M. Davis of the Air Force Flight Dynamics Laboratory. Experimental information referenced in Section 6 was collected by Mr. R.I. Lowndes, Aerodynamics Branch, AEDC, Tullahoma, Tennessee.

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ABSTRACT

A status report of the MODEL FLY program is presented. The report presents the results of the preliminary considerations for development of the MODEL FLY testing technique.

The MODEL FLY program objective is to develop a wind tunnel testing technique for simulating the effects of aeroelasticity on the structural loads and stability and control response of full scale vehicles. A dynamic mounting system is to be developed which allows scaled elastic models to perform "free flight" maneuvers in the wind tunnel.

The MODEL FLY development encompasses six technical areas: similitude theory, model construction, mounting system, control theory, instrumentation, and data processing. The problems in each area are discussed and possible methods of solution outlined. The present status of the development is summarized and future direction discussed.

This report has been reviewed and is approved.

Lawrence C. Wright

LAWRENCE C. WRIGHT
Lt/Col, USAF
Chief, Flight Control Division
AF Flight Dynamics Laboratory

R. F. Hoener

R. F. HOENER
Acting Chief
Structures Division
AF Flight Dynamics Laboratory

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1.0 INTRODUCTION

1.1 STATE-OF-THE-ART

The design of modern high speed air vehicles is significantly influenced by aeroelastic effects. Considerable progress has been achieved in the art of constructing and testing flutter models. The effects of aeroelasticity on the other aspects of air vehicle design, for example: structural loads, maneuverability, and control system design, have been engineered largely by theoretical methods. Wind tunnel techniques for measuring both static and dynamic force and pressure data have proven very reliable and have provided excellent data. It has become increasingly apparent that these achievements could be consolidated into a technology for simulation of aeroelastic effects on both static and dynamic structural loads and response characteristics. As a result, the MODEL FLY program was established with the objective of developing this aeroelastic modeling technology.

Table 1 technically orients the MODEL FLY program with respect to other testing techniques currently being utilized. The table also includes rigid model testing, so that conventional loads and stability and control testing can be related. The MODEL FLY program embraces the area contained within the crosshatched lines. The areas marked with "N" (no capability) indicate the deficiency of present very high speed techniques.

1.2 ADVANTAGES OF THE MODEL FLY SYSTEM

The main advantages of this system are: (1) the development time of new vehicles will be considerably accelerated, and (2) the cost of development of new vehicles will be reduced considerably both by virtue of shortened development time and by replacing a part of the contemporary flight test program. There are many other significant advantages to such a technique, among them - (1) safety of test vehicles and test pilots since it will be possible to first explore potentially dangerous flight regimes in the wind tunnel, (2) examination of loads and/or stability of re-entry vehicles which at the present time cannot be flight tested in the transonic

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TYPE DATA	SUBSONIC	TRANSONIC	SUPERSONIC	HYPERSONIC	ORBITAL & SUPER ORBITAL VELOCITIES
1. Rigid Body Static Derivatives	Conventional	Conventional Control terms difficult	Conventional	Conventional with heating and considerations	N
2. Rigid Body Steady Airloads (Pressures)	Conventional	Conventional (possible tunnel unsteady effects)	Conventional	Low pressure instrumentation limits accuracy	N
3. Rigid Body Dynamic Derivatives	Combined derivatives can be measured using appropriate techniques. Cross dynamic derivative and pure dynamic derivative difficult.			Tests possible in continuous tunnels	N
4. Rigid Body Dynamic Stability and Response	Spin, free flight tunnel and special mounts	Special cable mounts	N	N	N
5. Static, Servo Driven Sting Mounts	Computer controlled sting(s) used to measure derivatives along a trajectory segment - particularly useful for separation study.			N	N
6. Static, Flexible Airload Models	Possibility demonstrated by successful wind tunnel tests.			N	N
7. Flutter Stability	Conventional	Conventional	Conventional	Research	N
8. Limited or Simulated Unrestrained Flutter Stability	Conventional using cables, rods	Subsonic methods extension	Limited to wall mounts or rotations on sting	N	N
9. Gust and Dynamic Response (Wind Tunnel only)	Recent mount and gust generator research at NASA		N	N	N
10. Maneuvering Aeroelastic Response	Demonstrated with special mounts	Research	Research	N	N
11. Lifting Trajectory Simulation	See #5 - No dynamic study reported	Research	Research	N	N
12. System Simulation (complete lashup with wind tunnel aerodynamics)	N	N	N	N	N

TABLE 1

TECHNOLOGICAL STATUS OF AERODYNAMIC WIND TUNNEL SIMULATION FOR STRUCTURAL LOADS AND STABILITY AND CONTROL - 1963

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region (for reasons of economy among others), (3) investigation of gust phenomenon wherein vehicle transfer functions can reliably be determined and gust alleviation schemes evaluated, (4) accurate airloads will be available for maneuvering vehicles in the transonic region prior to flight and in time to take corrective action where required without major modification to full scale hardware, (5) vehicle control systems can be evaluated and corrected effectively by iterative model experiments in conjunction with appropriate analysis rather than iterative full scale experiments with analysis, and (6) such items as control reversal, man-machine interactions, and stability augmentation can also be evaluated in the transonic region.

1.3 PROGRAM HISTORY

Technical efforts on MODEL FLY commenced in July 1961. It was intended at that time to design, fabricate, and test an aeroelastic model and dynamic mounting system in the 16 x 16 foot transonic circuit of the Propulsion Wind Tunnel at the Arnold Engineering Development Center (AEDC), Tullahoma, Tennessee. However, a destructive failure occurred in the compressor section of the wind tunnel precluding use of this facility. It was estimated this breakdown would necessitate a tunnel shutdown of approximately three years. Efforts were then made to obtain commitment from other large test facilities. A new concept - free pivoted servo wings - was to be an integral part of the dynamic mounting system. These servo wings were designed to provide an acceleration phased force which would null the inertial reaction of the mount on the model and in essence permit "free flight". Applications for commitment were not accepted because of doubt in the servo wing concept and stability of a mount-model-servo wing system. At this point it was decided that a low speed wind tunnel program would best serve to demonstrate the feasibility of the servo wing concept with a minimum in expenditure.

A demonstration mount allowing the longitudinal degrees of freedom and a rigid model were designed and fabricated. Tests were conducted in the 5 x 7 foot low speed tunnel at the University of

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Michigan during August 1962. The tests provided sufficient data to verify the servo wing concept and the stability of the mount-model-servo wing system. The results of these tests are unpublished; a brief summary of the effort is presented in Ref. 1. A motion picture describing these tests has also been completed and is available from the Air Force Flight Dynamics Laboratory.

The technical efforts through the demonstration program concluded the first development phase. Thereupon a program review was held to discuss the plan of attack for development of the high speed operational system. It was mutually agreed by the U.S. Air Force and the Contractor that development of a high speed operational MODEL FLY system would require a major technical effort but was justified and would represent a significant advance in the modeling and testing art. The technical effort would encompass similitude theory, model construction techniques, mounting system, control system, sensors, data transmission, and data handling.

1.4 CURRENT STATUS

This report answers the following questions concerning each of the below mentioned technical areas as of August 1963:

QUESTIONS	TECHNICAL AREAS
1. What are the requirements in this area? 2. What is the present technological status in this area? 3. What direction is being taken to solve problems in this area?	Similitude Analysis Model Maneuvering Analysis Model Construction Techniques Dynamic Mount Configurations Inertia Cancelling Devices Instrumentation Considerations Data Processing

The above areas are treated in separate sections. An abridged PERT chart follows (Fig. 1) which is intended to enable the reader to graphically position these technical areas in the total MODEL FLY program and to observe the balance of work to be accomplished under

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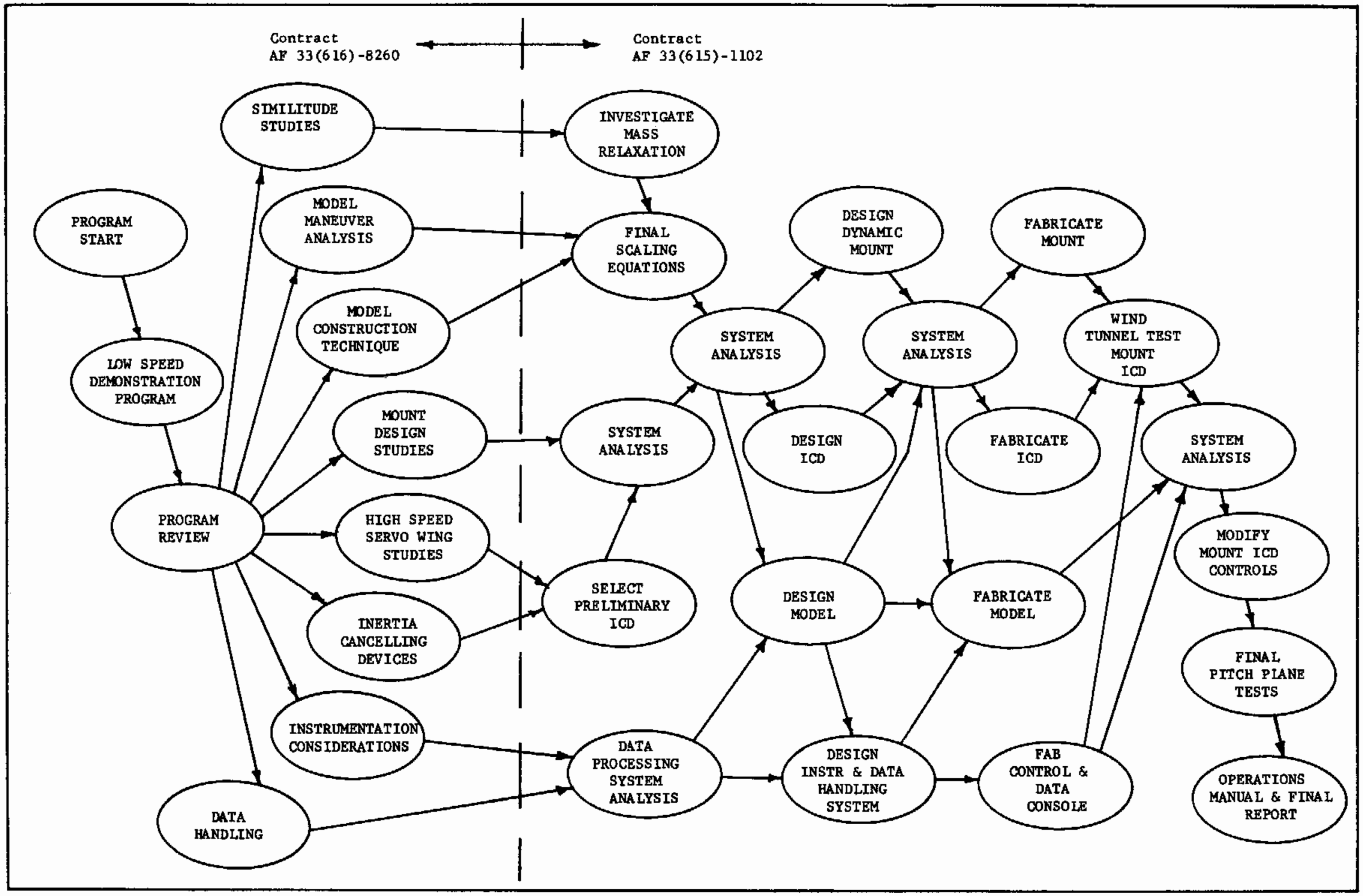

Figure 1 MODEL FLY PERT Diagram (Simplified)

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Contract AF 33(615)-1102. Contract AF 33(615)-1102 basically encompasses designing and installing a two degree of freedom (pitch and plunge) model and mounting system in the 16 foot PWT Transonic Wind Tunnel at AEDC. The aircraft to be modeled is the F-106A. Inherent in this contract is the treatment of subordinate technical disciplines leading to the successful completion of the program.

Attention is directed to the Bibliography (Section 11) which lists published technical reports associated with this program. These reports contain the bulk of the technical data supporting the results and discussions presented in this Summary Report.

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2.0 SIMILITUDE

2.1 INTRODUCTION

The MODEL FLY technology is aimed at simulating the effects of aeroelasticity on both the static and the dynamic structural loads, and stability and control responses of aerospace vehicles. The similitude requirements have been defined and are well known (e.g., Ref. 2 and 3). The approach used in Ref. 2 to derive the relevant dimensionless parameters is the inspection of the nondimensional equations of the physical system. Ref. 3 uses the dimensional analysis procedures and the familiar Buckingham π Theorem. Regardless of the method of derivation the same dimensionless parameters result.

An approach involving two parallel paths is used in defining the MODEL FLY similitude scheme. Path one takes the customary approach. It is known that complete true similitude (duplication of all pertinent dimensionless parameters) cannot be achieved practically except for full scale models. Therefore, the parameters are selected which when matched yield the most accurate simulation in accordance with the test objectives. Artificial schemes are then sought to minimize the effects of the unmatched parameters. The path two approach is to develop a scheme whereby the simulation of one or more dimensionless parameters is purposefully not achieved (i.e., relaxed) by a predetermined amount. This approach is desired when it is realized what practical considerations true similitude dictates, for instance, model mass requirements and system frequencies.

This section of the report summarizes the similitude studies. First the path one similitude approach is presented; the laws of similitude and areas of nonsimulation are discussed. A description is given of the schemes which will minimize the effects of the nonsimulated parameters. The quasi-similitude approach is defined. The prototype system is then studied to determine <u>how</u> and <u>why</u> quasi-similitude can be useful. The procedure for transformation from model to prototype data is defined and an example provided. Finally, future extensions to similitude studies are briefly considered.

2.2 TRUE SIMILITUDE/THE DYNAMIC AEROELASTIC PROBLEM

Similitude involves the selection of all physical parameters sufficient to describe the system to be modeled. Buckingham's π Theorem can be applied and nondimensional similarity parameters derived.

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If m is the number of physical parameters sufficient to describe the system and n the number of basic units for these parameters, then m-n similarity parameters will result. It is the task of the modeler to insure the matching of these similarity parameters between the prototype system and the model system in order that phenomena measured on the model system can be validly interpreted as scaled prototype phenomena. If the modeler can insure the matching of all but one of the similarity parameters (m-n-1) then the remaining one is automatically matched due to their interdependency.

$$\pi_i = f_i\ (\pi_a,\ \pi_b,\ \pi_c,\ \ldots\ldots\ \pi_{m-n})$$

To model a physical system, the model designer must first decide what variables are important in defining the system. The present MODEL FLY system is aimed at simulating the dynamic aeroelastic response in the longitudinal plane of an air vehicle at up to low supersonic speeds using a scaled model in the wind tunnel. The variables of interest are therefore those most important in defining the motions of a structurally elastic vehicle flying through a viscous, compressible fluid (air) at a particular velocity, attitude and altitude.

In the speed regime of interest the problem is aeroelastic rather than aerothermoelastic. Temperature effects are not significant and parameters such as coefficient of thermal conductivity, etc. are not necessary to describe the system.

The variables that describe the dynamic aeroelastic system can be classified in four groups: fluid environment, motion variables, applied forces and physical properties of the body. The pertinent variables are listed below under these classifications.

Fluid Environment

- T_∞ - stagnation temperature of fluid medium
- ρ_∞ - mass density of fluid medium
- γ - ratio of specific heats
- a - speed of sound in fluid medium
- μ - coefficient of dynamic viscosity

Motion Variables

- α - relative angle of attack
- θ - pitch attitude

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δ - control deflection

w_o - reference structural deflection

t - time

V - relative wind velocity

Physical Properties of the Body

m_o - reference mass

$m(\xi,\eta)$ - mass distribution

K_o - reference stiffness

$K(\xi,\eta)$ - stiffness distribution

ℓ_o - characteristic length

$\ell(\xi,\eta)$ - length distribution

Applied Forces

P_A - pressure of fluid over the model

g - acceleration due to gravity

There are a total of 19 variables describing the system, i.e., m = 19. The dimensions of the variables are based on the mass, length, time and temperature system, therefore, n = 4. Hence, from Buckingham's π Theorem there are m-n = 15 parameters which must be duplicated by the model to obtain true similitude. The dimensionless similarity parameters for the dynamic aeroelastic system are listed below. Reference 3 presents complete derivations.

1. Mach Number V/a
2. Reynolds Number $\dfrac{\rho V \ell_o}{\mu}$
3. Relative mass parameter $\dfrac{m_o}{\rho \ell_o^3}$
4. Relative stiffness $\dfrac{K_o}{\frac{1}{2}\rho V^2 \ell_o}$
5. Applied pressure coefficient $\dfrac{P_A}{\frac{1}{2}\rho V^2}$
6. Froude Number $\dfrac{V^2}{g \ell_o}$
7. Dimensionless time $\dfrac{Vt}{\ell_o}$
8. Relative structural deflection $\dfrac{w_o}{\ell_o}$
9. Dimensionless stiffness distribution $\dfrac{K(\xi,\eta)}{K_o}$

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10. Dimensionless mass distribution $\frac{m(\xi, \eta)}{m_o}$

11. Dimensionless length distribution $\frac{\ell(\xi, \eta)}{\ell_o}$

12. γ

13. α

14. δ

15. θ

2.3 PRACTICAL CONSIDERATIONS

The achievement of true similitude represents the ultimate goal; however, practical considerations have precluded attaining this goal. Indeed, total similitude can never be assured until infinitely flexible testing facilities and testing mediums have been established - an unreal expectation. In practice we have found that, depending on the particular problem under investigation, some similitude parameters have overriding significance compared to others. Those of little significance can remain unmatched while still maintaining adequate accuracy of required results. When significant parameters cannot be matched the effects associated with them are often duplicated by the employment of certain "fixes". The prototype to be modeled is the F-106A aircraft and model tests will be conducted in the AEDC 16 foot PWT Transonic Wind Tunnel. The first consideration in designing a model is the determination of the model's physical size. Based on wind tunnel size, interference effects and on-board instrumentation requirements, a length scale $\lambda_{\ell_o} = 1/10$ (where λ is the ratio of model to prototype parameter) was chosen. The practical application of the similitude requirements as affected by model scale and wind tunnel capabilities is discussed in the following paragraphs.

2.3.1 Mach Number ($M = \frac{V}{a}$) - The Mach number relates the flow velocity to the speed of sound. Mach number varies locally (over the surface of the model) and shock locations, intensity and orientation are affected. These shock characteristics in turn affect pressure distribution over the model and it is the measurement of this distribution that is often the goal of the experimenter. Physical parameters affecting the Mach number are seen in the equation:

$$a^2 = \gamma R T_\infty$$

R = characteristic gas constant

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The test medium is air and since the prototype also operates in air then $\lambda_{\gamma} = \lambda_{R} = 1$. If it were desirable to operate at other than prototype velocity (i.e., $\lambda_{V} \neq 1$) then T_{∞} could theoretically be varied to change a to match Mach number. However, there are physical limitations on the variance of T_{∞}. A cooling system in the tunnel is required to control the stagnation temperature. The AEDC 16 foot tunnel has such a system, but its upper and lower stagnation temperature values are 160°F and 70°F respectively. This represents a variance in a of only about ± 7%. Hence model velocity may differ from prototype velocity but only by a few per cent.

2.3.2 Reynolds Number ($Re = \frac{\rho V \ell_o}{\mu}$) - The Reynolds number is physically a ratio of fluid inertial to viscous forces. The requirements for varying either velocity, density or viscosity, alone, or collectively, to insure matching the Reynolds number are too extreme if the required model is much less than full scale. For a 1/10 scale model nominally with $\lambda_{\rho} = \lambda_{V} = \lambda_{\mu} = 1$ the Reynolds number would be less by a factor of ten. With the temperature restrictions of the tunnel, μ can only be varied ± 4% from nominal and V only ± 7%. Mass density could be varied at a maximum to 6.6 times the prototype value, but this would be for the extreme case of simulating the prototype at an altitude of 50,000 feet using the maximum tunnel density (sea level density). Hence, the Reynolds number cannot normally be simulated.

Models due to their lower Reynolds number will tend to have larger laminar regions than the full scale prototype. It is possible to duplicate the state of the boundary layer and locate properly the transition regions. Transition can be induced by use of boundary layer trippers which are often required near the model leading edge to duplicate a fully turbulent boundary layer. Above a Reynolds number of 4×10^{6}, based on mean aerodynamic chord, the behavior of the boundary layer tends to be turbulent and independent of Reynolds number and no fixes are necessary. Flow separation changes the entire character of the flow pattern and has a significant effect on the load distribution. Laminar flows separate at lower angles of attack than turbulent flows and for this reason boundary layer trippers are frequently used to energize laminar boundary layers and thus delay separation along the chord. The design of boundary layer trippers is largely

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empirical. For the F-106A, the Reynolds number problems will be investigated during the static aeroelastic tests of the complete model. Reynolds numbers based on mean aerodynamic chord will range from about $3 \times (10)^6$ to $25 \times (10)^6$ at Mach numbers .5 to 1.5. By using boundary layer trippers and with the beneficial combinations of sweepback and low aspect ratio (Ref. 5), separation at these Reynolds numbers is not expected to be a simulation problem. Therefore, large scale effects are not to be expected.

2.3.3 Mass and Stiffness Parameters $\left(\frac{m_o}{\rho \ell_o^3}\right)$ & $\left(\frac{K_o}{\frac{1}{2}\rho V^2 \ell_o}\right)$ - The mass and stiffness parameters are ratios of the structural inertia and structural stiffness to the aerodynamic force and aerodynamic stiffness, respectively. Also, in combination with the dimensionless mass and stiffness distributions, these parameters define the structural dynamic characteristics (mode shapes and natural frequencies) of the structure. Matching of these parameters is required to insure simulation of prototype response. For an F-106A model, considering previously defined restrictions on velocity and density, the model mass should be scaled by a factor of the length scale cubed, $\lambda_{m_o} = \lambda^3_{\ell_o} = 1/1000$, and the model stiffness should be scaled the same as the length, $\lambda_{K_o} = \lambda_{\ell_o} = 1/10$. Since the design weight of the F-106A is 30,000 lbs., the model weight should be 30 lbs. This could possibly be too constraining to an actual model considering all of the items that need be included in it, such as, control actuators and instrumentation. It is seen that the allowable model weight increases directly as the density ratio, λ_{ρ_∞}. For example if $\lambda_{\rho_\infty} = 2$, then the model weight could increase from 30 lbs. to 60 lbs. This density ratio can be provided by the AEDC tunnel for prototype altitudes above approximately 22,000 ft. More extensive studies must be performed before determining whether the approach of density scaling is necessary or desirable. Another method of relaxing the restriction on mass will be discussed under quasi-similitude.

2.3.4 Froude Number $\left(\frac{V^2}{g\,\ell_o}\right)$ - In general this parameter requires the scaling of the acceleration due to gravity (g). Once the length scale is fixed and the velocity relationship as outlined in Section 2.3.1 is accepted, simulation of Froude number is not possible; the gravitational constant "g" cannot be scaled. If $\lambda_V = 1$ and $\lambda_{\ell_o} = 1/10$, the Froude number requires that $\lambda_g = 10$.

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The physical implication of Froude nonsimulation is that a model built to simulate the mass requirements established by matching the relative mass parameter will not weigh enough by approximately a factor of the length scale. Because of this low total weight the model will fly at too small an angle of attack. An additional effect is present due to the improper dead weight distribution resulting in a difference in elastic structural deformations. Both of these effects can be accounted for as follows.

Simulation of angle of attack will be achieved in the MODEL FLY system by introduction of a force (simulating weight) independent of the model motion.

A warp (camber and twist) distribution will be built into the model lifting surfaces to approximate the deformation developed if the weight increment required for true similitude were applied. Built in warp affects the zero lift (basic load) distribution of the surface but the additional load distribution (load due to angle of attack) is essentially unaltered. Thus total aerodynamic load and moment distribution can be matched.

2.3.5 <u>Applied Pressure Coefficient</u> ($\frac{P_A}{\frac{1}{2}\rho_\infty V^2}$) - This parameter defines the relationship between the external pressures on the model and the prototype. It is the "unknown" in the similitude equation. As stated previously if all the other similitude parameters are sufficiently matched, this parameter will be satisfied. It is this parameter which will be measured in the wind tunnel.

2.3.6 <u>Dimensionless Time</u> ($\frac{Vt}{\ell_o}$) - This states that for $\lambda_V = 1$, time in the wind tunnel will be scaled the same as length scale. For the F-106A model to simulate prototype time scale, $\lambda_t = \lambda_{\ell_o} = 1/10$. That is, model events in the wind tunnel must occur faster than corresponding prototype events by a factor of 10.

2.4 QUASI-SIMILITUDE

It has been shown that practical considerations such as wind tunnel capabilities and nonvariable gravity render it impossible to obtain a truly similar model of a dynamic aeroelastic system. Hence the model designer is <u>forced to relax</u> the constraint of matching all of the dimensionless similarity parameters. He must select and match those parameters that

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significantly affect the model response of primary interest; he is forced to accept a partial model, which is however, the best achievable simulation.

A slightly different approach to the similitude problem would be to purposefully relax the constraints on matching some, or all, of the similarity parameters without employing "fixes" to duplicate the effects associated with the unmatched parameters. By relaxing the similarity constraints, a new model could be formed having response characteristics directly related to those of the true model. This approach has been named quasi-similitude and the resulting model has been called a quasi-similar model.

In this section, the quasi-similitude approach is discussed. First, the modeling variables that can be purposefully relaxed are discussed. Then the purpose or reasons why relaxation is desirable are presented. Finally, the theoretical basis and application of quasi-similitude to the determination of meaningful prototype response is outlined.

2.4.1 Quasi-Similitude Variables - Investigation of the prototype characteristics in view of the test objectives should reveal which parameters can be purposefully relaxed. Hence, a review of the MODEL FLY objectives would be the most suitable point to begin determination of quasi-similitude variables.

Foremost among the MODEL FLY objectives is the desire for true Mach number simulation. Secondly, since it is desired to simulate the vehicle aerodynamics, the outside geometry of the model is to be perfectly scaled. The built-in wing deformation outlined in 2.3.4 is an obvious departure from true geometric similitude in the strict sense. However, the end purpose of aerodynamic force simulation is more closely satisfied. Also, because simulation of aeroelastic effects is desired, the vibration mode shapes of model and prototype should match. Duplication of modes requires the relative duplication of mass and stiffness distributions; hence, static aeroelastic deformations will also be matched if aerodynamic similarity is achieved. These constraints can now be applied to the prototype establishing the parameters that remain as possible quasi-similitude variables.

Inspection of the equations of motion for an aeroelastic vehicle (Ref. 2) reveals that the following parameters may be varied within the above constraints:

1. Length
2. Time

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3. Air Density
4. Mass
5. Stiffness
6. Applied Forces

Of special significance, however, is the fact that only a few of these parameters may be relaxed independently in the design and formulation of the model and its testing. The only possible quasi-similitude variables are the model design parameters, mass and stiffness, and the model test parameter, air density. The applied forces cannot be varied independently. They must be simulated in both magnitude and rate of application to simulate structural response. Model length scale and time scale also cannot be used as quasi-similitude variables. The model designer does not have great freedom in selecting length scale. Upon selection of the test facility, the model size is dictated by wind tunnel interference and/or equivalent maneuvering altitude. Then the model length and previous Mach simulation constraint define the time scale.

2.4.2 Mass and Stiffness as Quasi-Similitude Variables - Except for the requirement of adequate strength to withstand the scaled limit loads, the design of an aeroelastic loads model is guided by the same requirements as a flutter model, and any relaxation of similitude parameters could alter the flutter velocity. A decreased flutter velocity may not permit simulation of the prototype flight envelope.

In Ref. 7, Zisfein and Frueh derived the following equation assuming no damping and utilizing piston theory aerodynamics; this assumes no unsteady aerodynamic effects. The effect of relaxing mass and/or stiffness on the flutter velocity can be seen using this expression for the pitch-plunge wing flutter velocity:

$$\frac{V_F}{b\omega_\theta} = \frac{-\left\{(r_\alpha^2 - X_\alpha^2)\left(\frac{\omega_o}{\omega_\theta}\right)^4 - \left(\frac{\omega_o}{\omega_\theta}\right)^2\left[r_\alpha^2\left(1 + \frac{\omega_h^2}{\omega_\theta^2}\right) + \frac{\omega_A^2}{3\omega_\theta^2}\right] + r_\alpha^2\left(\frac{\omega_h^2}{\omega_\theta^2}\right)\right\}}{(1 - 2X_o)\left(\frac{\omega_h}{\omega_\theta}\right)^2\left(\frac{\omega_A}{\omega_\theta}\right)} \tag{1}$$

where V_F = flutter velocity

b = semi chord

ω_A = aerodynamic frequency parameter $a/\mu\, b$

ω_θ = natural frequency of wing rotation (pitch)

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ω_h = natural frequency of wing translation (plunge)

ω_o = coupled frequency of flutter

X_o = distance from leading edge to axis of rotation

μ = $M/4\rho_\infty b^2$ nondimensionalized mass of wing

ρ_∞ = free stream density

X_α = static unbalance, semi chords

r_α = pitch radius of gyration, semi chords

a = speed of sound

M = mass of the wing

Mass and/or stiffness relaxation is the relaxation of the requirement for matching the relative mass parameter $m_o/\rho_\infty \ell_o^3$ and/or the relative stiffness parameter $K_o/\frac{1}{2}\rho_\infty V^2 \ell_o$ by altering m_o and/or K_o. Therefore, when mass and stiffness requirements are relaxed, parameters such as X_α, r_α, and X_o are unaltered. It should also be noted that although these relaxations may change the magnitude of the system frequencies, ω_θ, ω_h, ω_o, etc., the frequency ratios ω_θ/ω_o, ω_h/ω_θ and modeshapes remain unchanged.

The effect of mass relaxation on the flutter velocity can be seen from the equation which is formed by multiplying Equation (1) by ω_A/ω_θ. Substitution of the proper definitions for terms in the left hand side of the new equation and knowing that r_α, X_α, etc. and frequency ratios remain constant when mass is relaxed, the new equation can be expressed

$$\frac{4a\rho_\infty V_F}{M\omega_\theta^2} = \text{constant} \tag{2}$$

It is apparent that the effect of a mass relaxation is balanced by a corresponding change in the structural rotational frequency, $\omega_\theta^2 = \frac{K_\theta}{M}$, and the flutter velocity is unaffected. On the contrary, if structural stiffness is relaxed (K_θ varied), the flutter velocity will change.

An increase in stiffness will result in an increased flutter velocity which will not place any restriction on simulation of the complete flight envelope. If mass and stiffness are increased proportionally, then flutter velocity will be increased but structural system frequencies will be properly scaled. This is because structural frequencies must be scaled inversely as time, $\lambda_\omega = 1/\lambda_t = 10$, and therefore the ratio K/M must be kept constant, i.e., $\lambda_{K/M} = \lambda_\omega^2 = 100$.

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2.4.3 <u>Practical Benefits of Mass and Stiffness Relaxation</u> - An obvious practical benefit of mass and stiffness relaxation is the freedom gained by the model designer in constructing the model. With model weight and stiffness constraints eased, the designer may concentrate more effort in providing a structure which better duplicates the stiffness and mass distribution and provides proper strength. Also, the implicit MODEL FLY objective of utilizing one model to obtain both limit load and stability response appears more realizable. An accurate aeroelastically scaled model with the quantity of instrumentation necessary to measure loads and stability response will be a real challenge to the model designer's ingenuity. Quasi-similitude does not, by any means, eliminate this design challenge, but may ease some of the problems.

These practical benefits could significantly ease the system design requirements. These benefits, however, should not be realized at a cost in the accuracy of the predicted prototype response. Further investigation is definitely required to fully evaluate all of the ramifications of mass and stiffness relaxation.

2.4.4 <u>Tunnel Air Density as a Quasi-Similitude Variable</u> - The Mach number - altitude capability of the wind tunnel might not envelop the prototype flight envelope. A typical example of this shortcoming is illustrated in Fig. 2 which is taken from Bibliography Item 15. It is seen that the flight envelope for the F-106A airplane exceeds the capability of the AEDC 16 foot transonic tunnel. Also, neither the Langley 16 foot transonic nor the Ames 14 foot transonic tunnel possess sufficient capability. Therefore, quasi-similitude of tunnel air density is the only means whereby prototype response can be predicted in flight regimes outside the tunnel Mach-altitude envelope.

2.4.5 <u>Theoretical Basis of Quasi-Similitude and its Application</u> - Quasi-similitude is formally defined as the purposeful relaxation of one or more modeling parameters to allow practical modeling and testing techniques. Quasi-similitude is not an independent similitude scheme; it derives from a knowledge of the dimensionless parameters that physically describe the prototype system.

As mentioned previously, the nondimensional similarity parameters could be derived by dimensional analysis or through the equations of motion that govern the prototype system. To formulate the algebra of quasi-similitude, the dimensional analysis approach is used to minimize as much

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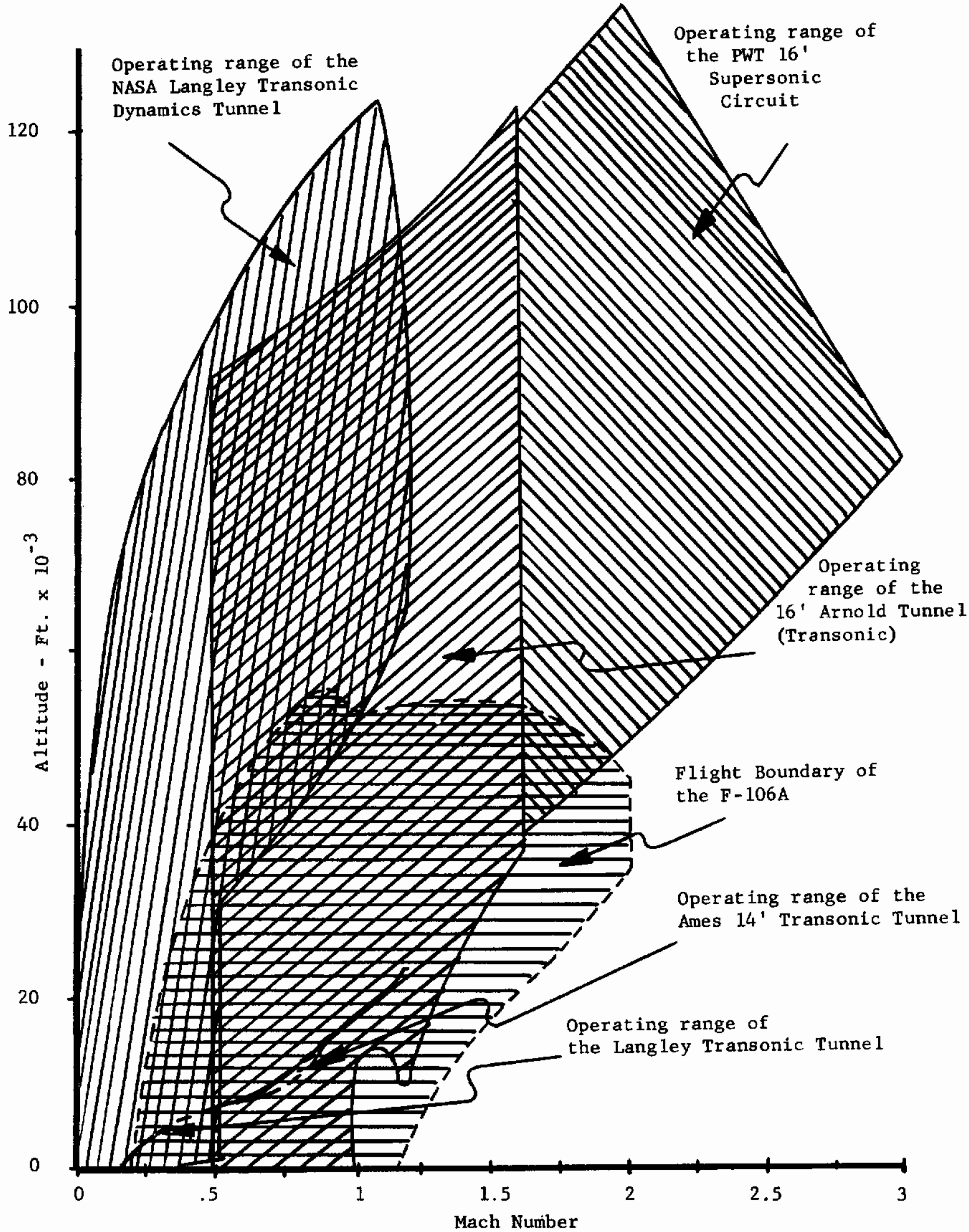

Figure 2 F-106A Flight Envelope Compared to Test Envelopes

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detail as possible.

For similitude the interdependency of the nondimensional similarity parameters is algebraically stated as Buckingham's π theorem (Ref. 6)

$$\pi_i = f_i \ (\pi_a, \ \pi_b, \ \ldots\ldots \ \pi_{m-n}) \tag{3}$$

True similitude between prototype and model requires that all of the dimensionless parameters are duplicated by the model. In quasi-similitude, the constraint of invariance is relaxed on simulation of some or all of the parameters. As a result, a model of the prototype, defined as a quasi-similar model (QSM), is obtained wherein the following algebraic relationships exist between its π_i parameters and those of the truly similar model (TSM).

$$(\pi_i)_{QSM} = R_{f_i} \ (\pi_i)_{TSM} \tag{4}$$

where R_{fi} = relaxation factor for the i^{th} parameter, the amount the parameter is allowed to differ from the true similitude requirement

An expression for the $(\pi_i)_{QSM}$ can be written similar to Equation (3) which then represents the functional dependency of the parameters for the quasi-similar model.

$$(\pi_i)_{QSM} = g_i \ (R_{f_a} \pi_a, \ R_{f_b} \pi_b, \ \ldots\ldots \ R_{f_{m-n}} \ \pi_{m-n}) \tag{5}$$

The equations of motion of the prototype can be expressed in terms of the nondimensional parameters, or in other words, a particular response of the prototype can be expressed as a function of Mach number, Reynolds number, reduced frequency, etc. Therefore, the response parameters for the TSM and the QSM can be written as

$$D_{TSM} = F(\pi_a, \ \pi_b, \ \ldots\ldots \ \pi_{m-n})$$

and $$D_{QSM} = G(R_{f_a} \ \pi_a, \ \ R_{f_b} \ \pi_b, \ \ldots\ldots \ R_{f_{m-n}} \ \pi_{m-n})$$

where D = a response, i.e., stress, load, frequency, etc.

Theoretically, therefore, a ratio of the TSM response relative to the correspond-

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ing QSM response can be obtained as follows:

$$\gamma_D = \frac{D_{TSM}}{D_{QSM}}$$

This ratio is the desired prediction factor for transforming experimentally determined QSM results to predicted TSM results. In practice prediction factors will be needed for all response parameters of interest, for example damping factor, total load and bending moment. It should be noted that in the limiting case (as the relaxation factors approach unity) the QSM equals the TSM.

Predicted TSM response will be in error if the theoretical QSM does not resemble the experimental result. Quasi-similitude requires equations of motion which adequately represent the behavior of the prototype and models. A gross difference between experimental and theoretical QSM response would indicate that the equations are not truly representative. The static aerodynamic characteristics can be obtained from static force tests. In general, then, if the experimental QSM response is not as expected, the difference is attributable to inadequate knowledge of the rate-dependent aerodynamics. A suitable adjustment could be made to correct the difference.

2.4.6 Model to Prototype Transformation Procedure - The prediction factors for relating experimental to prototype response are a key to the application of quasi-similitude. In this section, the procedure for applying these factors is outlined. The discussion that follows is limited to consideration for transforming a vehicle transfer function. The basic procedure is applicable to the calculation of any desired prediction factor.

The stability analyst and the control system designer utilize the transfer function as a basic "tool of the trade". Therefore, prediction of prototype transfer functions from model results will be an important result of the modeling technology. The necessary steps for obtaining the predicted prototype pitch/control transfer function from experimental results utilizing QSM with mass relaxation are as follows:

Step 1

Obtain the theoretical pitch/control transfer function for the TSM, $\left[\theta/\delta\right]_{TSM}$. The transfer function is formed from the equations of motion. The parameters that describe the TSM and the desired test conditions are substituted.

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Step 2

Obtain the theoretical pitch/control transfer function for the QSM, $\left[\theta/\delta\right]$ QSM. The parameters describing the QSM (basically the same parameters used for the TSM, but appropriately adjusted by the relaxation factors) and the actual test conditions are substituted into the expression for the transfer function.

Step 3

From the transfer functions obtained in Steps 1 and 2, form the prediction factor

$$\gamma_{\theta/\delta} = \frac{\left[\theta/\delta\right]_{TSM}}{\left[\theta/\delta\right]_{QSM}}$$

Step 4

Obtain the predicted TSM response by multiplying the experimentally determined transfer function by the prediction factor. Stated algebraically,

$$\left[\theta/\delta\right]_{\substack{TSM\\PRED}} = \gamma_{\theta/\delta} \left[\theta/\delta\right]_{\substack{QSM\\EXP}}$$

Step 5

Using predicted TSM transfer functions, $\left[\theta/\delta\right]_{\substack{TSM\\PRED}}$, apply the necessary reciprocal laws of similitude to obtain the predicted prototype transfer function.

2.5 FUTURE SIMILITUDE ANALYSES

All future similitude analyses will be directed at improving the quasi-similitude approach (refining the prediction factors). The present mathematical model as presented in Bibliography Item 6 is limited by linearized, steady state aerodynamics. The validity of the quasi-similitude approach depends on adequate representation of the aerodynamics. Therefore future work is planned to incorporate refinements in the areas of unsteady aerodynamics and nonlinear effects.

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3.0 SURVEY OF MODEL CONSTRUCTION TECHNIQUES

3.1 INTRODUCTION

Similitude requirements are an important part of the MODEL FLY technology. The laws of similitude dictate the bounds of simulation which are attainable. However, the practical elements of the problem, e.g. wind tunnel characteristics, mount dynamics, and model construction determine what simulation is practicably realizable. This section deals with one of these practical elements of similitude, model construction.

The advent of high speed, thin winged, high performance airplanes has spurred considerable progress in the aeroelastic modeling field during the past fifteen years, primarily in flutter modeling. A survey of the technical literature and of the engineers engaged in aeroelastic modeling has been conducted to determine possible construction techniques for use in aeroelastic loads models. This survey was intended to gather information pertaining to construction techniques utilized for contemporary aeroelastic models. The information gathered is to form the foundation for further development of modeling technologies under the MODEL FLY program.

This section presents the findings of the model construction techniques survey. A discussion is presented on the major structural properties which the model must simulate. Then the techniques for constructing models are discussed qualitatively. The advantages and disadvantages of each technique as it pertains to possible loads modeling applications are discussed.

3.2 PROPERTIES OF THE PROTOTYPE WHICH MUST BE SIMULATED BY THE AEROELASTIC LOADS MODEL

The prototype properties which must be simulated are:

1. External contours
2. Stiffness magnitude and distribution
3. Mass magnitude and distribution
4. Strength to withstand design maneuver loads
5. Internal model control systems capable of moving the control surfaces upon command

External contours, surface roughness, and elastic deformations under aerodynamic and inertial loads are parameters which interact, especially while an air vehicle is maneuvering in flight. In the transonic flight region the inherent nonlinearities in aerodynamic behavior dictate that all of the above properties be matched or suitably scaled.

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Contour and surface roughness affect airloads and boundary layer (shock attach point or boundary layer separation point). The stiffness and mass distribution determine the dynamic characteristics of the structure. The strength must be great enough to withstand maneuvering loads. The model must be able to maneuver with its own control surfaces since it will be essentially flying without external restraint (ideally only a pure thrust vector added).

In addition to the above properties it will also be necessary to provide means for carrying instrumentation (pressure transducers, accelerometers, etc.) for data measurement.

3.3 RESULTS OF THE SURVEY

The survey was conducted using literature searches and discussions with engineers engaged in modeling work. The latter means proved more productive as a way of learning simulation techniques. This was to be expected for two reasons: (1) this type of modeling is usually performed in conjunction with the development program of a production airplane, and (2) modeling techniques developed by various companies are generally regarded proprietary and as a result documents describing these techniques are not available through the usual information retrieval sources such as DDC or OTS.

The various techniques are discussed below particularly with regard to application for simulating lifting surfaces. The discussion is qualitative but points out advantages and possible disadvantages of each technique.

3.3.1 Lifting Surface Techniques - A correlation exists between the structural behavior and the aspect ratio of wing planforms. High aspect ratio wings can be approximated by beam theory but a more exact representation is required for low aspect ratio wings which deform in a plate-like manner. Model construction techniques must reflect this difference.

3.3.1.1 Single Spar Construction - Probably the best known and most widely used flutter model technique is the single spar contoured balsa section scheme developed by the Boeing Company (Ref. 2). This technique is generally limited to simulation of high aspect ratio lifting surfaces at low speeds.

In the simplest form of this method a rectangularly shaped beam simulates the torsion box of the surface, the beam centerline coinciding with the so called elastic axis. The ratio of torsional to bending stiffness is controlled by the ratio of the sides of the beam section. The

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beam strength is obtained by increasing the size of the section. Although this provides an efficient scheme for flutter models which generally are flown at low load levels, a straight extrapolation of the technique to provide adequate strength to withstand the design loads would be highly inefficient (weightwise), A more likely extrapolation of the technique would be a hollowed out rectangular beam. This technique with the addition of the balsa or mylar sections would yield a structure similar in appearance to that of the prototype.

Balsa sections contribute negligibly to the structural stiffness; the only function of these sections is to provide the geometric contour for aerodynamic simulation. Of course at high speeds a smooth continuous surface is desired to minimize flow distortions. However a one piece skin would contribute stiffness to the surface, thus adding complexity to the design for stiffness matching.

The technique as discussed above is fairly accurate for straight wings. However, for simulation of highly swept wings, a beam network should be used in the root chord vicinity for more accurate simulation. North American Aviation, Columbus, Ohio, has utilized this technique on models of the Navy Attack Airplane A-5 (Ref. 11).

3.3.1.2 The Foam Core - Aluminum Skin Technique - The foam core-aluminum skin technique was developed by the Cornell Aeronautical Laboratories (Ref. 12) to fill the need for a transonic flutter modeling technique. The technique provides a high strength to weight ratio, is rather inexpensive to use, and is flexible for many applications. It can be used for all aspect ratio wings and control surfaces.

In its basic form the strength and rigidity of the technique are concentrated in the metal skin. Ideally the plastic foam core provides neither bending nor torsional rigidity, its prime function is to prevent skin buckling. In addition the foam acts as the carrier of the ballast weights. The big problem area using this technique is the bonding of the skin to the core. It has been found that the bonding material (glue) amounts to as much as 25% of the total construction weight. Therefore, careful control in maintaining an evenly distributed glue line is required.

At Cornell Aeronautical Labs the skin ($.001'' \leq t \leq .006''$) was cut from a template of the developed surface, then fitted and bonded to the foam core. At NAA-Columbus a variation in the actual fabrication procedure was introduced which permitted working with thicker skins. A thick skin is

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chemically milled to the desired thickness. This technique permits use of tapered skins. In one instance a delta wing was fabricated which was tapered in three directions: spanwise; chordwise with different taper from middle to aft and middle to leading edge. This fabrication procedure adds more flexibility to the technique.

Another variation of the technique is the insertion of metal spars into the foam core. This variation decreases the strength to weight ratio, but appears to be a method for increasing the strength while maintaining the stiffness distribution.

3.3.1.3 Multi-Spar and Rib Construction - The multi-spar and rib technique has been utilized for simulating both high and low aspect ratio surfaces. However, since previously mentioned techniques are more efficient for high aspect ratio surfaces, the following discussion of the multi-spar and rib method will be limited to low aspect ratio applications.

The fabrication of an array of spars and ribs has been accomplished in two ways: (1) spar and ribs have been built up to the desired bending stiffness from channel sections with flanges bonded where needed, and (2) the spars and ribs can be machined from a pre-contoured piece of metal stock. In both cases the beam array almost duplicates the prototype structural arrangement. The first method was employed on scaled models of the F-102 and B-58 airplanes (see Ref. 13 and 14 respectively). The latter approach was used at Chance Vought on a low aspect ratio missile fin. In all instances a thin skin was bonded to the beam grid to provide the aerodynamic contours. This thin skin contributes stiffness to the total surface and must be taken into account.

When simulating the F-102 (Ref. 13) Convair elected not to match each rib because of complexity. The structure was idealized by combining three ribs into one. The spars and main ribs were simulated accurately. A similar approach was used at Chance Vought but the increased spacing and skin thickness used allowed the skins to buckle. The problem was alleviated by putting a soft foam plastic in the gaps between the spars before bonding the skin. The foam performs as a stabilizer.

3.3.2 Lifting Body Techniques - The construction techniques used for lifting bodies (fuselage and pod-pylons) can be relatively simple or complex depending on the purpose of the model. For instance, when modeling the pod-pylon combination for a flutter model of a current jet airliner, the

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elasticity of the pod-pylon itself is subordinated. Simulation of only the gross characteristics such as weight, center of gravity, and side bending, pitch, and torsional structural frequencies is obtained because only the interactions with the main lifting surface are desired. The complex full scale structure is generally idealized by a simple beam which matches the side bending and torsional frequencies. As a result, structural damping for the model pod-pylon does not match the prototype and the overall dynamic response is not matched. For an aeroelastic loads model, better response simulation is mandatory. A possible technique for designing damping into the structure, yet maintaining relative simplicity, is by employing visco-elastic materials as outlined in Ref. 15.

The simulation of fuselage structural properties is very important, especially for low aspect ratio and/or slender body vehicles. For low aspect ratio type vehicles, the fuselage provides root chordwise bending restraint for the lifting surfaces and thus influences the aero-elastic response of the vehicle.

Fuselage elastic simulation for many low speed flutter models is obtained by representing the structure by a single beam used to match vertical and lateral bending and torsional stiffness.

3.3.3 Model Control Systems - Model actuators capable of moving control surfaces at the rate required for MODEL FLY have not been historically employed. The survey yielded no information that could be directly used. Actuators which have been used in the past to power control surfaces are generally too heavy and too slow. Electric motors driving the control surfaces through appropriate linkages, gears, or chains have been used to trim models.

3.4 CONCLUSIONS

It is evident from the model survey conducted that it will be necessary to advance technological status in the following areas in order to satisfy the MODEL FLY objectives:

1. The model must be constructed to exhibit the correct dynamic - both rigid and flexible body - characteristics, and be strong enough to withstand the scaled limit load factor of the full scale prototype. This also implies an adequate factor of safety. The method selected will probably be a combination of metal skins over a foam core with intermediate spars and ribs to control both stiffness and strength levels. It is planned to construct a series of models to determine strength and stiffness effectiveness.

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2. The model control system must permit motion of the control surfaces at scaled rates at least as rapid as those exhibited by the full scale vehicle. This extends from a power source through the actuators and linkages, and finally causes a control surface, spoiler, flap or other surface to move in a prescribed fashion. Position control by feedback will probably be required. A definite advance is required in the state-of-the-art to design very small pneumatic or hydraulic actuators which exert large forces at high frequencies with good resolution.

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4.0 MODEL MANEUVER ANALYSIS

4.1 INTRODUCTION

The MODEL FLY testing technique represents a significant advance in the state-of-the-art for testing scaled models in the wind tunnel. A MODEL FLY model will be capable of flying maneuvers utilizing forces created by its control surfaces. The maneuverability of the model, in practice, will be limited by tunnel test section dimensions, and tunnel and model safety considerations. The equivalent tunnel maneuvering altitude is a function of the selected model length scale and tunnel size.

This section presents the results of an analog analysis of a scaled F-106A aircraft performing maneuvers which are directly comparable with measured full scale flight maneuvers. The section also presents analyses which duplicate only critical portions of maneuvers (for example, the portion of a symmetric climb dive wherein maximum normal loads occur).

4.2 FLIGHT MANEUVERS REQUIRED

The initial development of the MODEL FLY system is aimed at simulation in the longitudinal plane of motion. The prescribed symmetric flight maneuvers for demonstrating flight stability and structural integrity of an airplane are set forth in Military Specifications MIL-F-8785 ASG and MIL-S-5711 (USAF). It is these maneuvers that the MODEL FLY system must simulate.

The flight stability maneuvers consist of demonstrating that the transient dynamics excited by the control pulses throughout the flight envelope of the airplane are stable. The airplane controls are pulsed both in steady (trim) or in accelerated flight. Incremental load factors $\Delta n_z = \pm 1g$ are the usual maximum attained. This type of maneuver does not require much altitude; hence should be relatively easy to duplicate in the wind tunnel.

The longitudinal or symmetric flight loads maneuvers in the flight regime of interest are:

1. Straight and Level Mach Altitude Survey
2. Abrupt symmetrical pull out
3. Normal symmetrical pull out
4. Abrupt symmetrical push down
5. Normal symmetrical push down

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The first of these maneuvers is the initial phase of a flight demonstration program. Load-Mach number trends are established from which peak load areas are determined. The remainder of the flight program consists of maneuvers of the types (2) to (5) at various percentages of the design limit load factor in the peak loads region of the flight envelope. This portion of the test determines the critical loading conditions. The critical conditions are then demonstrated at 100% design limit load factor.

4.3 ANALOG COMPUTER SIMULATION OF THE F-106A

The maneuvers discussed above must be simulated in the wind tunnel if the MODEL FLY objective is to be satisfied. Analytically the feasibility of maneuvering wind tunnel models can be demonstrated by simulating the airplane on the analog computer and using the necessary inputs to fly the maneuver. This is the approach taken herein, using a simulation of the F-106A airplane.

The equations of motion which best describe the F-106A longitudinal dynamics can be written in a wind axis system for the force equation and a body axis system for the moment equation. The force equation is written as

$$M_A \ddot{Z} = - (C_{L_\alpha} qS/_V)\dot{z} - (C_{L_\alpha} qS)\theta - (C_{L_\delta} qS)\delta \tag{6}$$

and the moment equation is written as

$$I_{yy}\ddot{\theta} = ((C_{M_q} + C_{M_{\dot{\alpha}}})qS\bar{c}^2/2V)\dot{\theta} + (C_{M_\alpha} qS\bar{c})\theta + (C_{M_{\dot{\alpha}}} qS\bar{c}^2/2V^2)\ddot{z} + (C_{M_\alpha} qS\bar{c}/V)\dot{z} + (C_{M_\delta} qS\bar{c})\delta \tag{7}$$

where

θ	= pitch angle	radians
Z	= translation perpendicular to wind axis	feet
M_A	= mass of the airplane or model	slugs
C_{L_α}	= lift curve slope $\partial C_L/\partial\alpha$	1/radians
V	= airplane forward velocity or tunnel airspeed	feet/second
C_{L_δ}	= elevon lift effectiveness $\partial C_L/\partial\delta$	1/radians
C_{M_α}	= pitching moment curve slope	1/radians
$C_{M_{\dot{\alpha}}}$	= $\partial C_L/\partial(\dot{\alpha}\xi/2V)$	1/radians
C_{M_δ}	= elevon power $\partial C_M/\partial\delta$	1/radians
C_{M_q}	= pitch damping derivative $\partial C_M/\partial(\dot{\theta}\bar{c}/2V)$	1/radians

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I_{yy} = pitching mass moment of inertia — slug-feet^2

S = wing area — feet^2

$\bar{c}$ = mean aerodynamic chord — feet

q = dynamic pressure - $\frac{1}{2}\rho V^2$ — pounds/feet^2

The derivation of these equations is found in Ref. 10. The assumptions used in the derivation are: (1) constant forward velocity, (2) small-angle approximations ($\cos\measuredangle \approx 1$, $\sin\measuredangle \approx \measuredangle$), and (3) constant aerodynamic and mass coefficients for the duration of the maneuver. It should be noted that these are the perturbation equations, that is, the airplane is in a trim condition prior to the maneuver. Also it is assumed that the dynamic mount system is perfect; it follows the model motion exactly and hence does not impede its motion.

A comparison between the simulated and true airplane response is shown in Fig. 3. The maneuver depicted is a symmetrical pull out for the F-106A at combat gross weight. The pertinent inputs are listed in the following table.

PARAMETER VALUES USED IN F-106A ANALOG SIMULATION

PARAMETER	VALUE
V	1526
q	1190
M_A	924
c.g.	-.268
C_{L_α}	3.00
C_{L_δ}	.292

PARAMETER	VALUE
C_{M_α}	-.576
$C_{M_q} + C_{M_{\dot\alpha}}$	-.426
C_{M_δ}	-.209
I_{yy}	178,000
S	695
$\bar{c}$	23.75

To achieve the desired simulation, coefficients based on theoretical and on faired flight data were used first. These coefficients were adjusted to match as closely as practical the magnitude, frequency and damping of the real airplane response. It can be seen in Fig. 3 that the simulated response adequately approximates the true airplane or model response.

It should be observed that the airplane has been simulated with linear equations. Therefore, simulated response is not expected to exactly match the real airplane response. However, the simulation does agree quite closely with the real airplane. Thus it may be concluded that the simulation

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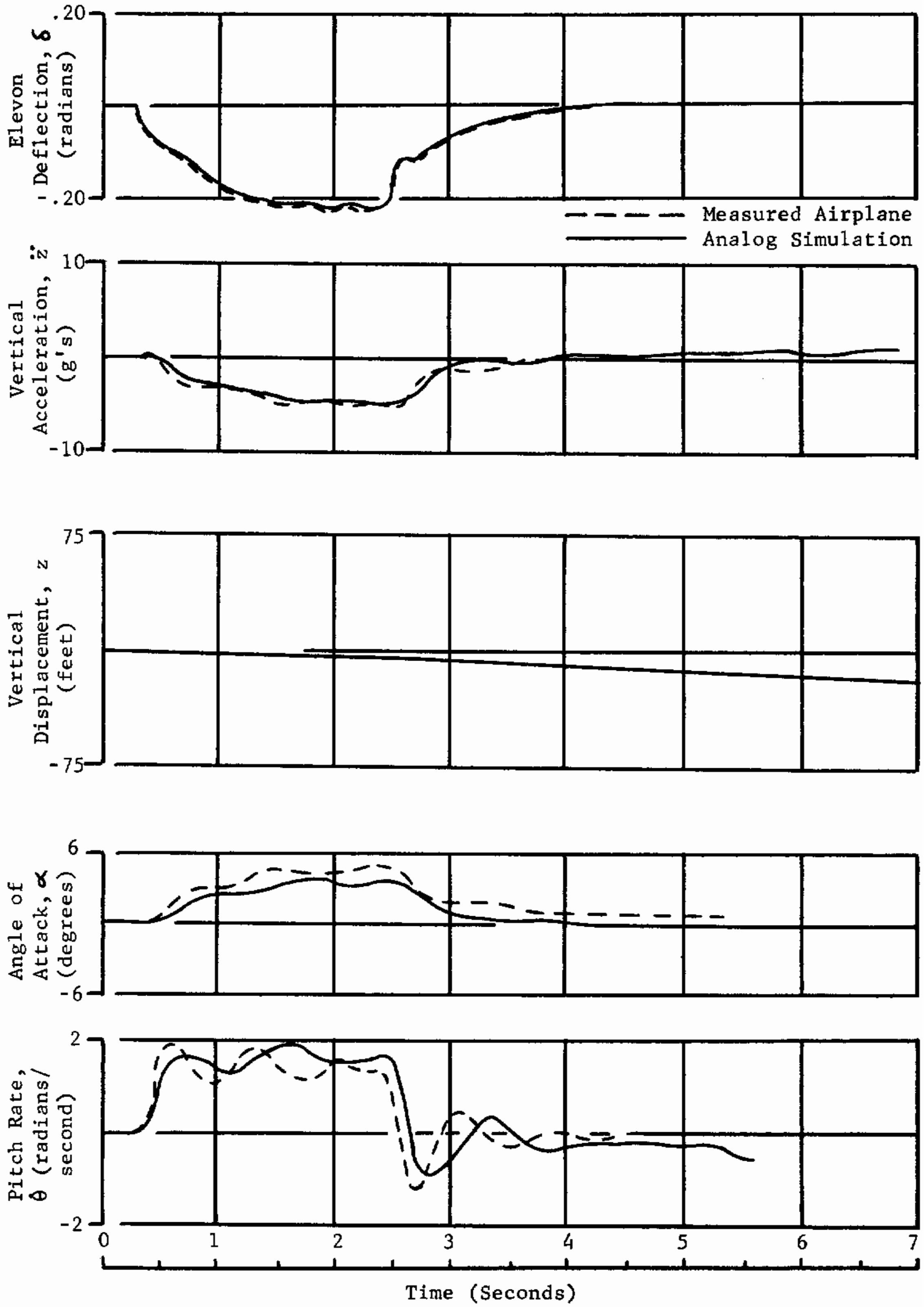

Figure 3 Comparison of Simulated and True Airplane Time Histories

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yields representative predictions of the maneuverability of the scaled model in a wind tunnel.

Other maneuvers are presented in Bibliography Item 8 along with equations, block diagrams, and physical values used for analysis.

4.4 DEMONSTRATION OF WIND TUNNEL MANEUVERABILITY

During the full scale maneuver program the pilot is not generally concerned with the change in altitude required to perform a demonstration maneuver. The primary objective is to achieve the desired load factor. This is evident from the time history of altitude change z, in Fig. 3. The pilot has achieved the load factor but the maneuver has not been checked; the airplane continues to rise. In the wind tunnel this condition is not tolerable for obvious reasons. It is not, however, required to duplicate the exact flight profile. Maneuvers which duplicate the limit load factor and efficiently utilize the tunnel altitude are to be considered. To illustrate efficient utilization of tunnel altitude two different maneuvers were simulated, each maneuver achieving a symmetrical maximum upward acceleration of 7 g's (actually Δn_z = 6 g's from trim). The 7g load factor is the maximum design limit load factor for the F-106A airplane. The purpose of these maneuvers is to demonstrate the most efficient utilization of the limited tunnel height while performing an extreme maneuver.

The maneuver depicted in Fig. 4 is a simple climb-dive maneuver reaching a peak upward acceleration of 7 g's and downward acceleration of 2.25 g's (Note - positive Z is down). The maneuver can be classified as a symmetrical pull out. The maximum displacement (Δz) is 69 feet; this displacement is in real airplane scale. For a 1/10th length scale model, the maximum displacement would be 6.9 feet. This maneuver is easily performed in a 16 foot tunnel by a 1/10 scale model of the F-106A. However to demonstrate how the tunnel space is used more efficiently the 7g maneuver shown in Fig. 5 is performed.

Fig. 5 depicts a dive-climb-dive maneuver achieving 7 g's upward acceleration and 3 g's downward. The maximum displacement differential is 19.5 feet real airplane scale. For a 1/10 length scale model the maximum displacement is 1.95 feet, 1/8th of the available tunnel space.

The maneuvers depicted in Fig. 4 and Fig. 5 were arrived at using an analog computing circuit to program the control surface deflection

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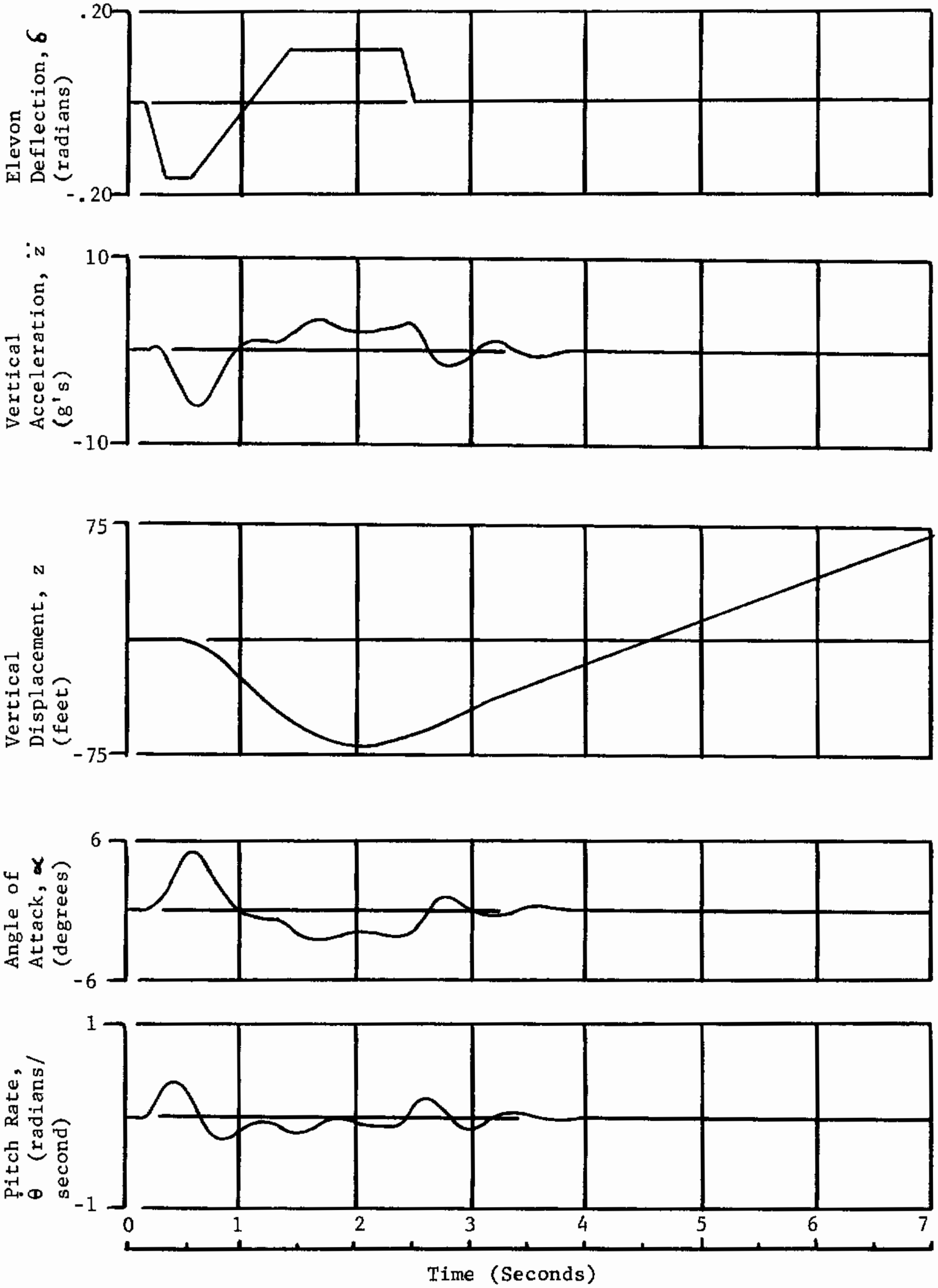

Figure 4 Time History of Simulated Climb-Dive Maneuver

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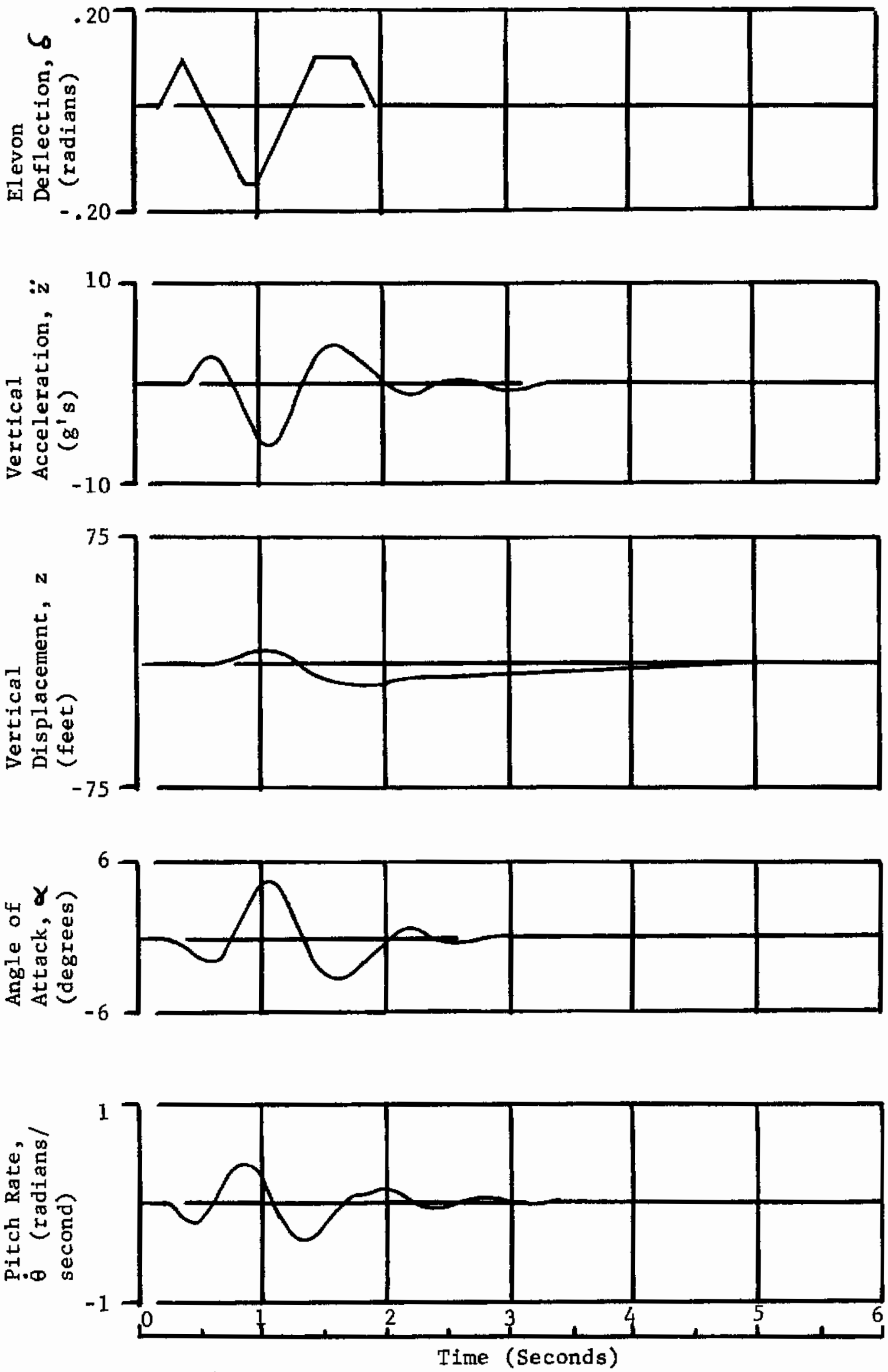

Figure 5 Time History of Simulated Dive-Climb-Dive Maneuver

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input. Basically the circuit provides a means for adjusting the deflection function slopes, break points and upper and lower deflection limits. Final selection of the deflection function was made by iterative adjustments to obtain the desired vertical acceleration. Both deflection functions have slopes and limits within the capability of the real airplane control system.

The example maneuvers demonstrate that maneuvering wind tunnel models are feasible. It is reasonable to assume that with proper planning all symmetric maneuvers can be flown within a 16 foot wind tunnel.

4.5 CONCLUSIONS

The results of these analyses indicate that it will be possible to perform all required symmetric maneuvers in a 16 foot wind tunnel for the F-106A aircraft. Extrapolation to other aircraft and missiles indicates that with proper scaling each will fit within the general capabilities of the mount-wind tunnel combination for similar testing.

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5.0 PRELIMINARY DYNAMIC MOUNT DESIGN

5.1 INTRODUCTION

The primary requirements for the MODEL FLY mount are structural and kinematic in nature. The structural loads imposed by the model under test are rather conventional, except that they can be divided into "normal" and "emergency" requirements wherein the "normal" requirement is unusually small, being due to equilibrium lift and drag force only, excluding large lift forces. The kinematic requirements are of course quite unconventional.

The requirements on the mount can be expressed as a set of ground rules inherent in the task undertaken:

1. The longitudinal force to be supplied to the model during normal operating conditions is a substitute for engine thrust, and ideally should be always directed along the model engine center line, while the fore-and-aft motion of the model center of gravity is constrained to zero. (This provides for a realistic vertical component of thrust when the model is pitched.)

2. A vertical force will normally be supplied to the model during test as a substitute for the gravity force lost because of scaling.

3. An emergency vertical force opposing and exceeding the maximum lift force (typically, 7 times equilibrium lift) should be available.

4. An emergency pitching moment opposing and exceeding the runaway elevon pitching moment should be available.

5. Roll, yaw, and sideslip are not to be intentionally excited, but the mount should constrain these motions and should be designed with a view toward eventual operation in these modes.

6. The design should be aerodynamically clean, structurally sound, and capable of dynamic response sufficient to follow at least the rigid-body modes of the model. It is not expected to follow flutter-induced motions of the fuselage, nor to follow fluttering surfaces.

7. The mount shall be designed on the basis of a model which is scaled to a size appropriate for the 16 foot tunnel, for instance, a 1/10 scale F-106A.

If not for the emergency force and moment requirements and the need for artificial gravity force, the mount requirements might be met by a very light structure, virtually a "broomstick" supplying only a few hundred pounds of force in pure compression while being carried at its front end by

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the model and at its rear end by some light, fast actuating mechanism. The power requirement for such a design would be on the order of 100 horsepower.

If a scheme could be devised to catch or otherwise independently constrain the model, in case of loss of control, this ultralightweight type of mount could be very desirable. However, no feasible design is immediately obvious, and, even if found, would have to be justified against a more conventional type of design, which is therefore being pursued.

When a single mount is accordingly designed for both the high kinematic speeds of normal testing and the high loads of emergency operation, the power capability required expands rapidly; the total is presently estimated at 5,600 horsepower, with perhaps an additional 5,600 horsepower needed to accelerate components inside the various actuation systems. Thus the penalty to be paid for "one-unit fail-safe" design is considerable; however, the approach presently appears to be feasible.

The state-of-the-art of mechanisms is not in general advanced to the point demanded by the above ground rules. A certain size and weight of structure is dictated by the emergency force requirement; then, the speeds and accelerations which this structure must attain are dictated by the "free-flight" model behavior. Together, these conditions specify considerably more manipulative power per pound of structure than is the rule in ordinary practice, except within an unloaded prime mover itself. However, it has been found that there is enough margin so that prime movers can indeed be provided which can manipulate themselves plus the necessary structure. The resulting assembly may extend the present state-of-the-art even while being composed of strictly state-of-the-art components.

5.2 GENERAL CONFIGURATION

Various configurations were considered during the evolution of a practical design to fit the above requirements. All proposals shared the common feature of a sting extending rearward from the model fuselage to structure somewhere beyond the model, so as to avoid aerodynamic interference with the model. All to date have also shared the feature of a vertical track-and-trolley to constrain horizontal motions (both lateral and longitudinal). The former feature seems inevitable; the latter is not, but at least is a good basis for development of one design with which later alternatives may be compared. The present design is not necessarily final.

The main kinematic problem then is how to connect the axial sting

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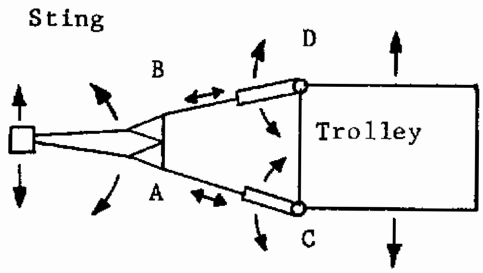
with the vertically movable trolley. A mere hinge would require that the trolley move vertically due to model pitch, at higher rates than those at the model mass center. To avoid this effect, various arc-segment arrangements were considered; these place the pivot center at the model mass center, but are not aerodynamically clean if liberal pitch angles (circa $\pm$ 30°) are to be permitted. Instead, a bifilar arrangement has been chosen, as illustrated by Fig. 6. The adjacent small sketch also shows the principle involved; the two inclined arms AC and BD permit vertical and pitching motions of the sting relative to the trolley. The lines AC and BD converge at the model center of gravity, so that small pitch motions about that point occur without need for trolley motion and with no changes in the lengths AC and BD. Large pitch motions require horizontal actuation of the longitudinals, such motion being able to provide both horizontal motion (foreshortening compensation) and vertical compensatory motion. Desired vertical motion of the model center of gravity can also be provided by the same means, without trolley motion; this seems desirable for high frequency motions, but the preliminary design does not yet utilize this option, by way of conservatism.



The top view is assumed to be the same as the side view as sketched except that all lateral motions are constrained to zero. The design is thus capable of extension to more degrees of freedom, especially if the vertical track is movable laterally for large, slow lateral motions. Roll is constrained by provision of torsionally stiff (yoke-type) joints throughout.

5.3 DESIGN DATA

The various dimensions of a typical test model are given in Table 2. These data are based upon an F-106A aircraft with both full-scale and 1/10 scale dimensions. Model scale is determined from tunnel considerations only in that the model must be sized such that the maneuvers can be executed within the confines of the tunnel. The mount itself will not place greater restrictions on scale than the tunnel. Since the F-106A

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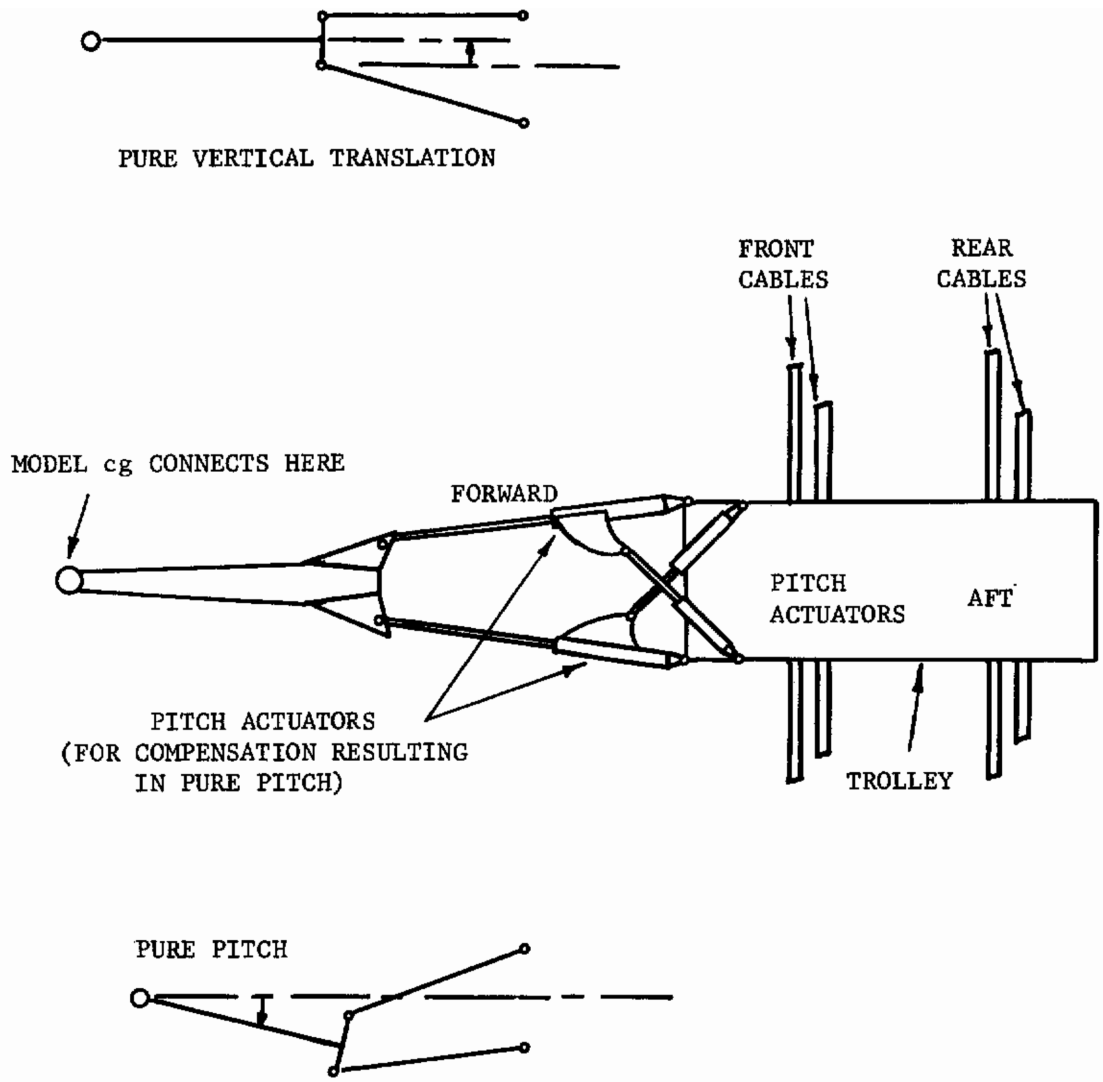

Figure 6 Schematic of the MARK I Mount Configuration

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	F-106A	1/10 Scale Model
Configuration		
Length (inches)	840	84
Wing Span (inches)	458	45.8
Tail Height (inches from center line)	152	15.2
Base Diameter (inches at tail)	48	4.8
Diameter at Center of Gravity (inches)	73	7.3
Location of Center of Gravity (inches from base)	268	26.8
Weight (pounds)	30,000	30
Moment of Inertia (ft-lb/radps^2)	181,000	1.81
Performance		
Equilibrium Lift (pounds)	30,000	300
Maximum Lift (pounds) Up	210,000	2100
Down	90,000	900
Maximum Drag, Approx. (pounds)	21,000	210
Max. Angle of Attack, Approx. (deg)	15	15
Max. Pitch Acceleration (rad/sec^2) Flight Test	2.45	245
Theoretical	2.90	290
Max. Vertical Jerk (g/second) Flight Test	21	210
Short Period Frequency (cps)	1	10
First Structural Frequency (cps)	8	80

TABLE 2 Typical Configuration and Performance (F-106A)

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has a rather small fuselage compared with its wing area, the problems of attachment are as severe as need be considered.

Some fundamental maneuvers of a 1/10 scale F-106A in a 12 foot height (corresponding to a 16 foot tunnel) are shown in Fig. 7. For the most part, these maneuvers are hypothetical and merely establish limits of attainability based on simple acceleration and displacement limits. The two nonclosed patterns do, in contrast, represent scale F-106A maneuvers as presented (versus time) in Fig. 4 and Fig. 5. Conclusions based on Fig. 7 are:

1. In a 16 foot tunnel, using + 60g and - 40g acceleration limits the peak vertical airspeed cannot possibly exceed $\pm$ 130 feet per second; a peak value between 50 and 100 feet per second is more likely, the former being adequate for many maneuvers and the latter being liberal.

2. If perfectly sinusoidal maneuvers at $\pm$ 40g peak acceleration and 12 foot vertical stroke are performed, the frequency must be 2-1/3 cps and the peak vertical speed must be 88 feet per second. The peak vertical speed is less for either higher or lower frequencies, due to the acceleration limit at higher frequencies and the stroke limit at lower frequencies. A 10 cps sinusoid uses only 0.6 feet of altitude and $\pm$ 20 feet per second of vertical velocity.

3. A realistic but unrefined (Climb-Dive) maneuver with a scaled F-106A would use only about $\pm$ 3.5 feet of altitude and 70 fps of velocity. A bit of care in control gives an improved (Dive-Climb-Dive) maneuver using only about $\pm$ 1 foot and 45 fps in the tunnel. Corresponding pitch angles, incidentally, are about 6 degrees in both instances. The mount capability tentatively being provided is liberal by these standards.

Figure 7 illustrates the effects of pitching acceleration and pitching velocity upon the sting tip (or model center of gravity) only. For a sting five feet long, an increase of as much as 80% in the illustrated accelerations and velocities may be required at the junction between the sting aft end and the pitch actuators (points A and B in the sketch on page 38). With the chosen design, the trolley is not subjected to this incremental increase in acceleration or velocity.

A rather liberal design is one which matches all the 1/10 scale F-106A performance parameters of Table 2, and further has a vertical speed capability (at the model mass center) of about 80 feet per second, a

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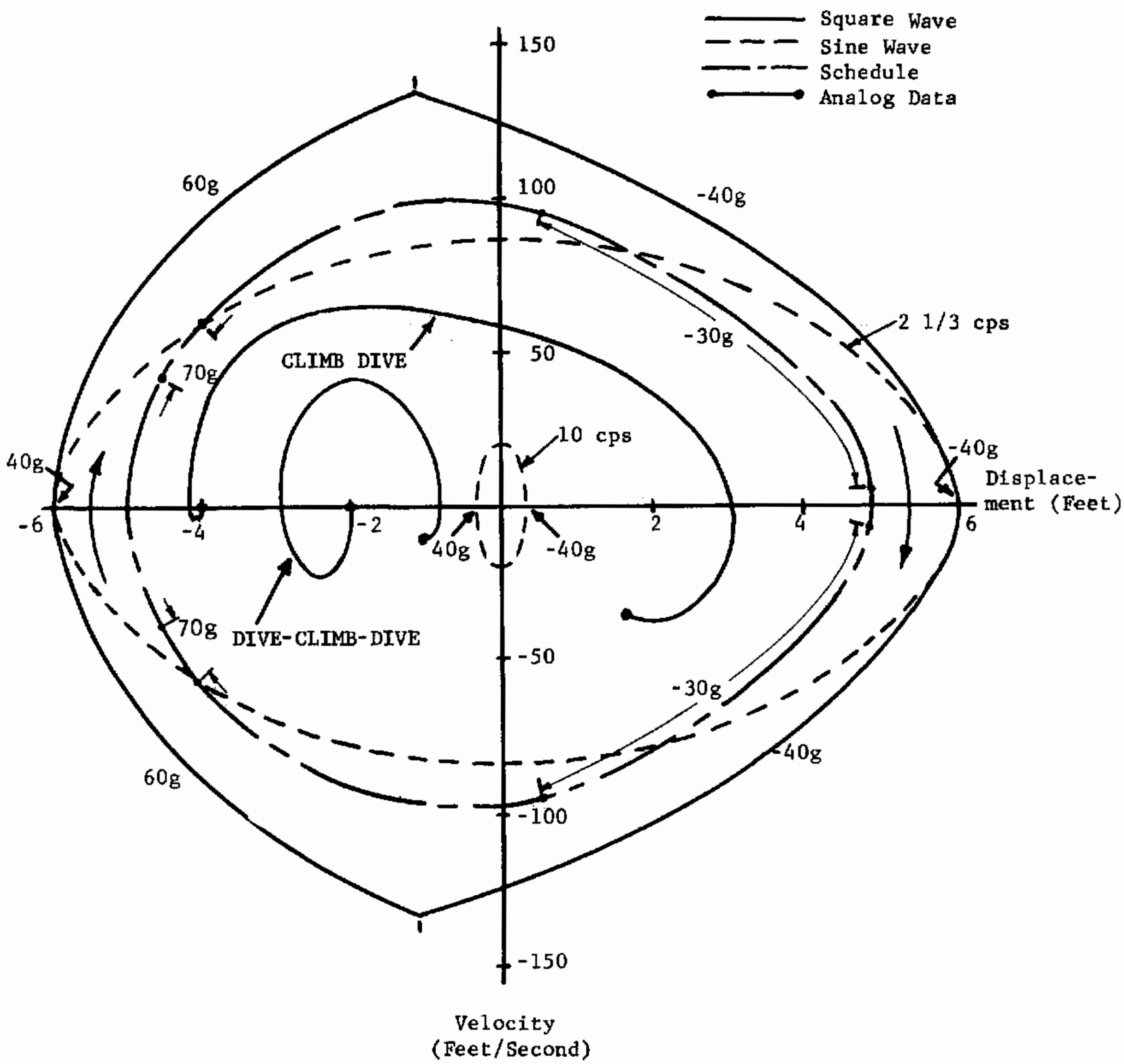

Figure 7 Model Maneuvers in the Phase Plane

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vertical stroke of about 10 feet, and a pitch stroke of about $\pm 20^{\circ}$ (say $+25^{\circ}$, -15°). Reductions in the latter figures can be tolerated if it is desired to limit the design more closely to truly practical maneuvers. In addition to kinematic requirements, the table specifies the maximum lift as 2100 pounds; multiplying by the standard AEDC safety factor of 4 gives a sting tip vertical load of 8400 pounds to be met (at yield) by the mount structure. Note that this is a conventional static design factor assumed for convenience in preliminary design, and could provide too much strength and/or too little stiffness in the final analysis. A complete static/dynamic analysis will no doubt alter this design factor.

5.4 STATUS OF THE MOUNT DESIGN

A mount design is presently available which meets the primary structural requirements and probably can be actuated to meet the kinematic requirements. With regard to stiffness and vibration, an analog study is in progress, but results are not yet available.

A detailed description is not needed, but it is probably of interest to mention some of the sizes and weights of structural members of the mount. All members are 7075-T6 aluminum unless otherwise specified. Proceeding from the front rearward, these are:

Load Cell and Sting End Fitting - Length 4 inches. Weight 2 pounds.

Sting (including four foam-filled fins, and journalled steel fin tips) - Length 61 inches. Weight 36 pounds.

Rod Ends and Pins (4) - Weight 9 pounds.

Longitudinals (first 50 inches only) - Weight 50 pounds.

Trolley and Eight Actuators - Approximate weight 200 pounds.

These weights total approximately 300 pounds, open to revision after further study of trolley structure and actuator sizes in particular. To accelerate such a mass at 70 g's, which is the F-106A capability without any vertical load bias, therefore requires a force of 21,000 pounds applied to the trolley; application of this force at a vertical speed of 80 feet per second requires either the absorption or the delivery of slightly over 3000 horsepower. There are viscous clutches available which, driven by flywheels, can meet these requirements when connected via drums and cables. The actuation of the sting relative to the trolley can be accomplished hydraulically. Alternative means of actuation have been considered but rejected (see Section 6).

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5.5 FUTURE DIRECTION

Aside from the analog study of vibration the design of the present mount has been carried to a point at which it is appropriate to start consideration of a Mark II version. The Mark I version described in this report appears feasible but does require great amounts of power, due to weight. It is currently believed that weight and required power can be reduced by eliminating the trolley while extending the longitudinals, anchoring each to a bracket on the tunnel structure. Further, it is believed that radial cables can be attached directly to the rear end of the sting to locate that point, relieving the longitudinals of bending moments used for sting translation.

Future efforts will investigate such a Mark II configuration, pursuing both Mark I and Mark II to a point where one or the other is clearly shown to be superior, and then continue the superior design to its completion.

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6.0 INERTIA CANCELLING DEVICES

6.1 INTRODUCTION

The successful achievement of wind tunnel free flight requires that the wind tunnel and the model supporting equipment exert no appreciable influence on the model response characteristics. To truly simulate free flight, the model should be sized to minimize wind tunnel interference effects on its aerodynamics and mounted in a way such as to provide only a simulated thrust vector. In addition, since dynamically scaled models do not truly scale weight, it would be desirable for the mounting system to provide a vertical force which compensates for the weight nonsimulation (Ref. 1 and 2). The dynamic mounting system which permits the model to safely maneuver in the wind tunnel and is capable of satisfying the above requirements will probably have considerable moving mass compared to the model mass. For the model to fly freely a device is required to provide a force of phase and magnitude that will move the mount precisely along with the model. The development of this device is essential to the successful design of a maneuvering dynamic mount.

The device described above is defined as an Inertia Cancelling Device (ICD). By using an ICD, unrealistic model response due to extraneous inertial forces imposed by the mount will be eliminated or minimized. Various types of inertia cancelling devices have been considered for use in the MODEL FLY program and show various levels of feasibility, as recounted below.

6.2 RELATIONSHIP TO THE STATE-OF-THE-ART

The ICD task is one of actuation of the mount mass utilizing a system with considerable power and excellent dynamic response. There is at least one method currently well developed, namely the method of the hydraulic (or pneumatic) servo mechanisms. Other less conventional methods can be envisioned and deserve investigation. The flow of air past the mount represents a source of power which might be tapped (indeed was tapped subsonically) by means of servo wings. Transonic conditions are obviously more complex, requiring a re-evaluation of this method. Other methods which have been used elsewhere for controlled force application and might be suitable here, are:

1. Fluid Jets
2. Viscous Clutches
3. Propellers (Variable Pitch)

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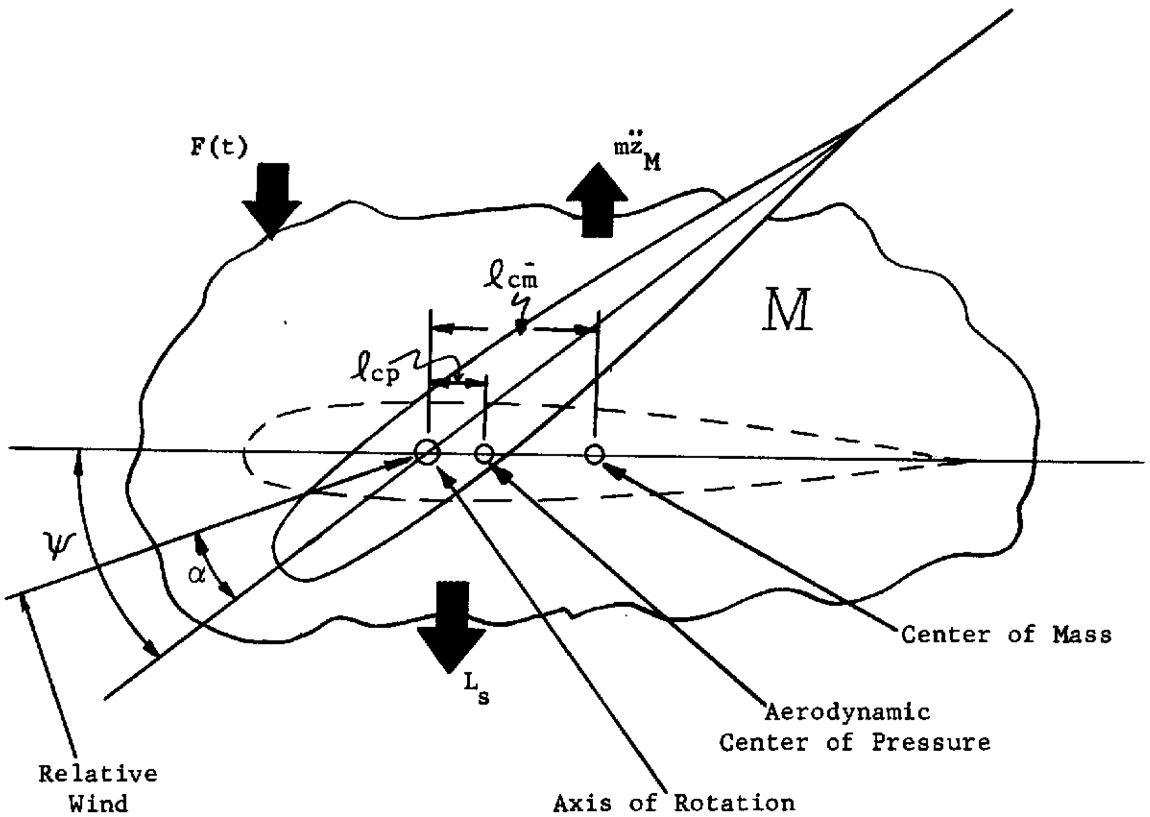
The five methods thus available will be discussed in the following section.

6.3 METHOD OF INERTIA CANCELLATION

6.3.1 Servo Wings - The name servo wings is misleading. The servo wing is not an externally controlled lifting surface. The mechanism whereby the inertia nulling force is controlled is seen from the accompanying sketch. A servo wing is a pendulous lifting surface (center of mass is located aft of the pivot point or axis of rotation). In the sketch, the



force F(t) is a forcing function derived by the model. The force accelerates the mount mass M. Because the center of mass of the servo wing is eccentric to the axis of rotation it will lag any translation of the axis of rotation resulting in a deflection of the servo wing. The lifting force thus created is proportional to the deflection of the wing.

Proper adjustment of the system geometries insures sufficient force to null the effects of the mount inertia reaction on the model.

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The servo wing concept was demonstrated subsonically (see Ref. 1). The feasibility of the servo wing in transonic flow has been studied (Bibliography Item 1) and the results of the study are briefly summarized.

In Bibliography Item 1 a dynamic analysis of the servo wing system is presented. The analysis considers a two degree of freedom system, vertical translation and servo wing rotation. The model dynamics are not included in the simulation. The model is assumed to be rigidly attached to the mount mass, M_{mt}, and to contribute only an arbitrary force input, F(t) (see sketch p. 46). Both the servo wing and mount are considered rigid.

The equations of motion for the system in Laplace operator form and matrix notation are:

$$\begin{bmatrix} C_{L\alpha} qS + ml_{cm}s^2; & C_{L\alpha} q \frac{S}{V} s + (M+m)s^2 \\ C_{m\alpha} qS\bar{c} + (C_{m_q} + C_{m\dot{\alpha}})q \frac{S\bar{c}^2}{2V} s & C_{m\alpha} qS\frac{\bar{c}}{V} s + (C_{m\dot{\alpha}} qS\frac{\bar{c}^2}{2V^2} - ml_{cm})s^2 \\ -(I_{cm} + ml_{cm}^2)s^2; & \end{bmatrix} \begin{Bmatrix} \psi \\ Z_M \end{Bmatrix} = \begin{Bmatrix} F(s) \\ 0 \end{Bmatrix}$$

From the servo wing rotation per unit acceleration transfer function it is possible to show that the ratio of the aerodynamic static margin (l_{cp}) to center of mass eccentricity (l_{cm}) is a most important servo wing parameter. For ideal servo wing performance the acceleration of the mount must equal the acceleration sensed by the model mass alone when acted upon by a force. This requirement dictates that the transfer function must have a static gain of unity and that the following relationship holds

$$l_{cp}/l_{cm} = \frac{m}{M + m - M_{mod}}$$

System stability considerations impose the following limits:

$$\left(\frac{m}{M+m}\right) < \left(l_{cp}/l_{cm}\right) < \left(\frac{I_{cm}}{ml^2_{cm}} + 1\right)$$

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The upper limit is set by the requirement of zero or positive total system damping and the lower limit by the requirement of zero or positive total system stiffness.

Obtaining a high system natural frequency is a difficult servo wing design problem. Fig. 8 shows the effects of variations in l_{cm}, I_{cm}, and C_{m_q} on the undamped natural frequency of the system. Parametric variations are made maintaining a static gain of unity. The varied quantities C_{m_q}, I_{cm}, l_{cm} and the system frequency are normalized by the indicated nominal values of each so that a percentage frequency increase or decrease is assessable for a corresponding percentage change in the varied parameter.

This plot indicates that all three parameters can be changed in size to increase ω_D. I_{cm} should be made as small as possible. There is of course physical limitation on this requirement and the benefit of an I_{cm} decrease is small. If it were possible to reduce I_{cm} to .1 of the nominal, the system natural frequency would increase only 50%. A C_{m_q} of (-40) or (10) times nominal increases ω_D only 27%. ω_D^2 is directly proportional to C_{L_α}; but the prime consideration in servo wing design is for minimum center-of-pressure shift. Therefore the system designer must live with the C_{L_α} value of the planform which has minimum c.p. shift.

A minimum c.p. shift is a requirement because of servo wing sensitivity when the geometry is adjusted to meet the static gain requirement. Fig. 9 depicts the system sensitivity. For this plot parametric variations are made without constraint on the static gain. Any shift in static margin near the nominal unity static gain (i.e. K = 1) operational point is critical.

A sensitivity of the servo wing to aeromechanical changes brought about a program to determine suitable servo wing planforms. A systematic search was made of available test data for various planforms. Wings of "M" and "W" planform seemed favorable; the c.p. remains fairly constant because the forward shift of load on one panel is generally balanced by a rearward shift on the other panel. A few of the more promising wings were tested in the 1 foot AEDC transonic tunnel.

The wings tested did not yield satisfactory c.p. characteristics. From Fig. 9 a maximum allowable change in c.p. versus angle of attack might be .25% or .0025 chord. For the "best" wing tested, an "M" type

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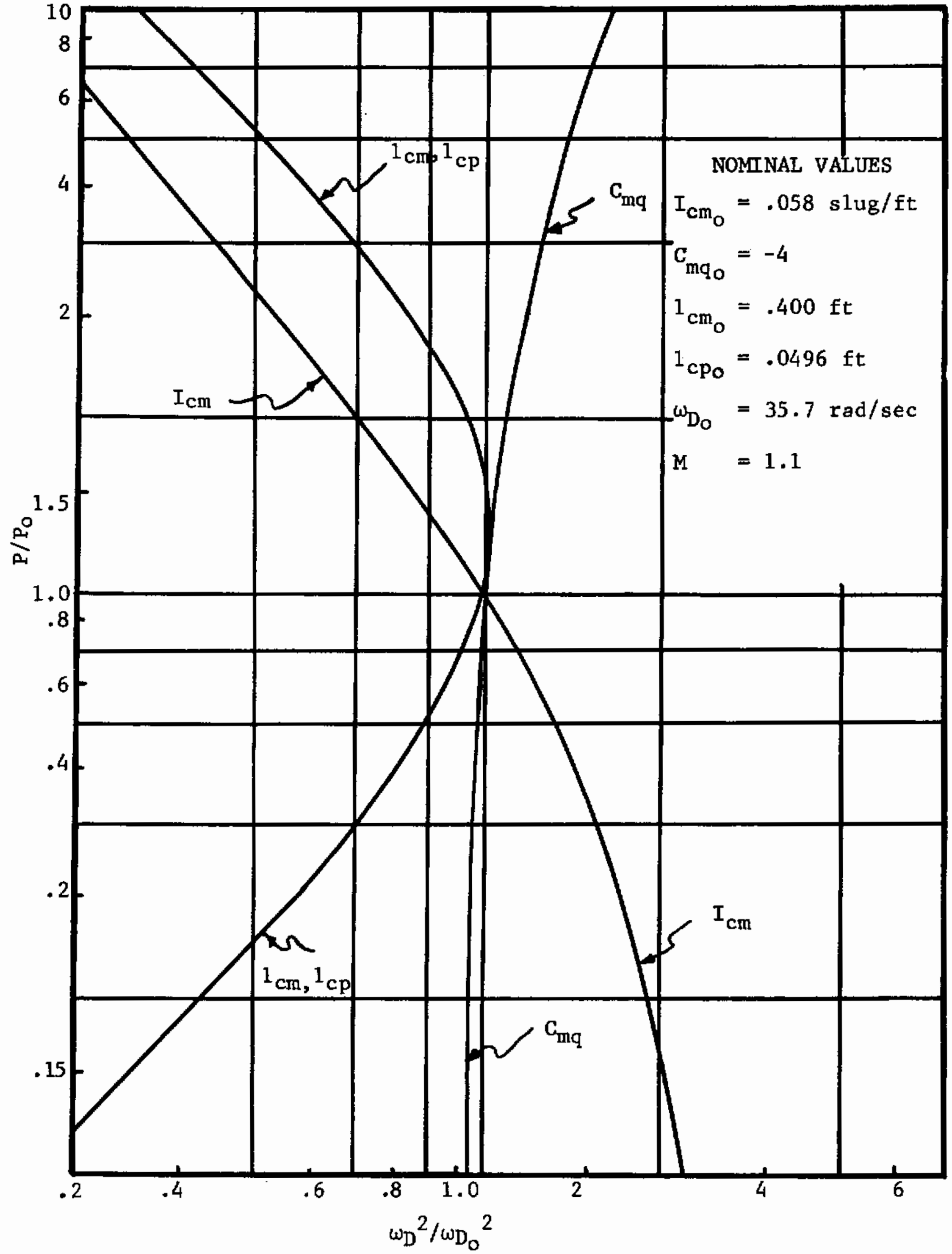

Figure 8 Effects of Parameters on Undamped System Natural Frequency Normalized by Nominal Configuration Values (Mach = 1.1)

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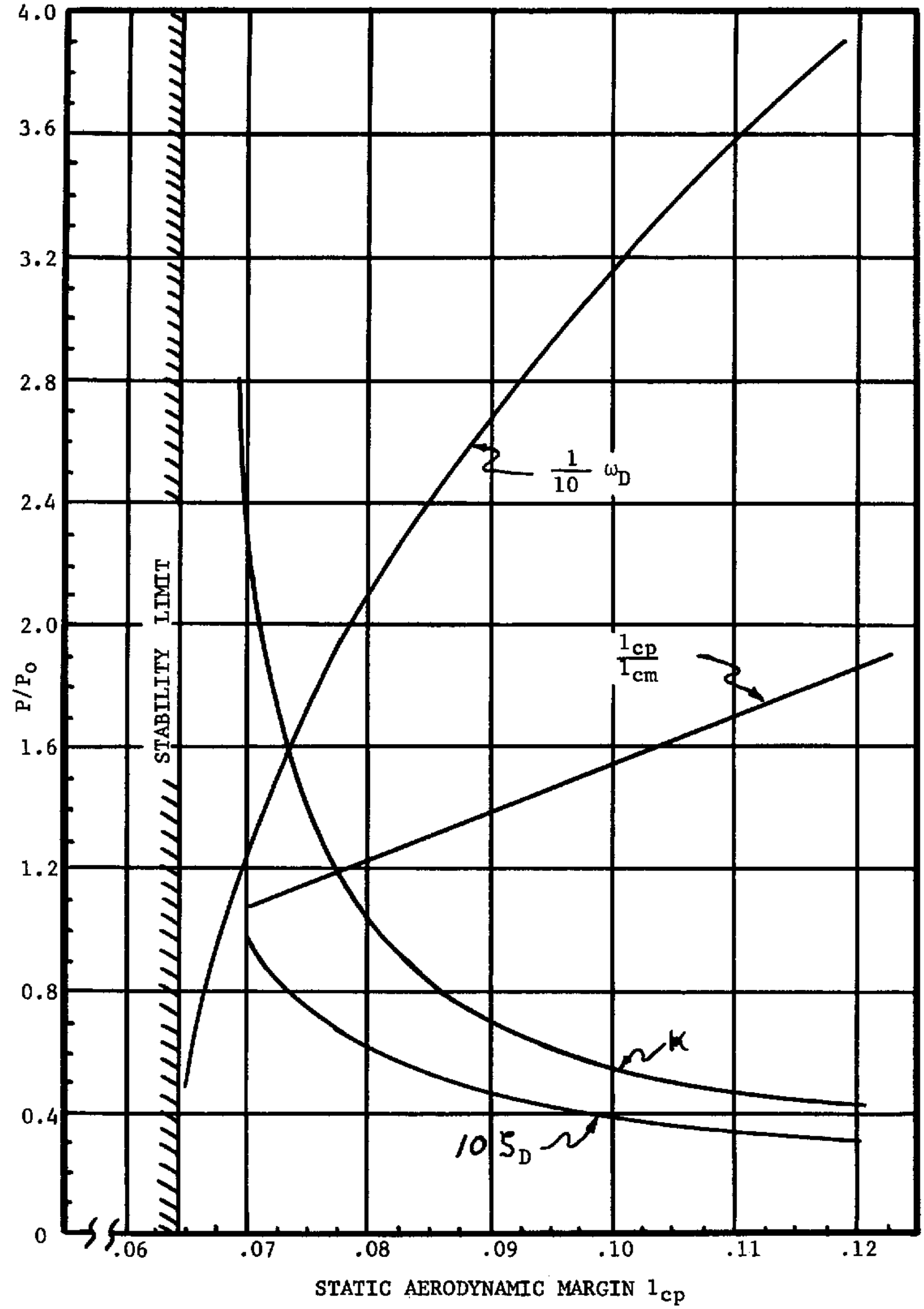

Figure 9 Sensitivity of the Servo Wing to Variations of Static Aerodynamic Margin

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wing, the minimum measured c.p. change was approximately 1% c at Mach 1.2. The results varied with Mach number from 3% at Mach .7 to 9% at Mach .95. All of the wings tested depict the same trends, but of course, magnitudes of c.p. shift versus angle of attack were different.

In summary, the transonic servo wing is not a feasible ICD. Its sensitivity to small aeromechanical changes is critical when adjusted to meet the static gain requirement.

The resulting requirements for static margin for l_{cp} shift with angle of attack are so extremely small that it is doubtful that a wing planform is available to meet the requirement, especially in transonic flow. Also, the inability to adequately increase servo wing natural frequency makes it difficult to obtain good system performance.

6.3.2 <u>Fluid Jets</u> - The suitability of fluid jets for inertia cancellation is discussed in Bibliography Item 10; presented here are the principal elements of that report.

The use of either liquid or fired jets would cause excessive tunnel contamination. The jet must move through a distance to do work on the mount, and thus in general would move out of alignment with a tunnel exhaust scavenger that is available at AEDC.

The remaining possiblity is an unfired jet using compressed air which is available in rather large quantities and high pressures at the test site. Thrusts from 1,000 pounds to 22,000 pounds can be developed using simple (converging) nozzles of 1 inch to 3 inches outlet radius with supply pressures of 1100 psi (normally available) to 3000 psi (potentially available). The thrust per rate of flow is virtually independent of pressure and only moderately dependent on thrust or nozzle size; it ranges from 47 pounds per pound/second for the larger nozzles to 57 for the smaller nozzles (in effect, a specific impulse of 47 to 57 seconds). As the available steady rate of flow is on the order of 100 pounds per second for short periods at 1100 psi (less at higher pressure) the maximum sustained thrust available without accumulators is on the order of 5,000 pounds.

The nozzles used would have to be variable (e.g. rectangular with movable sides) for sonic flow at all flow rates. Such nozzles would require development at least for the larger sizes.

The mass to be driven by these jets is the total mass of the trolley and sting, about 300 pounds. The acceleration needed is about

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70 g's; the thrust needed is therefore 21,000 pounds, available only from the largest (3") nozzle and the highest (3000 psi) pressure. The mass flow rate would then be 21,000/47 = 445 pounds per second. As the supply is only about 40 pounds per second at the higher pressure, this flow must be provided from an accumulator.

To operate for 2.5 seconds at full thrust (or for longer periods at less thrust, the limit being for approximately 175 seconds acting against one "g" supporting the weight only) would require 1100 pounds (71 cubic feet) of air at 3000 psi. To supply this flow, a spring-loaded accumulator would have to be 2 feet in diameter by 22.5 feet long, and exert more than a million pounds of spring force, all of which is impractical. A simple pressure tank charged initially to a one-sixth higher pressure (3500 psi) would have to contain about six times the output volume; this is technically feasible but would be prohibitively expensive.

Thus it is technically but not economically feasible to actuate the 300 pound trolley by means of compressed air jets. The cost of the necessary accumulator, the variable opening nozzle, and other items is excessive.

Assuming that the trolley is actuated by some other means, jets still might be used for sting actuation (to induce motion of the sting relative to the trolley). The longitudinal rod end forces required in the baseline design are 13,000 pounds static (which could be provided by a brake) and 3700 pounds dynamic, at each of at least two locations. Assuming a brake is used, a relatively small accumulator would be sufficient. However, hydraulic actuation seems much more attractive and reliable for this purpose.

Excluding safety considerations, the largest vertical force required for sting actuation is equal to the mass of the sting times its acceleration, or about 2200 pounds applied vertically at the junction of the rod ends. This force could be supplied by jets without accumulators, but there is little reason to supply both oil and high volume air to the trolley; oil can do both jobs while air is a doubtful candidate.

A further factor in the use of jets (especially in smaller sizes) is signal to noise ratio; this factor may weigh heavily against the jets.

The response time of a jet depends first on the means of actuation of the nozzle, which can be hydraulic (or pneumatic) and quite

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fast, and second, upon the time required to accelerate the mass of the fluid in the supply line. With the help of a nearby accumulator this time can be quite brief, so that a jet cannot be rejected on this basis alone.

6.3.3 Propellers - The variable pitch propeller can deliver large, rapidly modulated amounts of thrust, but from its necessary external location cannot be readily connected to the MODEL FLY mount. Further, there is the problem of either accelerating the mass of the engine along with the propeller or of driving the propeller from a stationary engine. Except as a last alternative this type of device can be disregarded.

6.3.4 Fluid Controls - Fixed displacement motors, actuated by fluid under pressure, are considered as a power source. Of this type, hydraulic servomechanisms are the most conventional method for the finely controlled application of large forces at high speeds. Pneumatic servomechanisms are in the same category but offer no advantages for present purposes and are less well developed.

The power required for ICD functions depends upon actuation area, dependent in turn upon pressure available and load required, and upon maximum speed. There is no relief in design requirements if the load and the speed are not simultaneous. Excess pressure will merely be wasted at the valve (or diminished at the pump) without reducing the required flow rate.

The amounts of power required in the various actuation modes are listed below. Power requirements are based on the dynamic loads resulting from consideration of the required acceleration and estimated mount inertia, or if greater, for the static loads based on four times the maximum lift of the model. Loads within the actuators are excluded. The corresponding amounts of flow required at a motor pressure of 2500 psi are also given.

1. For vertical actuation of the 300 pound trolley at 100 feet per second and 70 g's, - 3820 horsepower (2180 gpm)

2. For pitch actuation of the sting relative to the trolley at maximum pitch rate, against emergency static load - 361 horsepower (207 gpm)

3. For longitudinal actuation of the sting relative to the trolley during a sinusoid which reaches maximum pitch rate, against emergency static load - 1420 horsepower (812 gpm); or, against dynamic load only - 392 horsepower (224 gpm)

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The latter alternative is relevant because it is permissible to suddenly brake the longitudinal motion to a stop as an emergency action. The model becomes constrained, in this case, to essentially pure pitch motion relative to the trolley. This allows design of these actuators for only the dynamic load.

Horsepower of less than 1,000 can reasonably be obtained from hydraulic actuation. Valves are well known for flows up to about 60 gpm, but are also on the market for flows up to 500 gpm (875 horsepower).

Also, the loads above can and, in some cases, must be shared by several actuators and valves. There are four pitch actuators acting in pairs, and likewise for longitudinal motion. This permits the use of smaller valves. Although allowance must be made for inertia in each actuator, it is thus evident that pitch actuation and dynamic longitudinal actuation can be performed hydraulically.

The static longitudinal braking as well may be performed hydraulically. A larger actuator can be connected in tandem with each dynamic actuator; then, oil which normally recirculates from one end of this actuator through a line and wide open servo valve to the other end can be shut off by centering the servo valve. Although the flow is high, it is at least theoretically possible (and probably feasible) to use the valve with two flows in parallel rather than in series. This gives one-eighth the pressure drop normally associated with the same flow in the same valve.

The greatest barrier to continued attractiveness of this approach is the weight of the components. Based on their volume, piston type actuators carried by the trolley may weigh almost 100 pounds. The large valves discussed above may weigh about 14 pounds each. There could be twelve such valves in the scheme as described. The resulting weight would be 180 pounds for valves alone, all on the trolley. It may be possible however to reduce the number of valves, to find smaller suitable valves from other vendors, to relax the system requirements, and to allow some increase of trolley weight.

The trolley should also carry some amount of accumulator capacity. The volume-time histories indicate that 1 to 3 gallons of accumulator capacity on the trolley would be very beneficial and would permit either (1) much smaller hydraulic supply lines to the trolley, (2) a pneumatic (nitrogen) supply line used to alternately charge two accumulators

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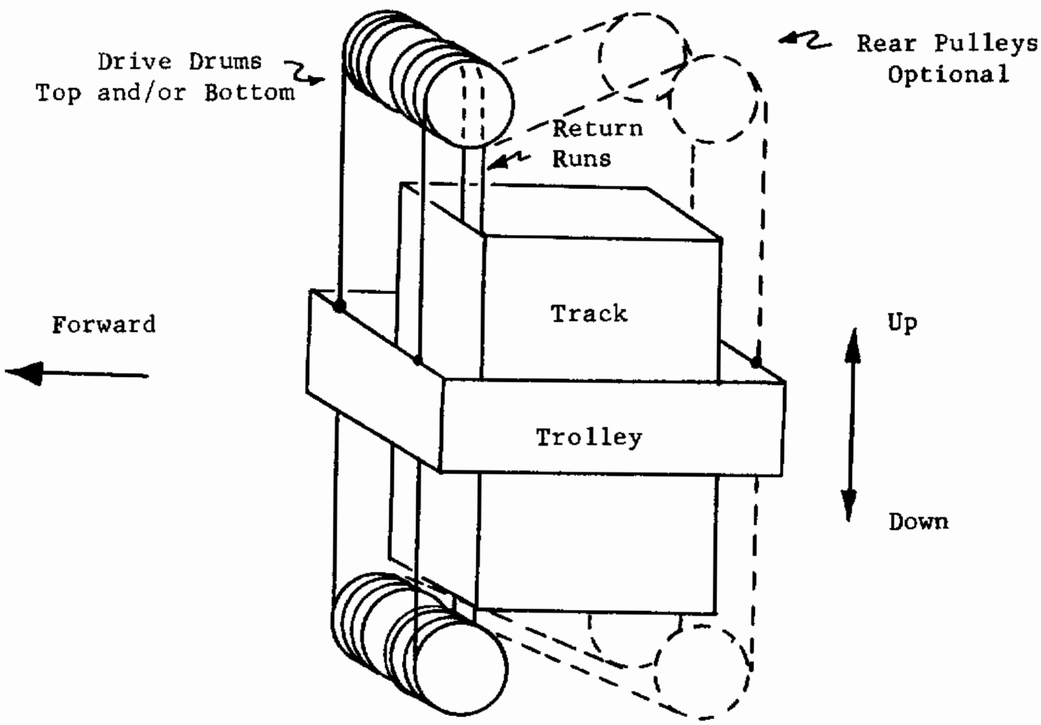
or to drive a pneumatic motor and pump, or (3) a pump or pumps driven either by air turbines or, through one-way clutches, by a roller chain either anchored to the structure (while the trolley moves) or driven by a motor pulley. Any one of these approaches might prove best in the final analysis.

In summary, fluid controls are not feasible for vertical actuation nor for multi-purpose longitudinal actuation because of excessive flow rate requirements. However, they are quite feasible for pitch actuation and for modified longitudinal actuation, that is, for longitudinal dynamic actuation backed by a brake for emergency constraint. (Such an arrangement is a safety feature in itself aside from flow rate benefits.) The use of accumulators in the trolley seems advisable.

6.3.5 Viscous Clutches - A motor, flywheel, clutch, and cable drive arrangement is a logical candidate for vertical actuation of the MODEL FLY mount. A cable drive arrangement is sketched below.



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The largest viscous clutch available off the shelf develops an output of 972 horsepower when fully engaged at maximum speed, rated conservatively for industrial use. These clutches can be up-rated by a factor of from 2 to 4 for military use (as in catapults). This indicates that eight such clutches (four "up" and four "down") may readily supply the 3820 horsepower necessary for vertical trolley actuation. Each clutch would be actuated by a small hydraulic servomechanism; in essence, the viscous clutch offers an order-of-magnitude increase in the third stage power output of a servo system, with negligible loss in frequency response compared with other third stages. The only barrier here seems to be economic; it would be desirable to decrease the total mount inertia in order to reduce the investment in high power actuation, but there is no primary technical limitation when viscous clutches are applicable.

There has been no mention above of using viscous clutches for pitch and longitudinal actuation. The weight of clutch, flywheel, and motor on the trolley would be prohibitive, and mechanical transmission to the trolley would be unfeasible.

6.4 PRESENT STATUS

The inertia cancellation device of apparently greatest merit consists of a set of hydraulic servomechanisms on the trolley, and a cable-and-drum trolley drive from a set of viscous clutches. The primary problem areas seem at present to be the weight of the on-trolley system, and the cost of the off-trolley system. These factors are sufficiently well understood at present to permit turning to some fundamentally different type of mount design and proceed rapidly to a point of valid comparison.

6.5 FUTURE DIRECTION

Due to the status just summarized, current plans are to designate the system studied to date as a Mark I model and proceed to a Mark II model. The premise in the Mark II will be elimination of the trolley, made possible by extension of the longitudinals to a length at which the possible vertical motion of the sting is adequate. This change will eliminate the problem of trolley actuation and excessive trolley weight, and will permit the consideration of viscous clutches in competition with hydraulic actuators as means of actuation for pitch and longitudinal motion.

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7.0 PRELIMINARY INSTRUMENTATION CONSIDERATIONS

7.1 INTRODUCTION

This Section is a preliminary analysis of the instrumentation requirements of the MODEL FLY program. In particular it considers the instrumentation associated with data transmission into and out of the wind tunnel, mount control, and structural response and flight response measurement.

Some of the hardware required lies at the fringe of the state-of-the-art. In some instances it is currently under development, and in other instances off-the-shelf hardware will require modification. In any case the hardware requirements are such that they can be met.

The problems presented by each of the above areas are defined and solutions outlined.

7.2 IN-OUT WIND TUNNEL INSTRUMENTATION

The communications link into and out of the wind tunnel must transmit information associated with the following:

1. The mount control, control response, and safety.
2. The model control, control response, and safety.
3. The measurements of the model structural response and flight response.

Fig. 10 is a generalized, in-out wind tunnel information flow chart. It depicts the various methods of transmission of information that are applicable. The information transmitted for the control and safety of both the model and mount must be in a form that can be utilized in real time. The measurements of a few selected parameters of the model structural response, flight response, and mount performance must be transmitted in a form that will allow recording for (nearly) real time visual monitoring of the flight test. The remainder of the information transmitted can be in a form suitable for off line data processing.

Information transmission linking the mount can be practicably achieved simply by analog parallel hardwire. Information transmission linking the model cannot be practicably achieved by analog parallel hardwire. It would require an unwieldy cable which would greatly limit the free flight maneuverability of the model. The bulk of the model cable is due to the flight loads measurements.

To reduce the size of the model cable link, a single wire or

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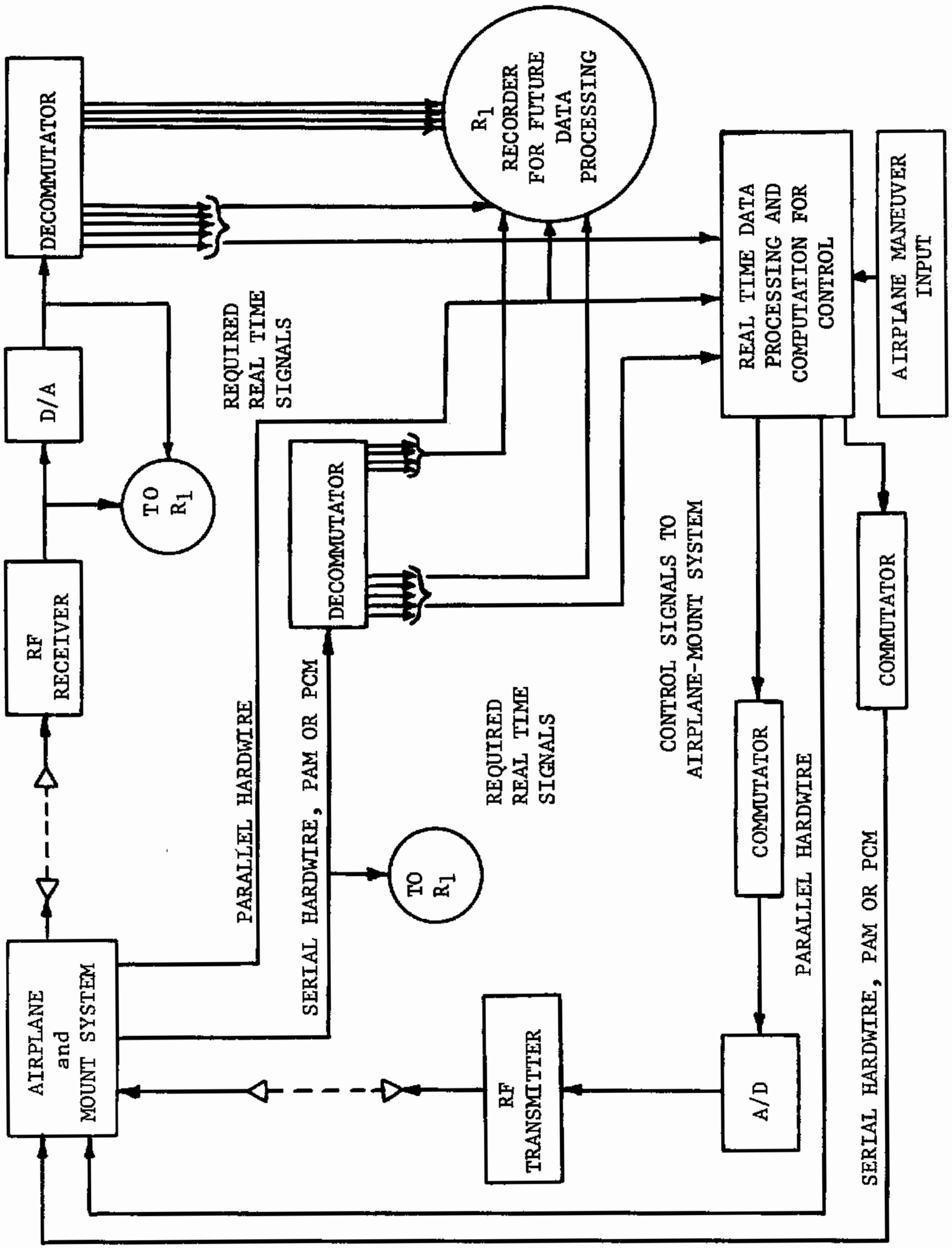

Figure 10 Generalized Chart of Information Flow into and Out of the Wind Tunnel

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possibly a few wires may be time shared or a radio frequency (rf) transmitter may be used. In either case, pulse code modulation (pcm) should be used for high signal to noise ratio. The availability at the wind tunnel of the required telemetry equipment for the above methods was discussed during a visit to the Aerodynamics Branch, Propulsion Wind Tunnel at the Arnold Engineering Development Center. It was learned that no telemetry equipment capable of satisfying the requirements was available nor would it be available in the foreseeable future.

Discussions with vendors of telemetry equipment have been conducted to establish the present state-of-the-art of off-the-shelf hardware that satisfies the number and type of signals that must be transmitted. It has been concluded that the most efficient size and weight telemetry package off-the-shelf for pcm hardwire is 5" x 6" x 6" and weighs approximately 11 pounds. For rf transmission of the pcm signal an rf transmitter approximately 3 cubic inches and weighing 5 ounces must be added to the telemetry package. One vendor has expressed the belief that it may be possible to reduce the size and weight of a telemetry package by special packaging techniques. However, no figures have been available because the package will depend a great deal on the vibration and g loads it can support, thus requiring an iterative design/test cycle.

It is concluded at this time that the most practicable solution is to time share a single wire using pcm techniques to extract the structural response and flight response measurements from the model. Thus the size of the model cable will be greatly reduced.

The packaging of existing off-the-shelf telemetry equipment will have to be altered to achieve a weight reduction; size does not appear to be a problem. A cursory review of the model weight requirements results in this opinion. Accurate weight requirements for the telemetry package and all other on board equipment cannot be fixed until the final design stages of the model.

7.3 MOUNT INSTRUMENTATION

Mount instrumentation is divided into three categories.

1. Mount instrumentation for mount control.
2. Mount instrumentation for mount and model safety.
3. Mount instrumentation for measurement of model flight response.

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Mount instrumentation for mount control involves the generation of the appropriate error signals and operation on these error signals to achieve the desired correction.

Shown in Fig. 6 is a schematic of the mount as it is being considered. It should be noted that two modes of actuation are utilized: (1) vertical translation actuation by means of actuators acting aft, and (2) pitch actuation by means of actuators acting forward. The pitch mechanism forward has been designed so that pure pitch can be obtained free of any resultant vertical translation of the model. Thus, the mount is designed to be mechanically noninteracting, i.e. pure pitch or pure translation can be obtained without one affecting the other. Small amplitude, high frequency vertical translation corrections can be made by the pitch actuators in addition to their regular pitch correction function. Thus large amplitude, low frequency correction can be left to the larger, aft vertical translation actuators. Simple electronic computing circuits will be used to operate on the error signals to generate the desired interaction described above.

Fig. 11 is a block diagram of mount control. It depicts error detection (detection of the reaction between the model and mount), operation on the error, and corrective actuation. The operation is as follows:

Assuming trim conditions, elevon position as a function of time is put into the model. As the model begins to leave its trim position, it begins to react with the mount (Z_A and θ_A begin to differ from Z_M and θ_M respectively). Thus, reaction forces F_Z and M_θ are generated. These reaction forces act also on the mount as shown and in addition are converted to signals fed to the analog computer. The computer operates on these signals generating the desired, modified error signals that are then sent to the actuators. The actuators act on the mount to make Z_M and θ_M approach Z_A and θ_A, thus maintaining the difference between Z_M and Z_A and θ_M and θ_A below a prescribed small value which ensures the free flight simulation of the model. A continuous real time recording will be made of the error signals for visual monitoring of the flight test validity. In addition, alarm circuits will be incorporated to sound an alarm if the error signals should swing outside their prescribed envelope.

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At present two error signals will be generated: (1) a signal proportional to the vertical reaction force at the model center of gravity (cg) due to the mount, and (2) a signal proportional to the pitching moment reaction force at the model (cg) due to the mount. Provision is made for generating the additional error signals required when the model freedom is later extended to the lateral plane of motion and roll.

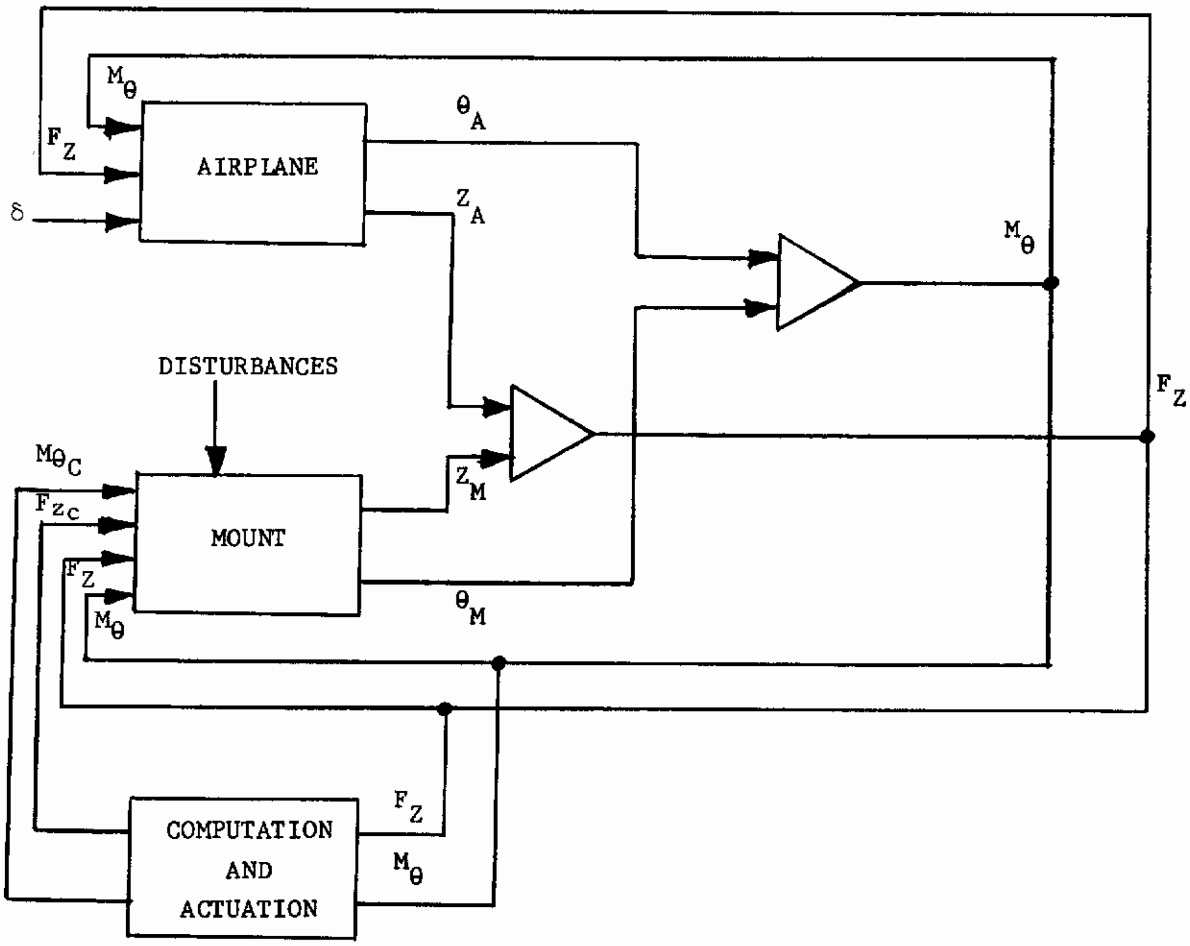

Figure 11 Block Diagram of Mount Control

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The mount instrumentation for mount and model safety consists of measurement of the appropriate parameters for computer computation and decision to automatically halt a flight test and return the model to a trim condition. Also included is a redundant set of transducers that will mechanically halt the flight test in the event the primary system malfunctions.

The primary safety system will operate as a function of the displacement, velocity, and acceleration of both vertical position, Z, and pitch, θ. A composite function of the above parameters will determine when the flight test should automatically be halted. For example, the distance required to safely brake the system and bring it to a trim condition is a function of the instantaneous velocity and the available acceleration. Thus the position of the system when the safety mode should be initiated is variable, but is easily computed in real time on the analog computer. The actual values of the parameters that should be used to determine safety mode initiation cannot be defined until the final design stages of the system and the completion of the computer simulation of the system.

The transducers for measurement of the above parameters are standard off the shelf items and pose no procurement problems. The redundant safety mode system will consist of a second set of transducers of the type manufactured by the Inertia Switch Company in New York. These devices are off-the-shelf also and combine sensing and triggering in one small package. They will operate independently of the primary safety system and will be adjusted to trigger at a higher value so that they will function only when the primary system fails. Further, a vibration sensor will also be incorporated to halt a flight test in the event of excessive vibration.

Mount instrumentation for the measurement of model flight response consists of measuring the vertical displacement, velocity, and acceleration, and pitch of the model by transducers on the mount. However, the effect of structural elasticity and performance on the mount must be determined to establish the validity of a one to one correspondence between the transducers on the mount and the actual model response. The same thought applies for vibration limiting.

7.4 MODEL INSTRUMENTATION CONCEPTS

Model instrumentation is divided into three areas:

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1. Measurement of structural and flight response.

2. Model flight control.

3. Data transmission for 1 and 2 above.

Measurement of structural and flight response consists of: (1) defining the parameter to be measured, and (2) selecting a transducer.

To define a parameter the following is considered.

1. Physical quantity to be measured.

2. Its magnitude, range, and frequency bandwidth.

3. Resolution of measurement.

4. Environment of the parameter, i.e. the temperature, humidity, air pressure, vibration, acceleration, and any other environmental quantity of significance.

5. Physical size and weight limitations imposed on a transducer.

For the most part the above items will be completely defined during the final design stages of the model and mount and advanced analog computer study of the system. The ambient conditions in the wind tunnel are contained in Ref. 16.

A cursory review of the parameters to be measured and their approximate definitions has allowed a preliminary survey of transducers that may be utilized. The most stringent requirement on transducers for structural response is physical size and weight. The requirements for flight response measurements are common and can be met by off-the-shelf transducers for acceleration, displacement, and velocity.

Model flight control is accomplished by means of position servos controlling the position of the model control surfaces. This requires an actuator to drive the control surface and a transducer to measure and feed-back the control surface position. These devices must be aboard the model; thus, size and weight limitations are imposed on them.

Miniature position transducers are available off-the-shelf and pose no problem. The size and weight limitations on the actuators is a very real problem.

To date a linear, electrical actuator has not been found in the power range desired. Rotary, electrical actuators are found to meet all specifications except size and weight. Linear, high pressure, pneumatic and hydraulic actuators appear to meet all the specifications. However, they are not off-the-shelf.

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Discussions with vendors are continuing in the search for appropriate actuators. Elevon actuators only are being considered at this time for the first part of the MODEL FLY program. However, in making provision for the rudder actuator a cursory glance at the rudder requirements indicates that a solution to the elevon actuators will also satisfy the rudder requirements.

The other aspect of model control is stability augmentation. It is desirable to be capable of "dialing in" any type or degree of stability augmentation. This is very easily achieved by closing the loop of the control surface position servos through the analog computer. Thus, any type or degree of stability augmentation may be simulated and adjusted during actual flight testing in the wind tunnel.

The third item of model instrumentation, data transmission, has been covered in Section 7.2.

7.5 CONCLUSIONS

The following conclusions are reached:

1. Transmission of data to and from the model is best accomplished by means of pulse code modulation of signals and time sharing one (or a few) hard wires.

2. Transmitted signals from the model will be stored and selected signals will be displayed for rapid on the spot analyses.

3. Mount instrumentation involves error sensing, control computation, and safety considerations. Redundant equipment will be required.

4. Model transducers and control systems present a problem because of size and weight restrictions.

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8.0 DATA PROCESSING CONSIDERATIONS

8.1 INTRODUCTION

Processing of the data will play a key part in proving the performance of the MODEL FLY technique. This section outlines the necessary considerations and the present status of the development of the MODEL FLY data processing system.

The approach used to present the data processing requirements is as follows. First, the underlying philosophy of the data system is presented. Then the data processing requirements are outlined in general. This section is then concluded with a discussion of the present status of the data system development and future work agenda.

8.2 PHILOSOPHY FOR THE DATA PROCESSING SYSTEM

The data processing system must be available for use as a tool during the test program. This means that on-site and, in some cases, on-line or real time data processing is necessary. The data system must, therefore, be capable of taking data signals from the model and with a single pass through the data system, provide pertinent test information such as stress at critical points in the structure, system frequency and decay factor, etc. This sytem capability will enable the test engineer to sample the results, examine the validity of the data, establish important trends, and allow the test engineer to direct the program more efficiently.

To develop the desired data processing system, it is necessary to integrate data handling equipment and analysis techniques which are both versatile and efficient. The analysis techniques must be established with sufficient lead time for coding and debugging computer programs. The programs should be planned to yield the pertinent information in an acceptable and readily usable form of presentation. The result of this planning and philosophy is a coordinated data system designed to produce fast answers to problems involving both analog and digital data which will enable command decisions to be made with reasonable certainty and speed.

8.3 DATA PROCESSING REQUIREMENTS

The data processing requirements are dictated by analyses typical to the problem areas being considered. These areas are: 1. control of model and mount, 2. determination of flight loads information, and 3. determining stability and control response. The particular requirements in each of these areas are discussed below.

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8.3.1 Control - Data processing for control of the model and mount must obviously be a real time, on-line computation. The mount-model system will utilize an analog computer in a closed loop control system. The controller must perform the function of initiating the maneuver through control surface deflection, see that the mount adequately follows the model motion, and insure the safety of the entire system (wind tunnel included).

8.3.2 Flight Loads - For the flight loads portion of the program it is desired to obtain total load distributions and total load magnitudes. Since each wind tunnel maneuver lasts only a short time, the loads processing will not be a real time computation. However, in line with the outlined philosophy, loads calculations will be performed directly after each maneuver, so that pertinent information is available prior to performance of the next scheduled maneuver or test.

The step by step procedure in computing the loads information is dependent upon the type of transducer used, that is, pressure transducer, accelerometer, or strain gage. However, the basic computations are, in any case, matrix operations. An additional consideration, in any case, is the need for interpolation if a transducer fails or malfunctions yielding a bad data point. A suitable interpolation and logic analysis is required to meet and solve this problem.

8.3.3 Stability and Control - The flight stability data requirements are considered for both steady state (trim) and dynamic flight response. The steady state requirements for control effectiveness, static margin, etc. are to be obtained using a straight-forward algebraic or possibly matrix operation. The data processing for the dynamic conditions will necessitate a more complex analysis.

For the dynamic conditions, a frequency discrimination technique and a method for calculating the decay or damping ratio are needed. For test demonstration of flight stability, the time to damp to 1/10th amplitude after a control pulse is of prime importance. Military specifications (Section 3.35, MIL-F-8785 ASG) require that the response of the vehicle to a control pulse damps to 1/10th amplitude in one cycle. These computations will not be real time or on line tasks, but results will be available to the test engineer before the next test point is started.

The vehicle transfer functions and Bode plots (frequency response) are parameters of interest to the control analyst. Computation

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of both the transfer functions and Bode plots are not to be performed while testing. All digital techniques for computing transfer functions utilize a large amount of storage capacity and consume considerable time. The transfer function and Bode plot computation can be deferred until conclusion of testing because they do not contribute any information which can re-direct or benefit the test procedure.

A further data processing requirement for both the flight loads and stability data, is a suitable program to transform from model to prototype response. The magnitude of programming requirements in this area are dependent upon the similitude analysis, that is, whether or not quasi-similitude is to be used (see Section 2).

8.4 STATUS OF DATA PROCESSING SYSTEM DEVELOPMENT

The desired data processing system is a sophisticated combination of equipment and analytical techniques. However, no new data processing breakthroughs are required to achieve the system. The problem is one of finding the most suitable techniques and integrating the software with the appropriate hardware to yield a good data system.

Most of the required hardware for the MODEL FLY data system is available as part of the data reduction facility at AEDC. The AEDC facility does not have a transfer function computer or spectrum analyzer. As reported in Bibliography Item 7, the specialized equipment just mentioned provides a more convenient and relatively faster method for obtaining transfer functions on a non-real time basis. The analog computer to be used as the model mount controller, has been purchased and is currently being employed in the study and development of the system.

The analysis for the real time data processing, the model mount controller, cannot be finalized until the mount design has been firmed. The control loop for system safety, also, cannot be established until the mount characteristics are firmed.

The data flow diagrams for both loads and stability calculations have been drawn and can be found in Bibliography Item 7. About 40% of the digital computer routines necessary for the system are either checked out or available in a nearly checked out state. Some of the programs available are:

1. Inverse Laplace transform
2. Harmonic analysis

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3. Simpson integration
4. Lagrangian integration
5. Solution of simultaneous ordinary differential equations
6. Matrix operation package
7. Interpolation for missing or erroneous transducer data

Programs which are currently required:

1. Model to prototype similitude transformation
2. Bode Plot (frequency response) from time history
3. Least squares transfer function routine

The model to prototype transformation is not completely developed at this time (see Section 2). Since the requirements are not completely defined, coding of the routine cannot be started. The routine for required programs 2 and 3 above will be coded if not available through the SHARE library.

The prime consideration in assembling all of the programs mentioned above into the MODEL FLY data system is the compatibility of the program code with the AEDC digital computer. The computer at AEDC is a UNIVAC 1102, whereas, most of the programs in hand are programmed for the IBM 7000 series computer. An interpretive program is available at AEDC for running these IBM packages.

8.5 FUTURE DEVELOPMENTS

The problem of compatibility of the available programs with the AEDC computer is the major item on the work agenda. Next in priority is coding the similitude transformation (as data becomes available) and obtaining routines for the least square transfer function and Bode plot from the time history. Once all the necessary analysis is coded, the integration of the entire system will be started.

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9.0 CONCLUSIONS

This report presents the technical considerations and the results of preliminary analyses requisite to the development of the MODEL FLY testing technique. The pertinent conclusions are as follows:

1. Practical modeling advantages are realizable by use of the quasi-similitude approach. Further development of the method should be pursued although it is not essential to achievement of program objectives.

2. It is possible to perform all required model maneuvers in the selected wind tunnel.

3. Model construction methods and techniques exist which with modest ingenuity can provide a modeling technique adequate to meet the MODEL FLY requirements.

4. The mounting problem is technically demanding but at least two feasible approaches are available.

5. Transonic servo wings are not a feasible inertia cancelling device because of sensitivity to small aeromechanical changes. However, several other promising choices are available, most notably a combination of hydraulic actuation and cables driven by stored energy via viscous clutches.

6. At present, not all instrumentation hardware problems can be solved with off-the-shelf items. However, these items are presently in development and should be available on schedule, or off-the-shelf items can be modified.

The MODEL FLY program is proceeding toward the basic objectives, following a planned program. All technical objectives appear feasible at this time.

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The following reports have been published generally for the use of the United States Air Force and Contractor personnel directly allied with the MODEL FLY program. The reports tend to be informal in character and contain quantities of detailed information, essential to the program, but not apropos to this report. Should more detailed information be required, the reports below may be requested on loan from the Air Force Flight Dynamics Laboratory, Attention: FDTE, Wright-Patterson Air Force Base, Ohio.

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