Cleared: October 25th, 1972
Clearing Authority: Air Force Flight Dynamics Laboratory

DESIGN, FABRICATION, TESTING, AND DATA ANALYSIS OF ADAM II CONCEPT (PROPULSIVE WING)

PART I GENERAL AND SUMMARY INFORMATION

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FOREWORD

The work reported upon herein was performed by the Vought Aeronautics Division (VAD) of the LTV Aerospace Corporation of Dallas, Texas, under Contract Nr. AF33(615)-3293, Project Nr. 1366, Task Nr. 136617, supported jointly by the United States Air Force and the United States Army. Air Force support for this effort was made possible through the use of Air Force Flight Dynamics Laboratory Director's Funds. After shakedown testing by the Contractor, the major tests were conducted by the NASA Langley Research Center, Hampton, Virginia, in the 17-foot test section of the LRC 7-foot x 10-foot wind tunnel and in the LRC 16-foot transonic wind tunnel.

The actual wind tunnel testing was started on 2 December 1966 and was completed on 7 July 1967. This report was submitted by the authors in December, 1967.

Acknowledgements are due to many individuals in the Air Force, the Army, and the Langley Research Center. Particular reference is made to Mr. P. P. Antonatos of the Air Force Flight Dynamics Laboratory, who suggested the use of a single model for high and low speed testing in the Langley Research Center wind tunnels; to Messrs. F. M. Rogallo, R. E. Kuhn, A. D. Hammond, K.E. Spreeman, and Garl C. Gentry of the LRC Low Speed Vehicle Branch, who conducted the low speed testing; to Messrs. B. W. Corson, Jr., J. F. Runckel, J. Schmeer, L. B. Salters, Jr., E. M. Brummal, C. F. Whitcomb, and E. E. Lee, Jr., of the LRC 16-foot Transonic Wind Tunnels Branch, who conducted the high speed testing; and to Messrs. J. Gaurino and Carl Roberts of the LRC Instrumentation Research Department, and Mr. John Wilson of the Air Force Flight Dynamics Laboratory, who assisted in resolving problems in the internal strain gage balance used for high speed testing.

The use of photographs of the model in the Langley Research Center Wind Tunnels, furnished by LRC, is gratefully acknowledged.

The program monitor for the Air Force was Major Edward P. Miller, FDMM, Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio, assisted by Mr. Robert R. Jeffries of the same organization. The program monitor for the Army was Mr. LeRoy T. Burrows, SAVFE-TP, U. S. Army Aviation Materiel Laboratories, Fort Eustis, Virginia. The principal investigator for the Contractor was Mr. B. R. Winborn.

This report is presented in four parts as follows:

Design, Fabrication Testing, and Data Analysis of
ADAM II Concept (Propulsive Wing), Part I
General and Summary Information

Design, Fabrication Testing, and Data Analysis of ADAM II Concept (Propulsive Wing), Part II Shakedown Testing in the VAD 7-foot by 10-foot Low Speed Wind Tunnel

Design, Fabrication Testing, and Data Analysis of ADAM II Concept (Propulsive Wing), Part III Hover and Transition Mode Testing in the 17-foot Test Section of the Langley Research Center 7-foot by 10-foot Low Speed Wind Tunnel

Design, Fabrication Testing, and Data Analysis of ADAM II Concept (Propulsive Wing), Part IV Cruise Mode and High Speed Testing in the Langley Research Center 16-foot Transonic Wind Tunnel

This technical report has been reviewed and is approved.

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ABSTRACT

Design, fabrication, testing, and data analysis of an ADAM II concept (propulsive wing) V/STOL airplane model has been performed by the Vought Aeronautics Division, LTV Aerospace Corporation, Dallas, Texas, under a contract supported jointly by the United States Air Force and the United States Army. The actual testing was conducted by the personnel of the NASA Langley Research Center (LRC) in the 17-foot test section of the LRC 7-foot by 10-foot low speed wind tunnel and in the LRC 16-foot transonic wind tunnel. The results detailed in Parts II, III, and IV are generally favorable and have led to a recommendation for continued development of the ADAM II concept.

This abstract is subject to special export controls, and each transmittal to foreign governments or foreign nationals may be made only with prior approval of Air Force Flight Dynamics Laboratory (FDMM), Wright-Patterson Air Force Base, Ohio, 45433.

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SYMBOLS

Ac	Capture area (highlight area) of inlets
Ao	Area of stream tube at free stream conditions of flow captured by propulsion system
Ao/Ac	Mass Flow Ratio
AF	Axial Force, pounds
AR	Aspect Ratio
$C_D = \frac{D}{qS}$	Drag Coefficient
$C_{D_i} = \frac{D_i}{qS}$	Induced Drag Coefficient
$C_L = \frac{L}{qS}$	Lift Coefficient
$C_T = \frac{F_N}{qS}$	Net Thrust Coefficient
$C_\mu = \frac{F_G}{qS}$	Gross Thrust Coefficient
$C_\ell = \frac{RM}{qSb}$	Rolling Moment Coefficient
$C_m = \frac{PM}{qS\bar{c}}$	Pitching Moment Coefficient
C_{m_o}	C_m at α for zero lift
$C_n = \frac{YM}{qSb}$	Yawing Moment Coefficient
C_p	Specific heat at constant pressure
$C_Y = \frac{Y}{qS}$	Side Force Coefficient
D	Drag, pounds
D_i	Induced Drag, pounds
D_{RAM}	Ram Drag, pounds
F_G	Gross Thrust, pounds
F_N	Net Thrust, pounds
G.B.Ht	Height of model above moving ground plane measured to Waterline 10 on the model
J	Joule's constant, 778,184 ft lb/BTU
L	Lift, pounds
L_{NF}	Distance from c.g. to center of nose fan exhaust, as noted

SYMBOLS (continued)

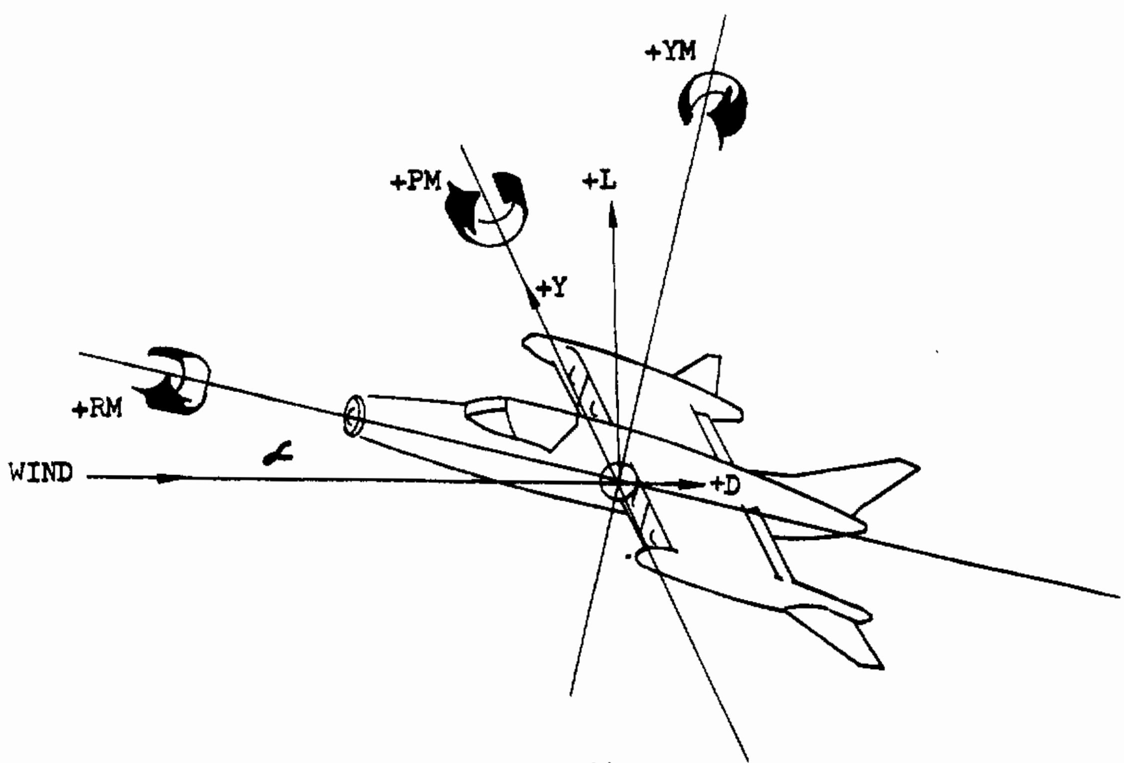
$\dot{m}_f$	Mass flow rate through fan, lbm/sec
$\dot{m}_c$	Mass flow rate of compressed air to model, lbm/sec
MGC	Mean Geometric Chord ($\bar{c}$)
NF	Normal Force, pounds
OGE	Out of ground effect
P	Pressure, psia
PM	Pitching Moment, as noted
R	Gas Constant, 53.3
RM	Rolling Moment, as noted
RNG	Denotes range or variation
S	Wing reference area (includes horizontal tail areas but excludes flap area), 7.2666 sq ft
T	Absolute Temperature, degrees Rankine
V	Velocity, ft/sec
W	Weight, pounds
WM	Windmilling fans test condition
X_W	Distance aft of wing leading edge, as noted
Y, SF	Side Force, pounds
YM	Yawing Moment, as noted
b	Wing Span, feet
b_{w+t}	Span between horizontal tail tips
$\bar{c}$	Wing mean geometric chord (excludes flap), feet
c.g.	Center of Gravity
g	Acceleration of gravity, 32.2 ft/sec^2
h	Specific Enthalpy
i_h or i_t	Horizontal tail incidence angle, degrees
i_v	Vertical tail incidence angle, degrees
p	Static pressure, psf
q	Tunnel test section dynamic pressure, psf

SYMBOLS (continued)

α	Angle of Attack, degrees
β	Angle of Sideslip, degrees
γ	Flight Path Angle, degrees or ratio of specific heats
Δ	Denotes incremental value
δ_T	Fan thrust deflection angle
δ_F	Flap deflection angle
δ_N	Nose fan thrust deflection angle
ρ	Density of ambient air, slugs/cu ft
σ	Density ratio (ρ/ρ_o)

SIGN CONVENTION

The sign convention used for the above forces and moments is shown in the following sketch. All data are referenced to a stability axes system.



SECTION I

ADAM II CONCEPT

ADAM II is a high bypass ratio V/STOL concept currently under study at the Vought Aeronautics Division (VAD), LTV Aerospace Corporation, Dallas, Texas. The keynote of the concept is a propulsive wing design which provides an effective approach to the fan packaging problem. Minimum wetted surface is achieved with a high degree of compatibility between propulsion, aerodynamic, and structural considerations.

ADAM tests conducted to date are shown in Table I. The tests of the early ADAM concept models led to a modified ADAM II concept. Application of the ADAM II concept to a strike-reconnaissance airplane design is shown in Figure 1(a). The corresponding wind tunnel model is shown in Figure 1(b).

The basic propulsion components include gas generators in the fuselage, which deliver a stream of hot compressed gas, power turbines which convert the proper amount of the power available in the hot gas into mechanical power, and fans driven by the power turbines which develop thrust by moving large flows of air. For a typical twin-engine configuration, one of the gas generators in the fuselage supplies gas to the two inboard wing fans, and the other gas generator in the fuselage supplies gas to the two outboard wing fans. Each gas generator supplies gas to one-half of the turbine which drives the pitch control fan in the nose of the airplane. The propulsion system continues to furnish a symmetrical thrust, and the airplane remains controllable following a failure of either engine in hover. Satisfactory cruise is possible using either engine alone. Safe runway landings are possible under these conditions.

Hover control is accomplished by means of a "gas power exchange" system. In this system, the amount of hot gas used by the turbines may be varied differentially from side to side for full control, and between the nose fan turbine and the wing fan turbines for pitch control. The thrust developed varies with hot gas consumption rate. Yaw control is provided by differential deflection of the doors in the wing flow vectoring devices.

In the lower speed portion of the flight spectrum, the fully augmented thrust of the ADAM II system may be continuously vectored throughout a sector from beyond vertical with a reverse thrust component to horizontal for cruise. Smooth transitions are achieved, with no step changes involved in flying all the way from a hover to maximum speed. The propulsive flows set up a jet flap effect which further enhances inflight mobility. The continuously running nose fan trims out the large nose-down pitching moment resulting from the jet flap lift augmentation. Very rapid deceleration may be obtained by harnessing the strong ram drag of the high bypass ratio fans. Very rapid reaccelerations are provided by the thrust augmentation of these fans at lower flight speeds.

TABLE I ADAM TESTS CONDUCTED TO DATE

ADAM I LOW SPEED WIND TUNNEL TEST	Model	0.15-scale model of an ADAM I light transport, ram-powered, interchangeable tail surfaces
	Facility	LTV Low Speed Wind Tunnel
	Test Dates	14-26 October 1959 and 7-11 December 1959
	Report	J. P. Young, "Low Speed Wind Tunnel Tests of an 0.15-Scale Model 'ADAM' VTOL Airplane Series I and II," Report No. AER-EOR-12775, dated 8 March 1960. Proprietary
	Key Results	1. Good lift. 2. Local stalls, which were eliminated by reconfiguration in ADAM II. 3. Encouraging results from crude outboard tail configuration, tested after centerline horizontal tail configurations exhibited low $\left(1 - \frac{d\epsilon}{d\alpha}\right)$ values.
ADAM I PROPULSIVE DUCT FLOW TEST	Model	0.25-scale model of twin fan propulsive duct
	Facility	Large plenum chamber using compressed air from LTV High Speed Wind Tunnel Air Supply System
	Test Dates	1-31 October 1959
	Report	W. E. Rice and K. D. Holliman, "ADAM-VTOL: Flow Tests of an 0.25-Scale Model of the Propulsive Duct," Report No. AER-EOR-12603 dated 15 January 1960. Proprietary
	Key Result	1. Low vectoring losses.

TABLE I ADAM TESTS CONDUCTED TO DATE (Cont)

POTENTIAL FLOW TESTS OF ADAM II DUCTED FAN AND EQUIVALENT COLLAPSED SOLID BODY	Model	(a) Representation of ADAM II ducted fan with Quermann inlet. (b) Equivalent collapsed solid body.
	Facility	LTV Research Center Rheoelectric Analog Potential Flow Facility
	Test Dates	1963
	Report	J. K. Davidson, "Pressure Coefficients on the ADAM II Shroud and the Ruden Inlet Diffuser from Tests on the Rheoelectric Analog Facility," Memo 0-71000/3A-485 dated 11 September 1963. Proprietary
	Key Result	1. Substantial differences in pressure coefficient and velocity coefficient profiles over the two models, but enough similarity for a first approximation of effective thickness of ADAM II propulsive wing
ADAM II HIGH SPEED WIND TUNNEL TEST	Model	0.03-scale model of an ADAM II light transport, ram-powered, interchangeable fan inlets and nozzle plugs
	Facility	LTV High Speed Wind Tunnel
	Test Dates	4-8 September 1963
	Report	F. L. Beissner, "Results of Initial High Speed Wind Tunnel Test on ADAM II," Report No. 2-53310/4R-50173, dated 7 February 1964. Proprietary
	Key Results	1. Flight through Mach 0.9 with very little drag divergence 2. Confirmation of outboard tail concept, with $\left(1 - \frac{d\epsilon}{d\alpha}\right) = 1.5$. 3. Good inlet performance.

TABLE I ADAM TESTS CONDUCTED TO DATE (Concluded)

ADAM II LOW SPEED WIND TUNNEL TEST OF SEMI-SPAN MODEL, Funded by Army Research Office, Durham, North Carolina	Contract	DA-31-124-ARO-D-262
	Model	0.19 scale semi-span model of recce/strike airplane with simplified fuselage, faired-over inlets, with compressed air brought aboard to simulate fan and turbine outlet flows. Interchangeable cascade boxes, adjustable flap and outboard horizontal tail.
	Test Dates	25 October - 13 November 1964
	Report	R. T. Stancil and L. J. Mertaugh, Jr., "Analysis of Low Speed Wind Tunnel Model of a High Mass Rate Vectored Propulsion Flow Model," Report No. 2-53310/4R-2166, dated 15 February 1965.
	Key Results	1. Excellent performance of outboard tails. 2. Suckdown forces small. 3. Good jet flap effect with trailing edge flap.
	Contract	DA-31-124-ARO-D-262-Mod-3
	Model	Further testing of the above model
	Test Dates	12-27 May 1965
	Report	L. J. Mertaugh, Jr., and J. K. Davidson, "Analysis of a Follow-on Low Speed Wind Tunnel Test of a High Mass Rate Vectored Propulsion Flow Model," Report No. 2-53310/5R-2206 dated 31 July 1965.
	Key Results	1. Ground effect was investigated, showing some loss in static thrust close to the ground. 2. Flow fields around the outboard tail were investigated.

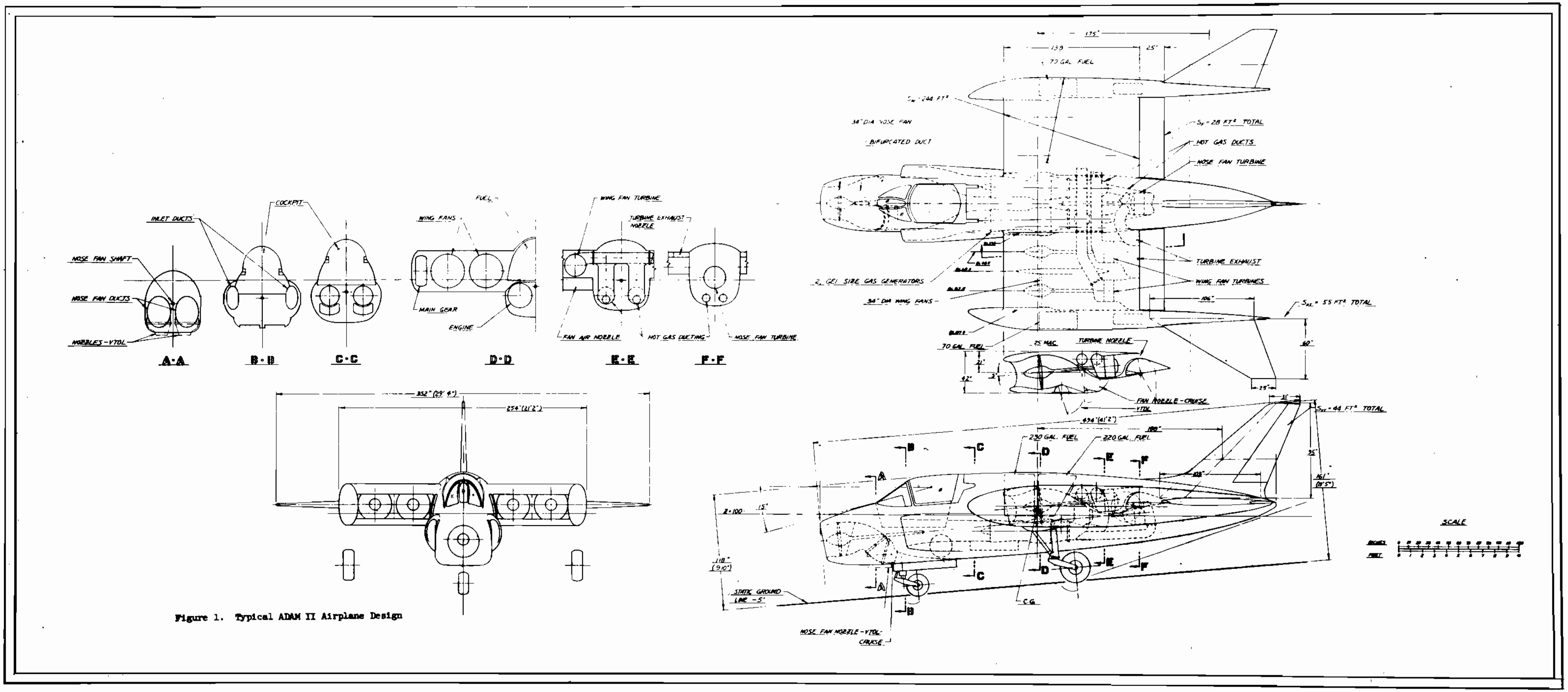


Figure 1. Typical ADAM II Airplane Design

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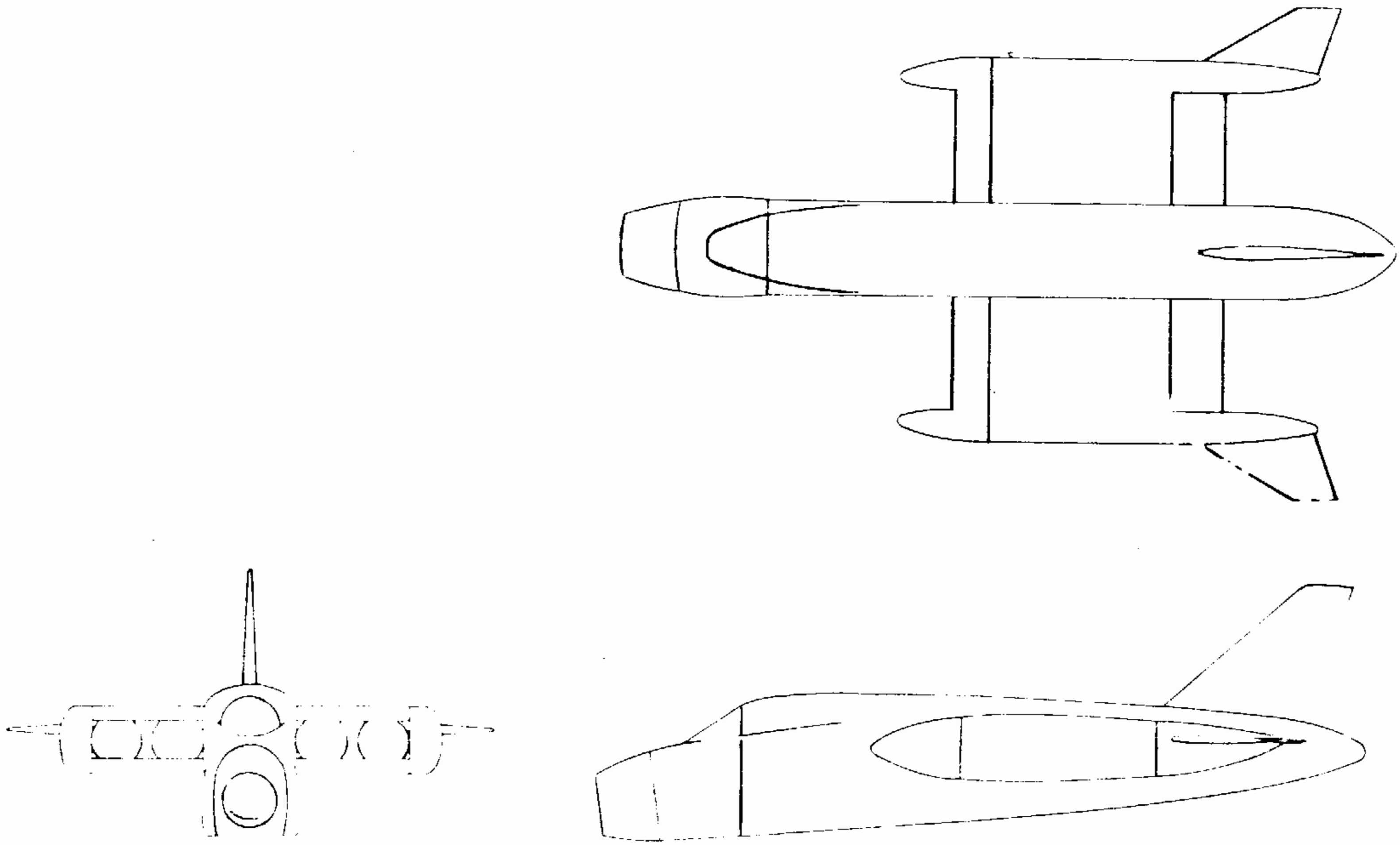


FIGURE 1(b) Typical ADAM II Airplane Design (Model Tested)

NASA research work has demonstrated the aerodynamic advantages of obtaining longitudinal stability in flight by using horizontal tails located outboard of the wing tips. Such surfaces operate effectively in the upwash portions of the wing tip vortex patterns. The theory of Reference 1 indicates that outboard tail lift supplements the wing lift so as to make the overall spanwise lift distribution approach the desired elliptical, thereby resulting in lower induced drag and good cruise efficiency.

Directional stability and control are obtained with a vertical surface mounted on the aft section of the fuselage.

The full depth of the physically thick wing is used for structural design purposes. It is obvious that the lift forces generated by outboard horizontal tails will load the wings in torsion. Structural design studies have shown that a lightweight trusswork structure in the thick wings can carry the applied loads efficiently.

Wind tunnel tests have shown that, despite its physical thickness, the propulsive wing acts aerodynamically as a thin wing with a relatively high Mach number for drag divergence. This characteristic is explained as follows. The crest of an airfoil may be defined as the high point in the upper surface relative to the direction of undisturbed flow. In a conventional wing, the boundary layer on the surface aft of the crest faces an adverse pressure gradient. This causes thickening of the boundary layer and may lead to inflections in the boundary layer velocity profiles with subsequent separations. At transonic speeds the flow over the forward portion of the wing becomes supersonic. The rear boundary of supersonic flow moves aft as Mach number increases. When the zone of supersonic velocity extends aft to the crest of the airfoil, the boundary layer over the rear portion of the wing thickens rapidly and drag increases. Shortly thereafter, a normal shock forms over the wing. This shock interacts with the boundary layer and causes a shock-induced separation with a very rapid divergence in drag. In the propulsive wing, the trailing edge of the wing proper is only slightly below the crest, with the rearward facing area of the thick wing being accounted for by the primary and secondary propulsion nozzles plus the trailing edge flap which is immersed in propulsive flows. The boundary layers on the wing surfaces face very little adverse pressure gradient. The propulsive flows from the nozzles exert an inducing effect on the subsonic boundary layers upstream of the nozzles, which strengthens them. VAD high speed wind tunnel tests of an unpowered model demonstrated capability of flight at Mach 0.9 with very little drag divergence. The tests reported on in Part IV of this report have approximately substantiated the earlier indications.

In summary, the ADAM concept calls upon the power of a VTOL propulsion system to enhance the performance of all modes of flight.

SECTION II

BACKGROUND FOR THIS TEST PROGRAM

The ADAM II V/STOL concept involves advanced technology relating to interactions between the internal propulsive flows and the external aerodynamic flows. The data needed to quantify this technology are not available in the literature, directly or by extrapolation. The ADAM tests conducted prior to the tests reported on herein, using rather primitive models, had indicated that the ADAM II concept offers unique advantages for V/STOL airplane applications. It was decided that the next logical step was to fabricate and test models affording a more realistic simulation of the freestream/propulsive flow interactions.

High bypass ratio turbofans are used for propulsion in the ADAM II concept. Test data are not reliable unless various fan parameters in the model are representative of the actual case. These parameters include duct Mach number and fan pressure ratio. A surprisingly large amount of mechanical power is required to drive the fans in a moderately sized scale model, about 190 hp being needed to drive the 5.5-inch-diameter fans in a one-sixth scale model. It was found that electric or hydraulic motors of adequate rating were much too large to fit within the contours of scaled models of turbofan airplanes.

Some of the earlier ADAM tests used models having open inlets, ducts, and exhaust jets. The internal flow was ram powered. By adjusting the outlet areas, it was possible to closely simulate inlet conditions. At high subsonic speeds, this approach yielded outlet flows with a momentum coefficient about 76 percent of the desired values. This was adequate for exploratory testing, but cannot be used to substantiate firm conclusions.

VAD had conducted two series of wind tunnel tests for the Army Research Office - Durham - using a model in which propulsion outlet flows were simulated by the release of large flows of compressed air. Because of the very large propulsive flows required, this approach dictated the use of the semispan model, reflection plane technique. Some very helpful results were obtained, but final conclusions should not be based upon tests of a semispan model having no inlet flows.

It was found that fans driven by small, high pressure, compressed air turbines could be closely representative of actual airplane fans. Turbine power could be transmitted to the fan by a shaft, or the turbine could be built into the periphery of the fan. The tip-turbine fan was lighter, less expensive, and more compact. Its characteristics were entirely adequate for aerodynamic models. The more expensive, shaft-driven fan would be more suitable for certain propulsion tests.

The turbine exhaust flows in the actual airplane are at an elevated temperature (e.g., 1200°F), and it is desirable that the simulated exhaust flows of the model be in the same temperature range. No feasible means have been found for simulating hot turbine exhaust flows in the scaled model. Electrical resistance heating was investigated. Even for a small, 0.05-scale model designed for the 4-foot by 4-foot tunnel, some 95 kw of electric power would have been required to heat the turbine exhaust to the desired temperatures. The development of such a heating system would have been a greater task than all of the rest of the wind tunnel program. NASA has successfully used the catalytic decomposition of hydrogen peroxide to generate flows simulating jet engine exhaust, and this sophisticated technique could presumably be applied to ADAM model testing. Analyses have shown that other properties of the hot exhaust gas, notably the ratio of specific heats, should have only very secondary effects. The high temperature of the hot exhaust should increase the Mach number for drag divergence because it increases sonic velocity in the hot jet. In any event, the major consideration in an ADAM model is to simulate the fan flow, because its mass rate is about eight times that of the turbine flow.

It was decided that turbine flows would not be heated for the test reported on herein and that determination of the effects of turbine exhaust temperature would be deferred until such time as full-scale models, powered by actual gas generators, are available.

Consideration was given to using two models. One model would have been powered by tip turbine fans and designed for testing in the VAD 7-foot by 10-foot low speed wind tunnel. The other model would have been designed for testing in the VAD 4-foot by 4-foot high speed wind tunnel (HSWT). The HSWT operated at high density, which improved the Reynolds number but subjected the model to very high dynamic pressures. It was found to be very difficult to obtain fans suitable for the small, extremely rugged models suitable for testing in the HSWT. Therefore, an extensive study was made of the use of a multiplicity of ejectors for propelling the "fan" flows. Analyses based upon published test results indicated fairly satisfactory characteristics. Outlet flows, however, were substantially greater than inlet flows. Considerable development was involved, and the technical risk was high.

In continuing conversations, a representative of the Air Force proposed that one model be designed, suitable for testing both at subtransition and near-sonic speeds, and that the Langley Research Center, NASA, be approached regarding actual conduct of the tests. It was found that this approach was entirely feasible. NASA was fully cooperative. It was decided that the subtransition testing should be conducted over a moving ground plane in the 17-foot section of the LRC 7-foot by 10-foot low speed wind tunnel and the higher speed testing in the LRC 16-foot transonic tunnel. In this tunnel, total pressure at the test section was equal to ambient static pressure, so that high speed testing could be accomplished at relatively modest dynamic pressures. Characteristics of the sting and support system of the 7-foot by 10-foot tunnel required, however, that the weight of the model be held to the low value of about 260 pounds. VAD had not previously been faced with a stringent weight restriction for a low speed wind tunnel model, and new methods of model construction were indicated. The model designers abandoned the familiar steel, aluminum, and mahogany designs and selected a cored magnesium casting structure with fiberglass fairings. The weight requirement was satisfied, and the new construction appeared to be no more expensive than the older type.

General agreement was reached that the model should be subjected to shakedown testing in the VAD low speed wind tunnel prior to delivery to LRC. A contract was then awarded covering the design and fabrication of the model, shakedown testing, transition mode testing in the 17-foot test section of the LRC 7-foot by 10-foot low speed wind tunnel, and cruise mode testing in the LRC 16-foot transonic wind tunnel. These three tests are reported on herein. The testing at LRC was performed by NASA personnel. Data reduction was performed by NASA. Test and support, data analysis, and preparation of the final technical report have been performed by VAD.

SECTION III

DESCRIPTION OF THE MODEL

1. AIRPLANES REPRESENTED

The model is generalized as representative of a variety of ADAM concept airplanes. It resembles the study design for the strike-reconnaissance airplane shown in Figure 1. Since the model is approximately one-sixth the size of this study airplane, it is sometimes referred to as an 0.167-scale model.

2. MODEL DESCRIPTION

The model, a photograph of which is shown as Figure 2, has four tip-turbine wing fans and one tip-turbine nose fan. An interchangeable tare nose without a fan is also provided. In accordance with the usual propulsion terminology, turbine flows (and simulated turbine flows) are designated "primary" flows, and fan flows are designated "secondary" flows. The tip-turbine fans are driven by compressed air whose exhaust mixes with the fan flows to provide the secondary efflux. The secondary efflux mass rate for the model is, therefore, greater proportionately than for the airplane. This requires an increase in fan nozzle area in the model. The effects on model geometry and performance are considered to be secondary. The primary efflux is obtained by releasing compressed air from the trailing edge of the wing over the flap. The compressed air is supplied from a wind tunnel source through the model support system to an air case assembly in the model where the air is metered and distributed to the various propulsion components. Gas generator inlets are not simulated in this model.

To simulate thrust vectoring, three sets of vane boxes, with doors, are provided for installation in each side of the lower wing surface to turn the airflow 30°, 60°, and 90°. The 60° and 90° boxes include adjustments for differential vectoring.

Four flap configurations are provided to ascertain the effect of flap shape on drag rise, flap effectiveness, primary thrust vectoring, and lateral control. These flaps are positioned by remote controlled motor-driven actuators.

Two horizontal and two vertical tail configurations are provided. The horizontal tails are of the same area but different aspect ratios. Two vertical tails of identical geometry and the horizontal tails are mounted on the booms with provision to allow for variation in tail length to determine tail effectiveness. The geometry of the second vertical tail configuration is the same as the two identical tails, except the area is equal to the sum of the tails. The large vertical tail is mounted on the centerline of the aft body shell.

Model instrumentation is provided to define the thrust of the primary and secondary airflow and to determine the pressure distribution at

Figure 2. Photograph of the ADAM II Model

the fan duct exits and externally over the wing, flaps, and fuselage. Fan rpm and fan bearing temperature sensors are provided for control and safety of the model and operating personnel.

The model includes provisions for interchangeability of three tunnel support and air systems for installation in the VAD 7-foot by 10-foot low speed wind tunnel, the 17-foot test section of the Langley Research Center 7-foot by 10-foot wind tunnel, and the Langley Research Center 16-foot transonic wind tunnel.

3. ASPECTS OF COMBINED HIGH AND LOW SPEED TEST CAPACITY

Significant advantages were realized by using essentially the same model for high and low speed testing. Data obtained from low speed testing are directly comparable to data obtained from high speed testing. The model cost was substantially less than the combined cost of separate high and low speed models. The "Hi/Lo" approach did, however, present a challenge to the model designers. It was necessary to restrict the weight of the model to a target value of 260 pounds for the low speed test. At the same time, the model had to be designed to withstand loads such as 6,000 pounds normal force and 36,00 inch-pounds pitching moment for the high speed testing. The problem was solved through the use of unconventional model construction techniques plus resorting to light and heavy versions of certain components such as model adapters, tails, and body shells.

4. DESIGN CRITERIA

The design parameters of the model were dictated by the structural and weight requirements of the Langley Research Center 16-foot transonic and 7-foot by 10-foot wind tunnels, respectively. Structural design and integrity were defined by predicted model performance and wind tunnel design criteria safety factor of two factors based on yield or of five factors based on ultimate strength. Weight restrictions were generated by balance and wind tunnel model support equipment limitations, necessitating duplicate lightweight components for the body shells, horizontal stabilizers, model adapters, and booms.

5. DESCRIPTION OF COMPONENTS OF THE MODEL

The components of the model, which are shown in an exploded view (Figure 3), are described below. VAD drawing numbers are shown for each component.

a. Wing 75-001936

To comply with the minimum weight requirement of the LRC 17-foot wind tunnel, a magnesium casting was utilized for the basic wing structure. The casting is also the primary structure of the complete model for all components; e.g., body shells, balance adapter, inlets, fans, flaps, vectoring system, booms, and spoilers are attached to it. The lower

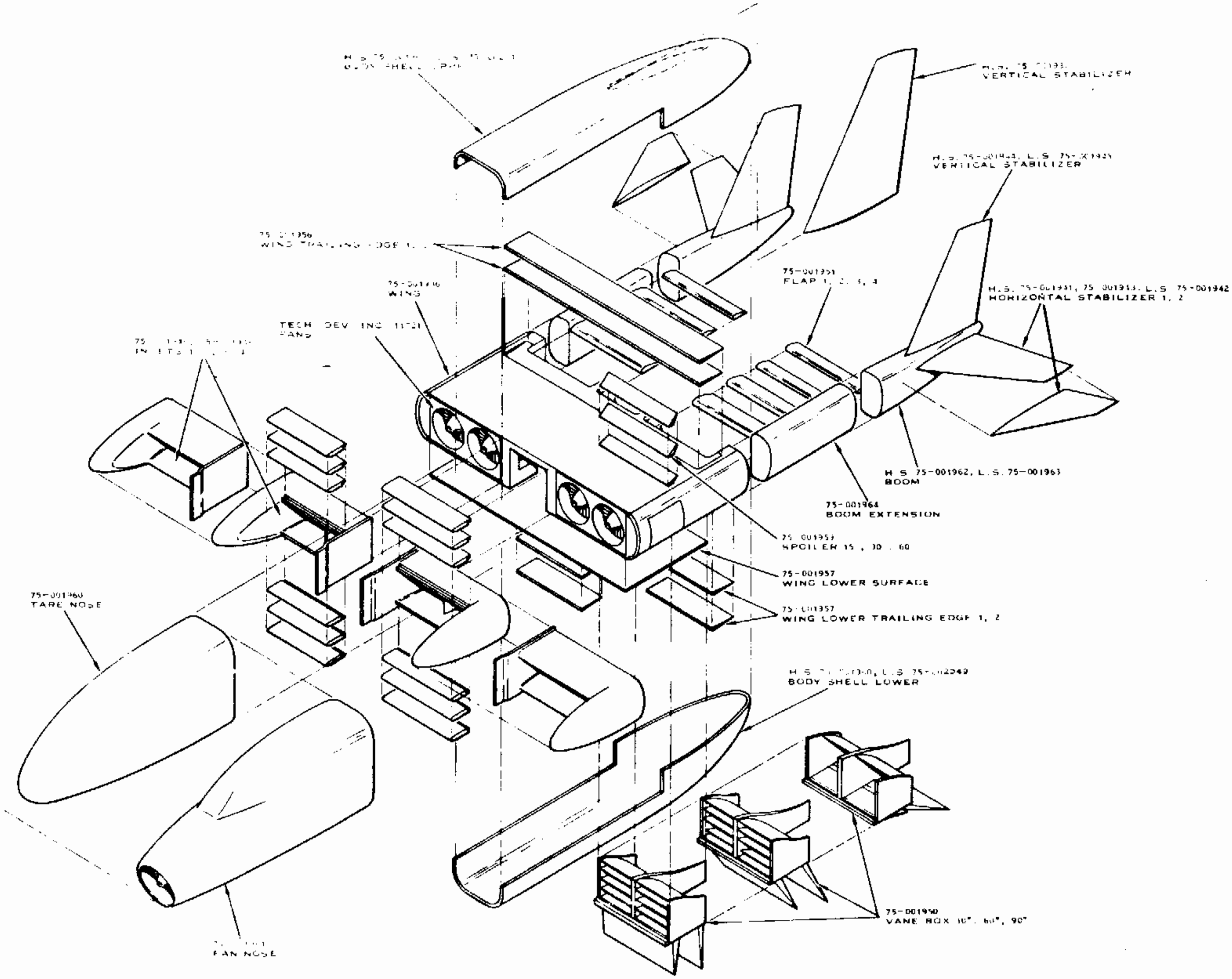


Figure 3. Exploded View of the ADAM II Model

surface of the wing consists of a removable structural aluminum panel used to simulate the external contour for the cruise and high speed configurations. A small panel attached to the casting and contained within the body is used for strength during low speed testing when the vectoring system is utilized. Fiberglass fairings with panels for access to inlet attach screws and primary air line are permanently attached to the casting as wing tip contours. The casting is machined for installation of various components including the fans, trailing edges, pressure instrumentation, and propulsion air system. A modification was added to the casting on the upper internal surface of the secondary duct to change nozzle areas and to provide means for permanently attaching instrumentation rakes and for installing pressure taps.

(1) Inlets

(a) Inlet - Basic 75-001946

Inlet assemblies with three interchangeable lips were fabricated, utilizing fiberglass shells for external and internal contours including the forward ends of the boom body sides at the inboard ends. Aluminum ribs and back plates are inserted into the shell and screwed and bonded into place. Attaching screws located at the inboard ends, splitters, and tips fasten the inlets to the casting. All instrumentation is contained within the left-hand assembly.

(b) Inlet - Extended 75-001948

These inlets are similar in construction and instrumentation to the basic inlet except the leading edge lip is an integral part of the inlet.

(2) Fan Installation 75-001954

The fans are attached to, and retained in, the wing casting by fan adapter plates. The plate assemblies, one each left hand and right hand, are 6.5 x 13.3 x 1/4-inch thick aluminum with inlet lower skin support block and duct fairings attached. The fairings extend aft of the plate and define the duct contour within the fan shroud. Duct contours and openings for the fan drive air lines are machined in the plates. Two fans with air lines attached to the back face of each adapter and the assembly is fitted into a recess machined into the forward face of the wing casting.

(3) Vectoring System - Wing 75-001950

The vectoring system consists of three right- and left-hand vane boxes with doors which direct the wing fan air 30°, 60°, and 90° downward. The vane box assemblies are used for the low speed or hovering configurations and are installed into the wing casting from the lower surface. Installation of the vane boxes necessitates removing the wing lower plate splitter and fillers located in the duct sides and installing a magnesium plate fastened across the inboard casting walls inside of the body to ensure structural integrity. One set of doors fits all vane boxes. The wing sectional configurations for 0°, 30°, 60°, and 90° of vectoring are shown in Figure 4. It is not necessary to use the same vectoring angle for the vane boxes and the trailing edge flaps as might be assumed from Figure 4.

(a) 0° Vectoring System 75-001950

The 0° vectoring system used for cruise and high speed configurations includes the aluminum lower plate with the right- and left-hand splitters. The lower plate has a chord of 10.4 inches and a span of 36 inches with the aft edge machined to accept two alternate trailing edge configurations. The trailing edge panels are magnesium plates approximately 5 inches chord by 14 inches span. Internal and external contours of the lower panel assembly simulate the wing duct and airfoils respectively.

(b) 30° Vectoring Systems 75-001950

The right- and left-hand vane boxes consist of steel side plates and vanes with magnesium splitters and doors. The doors are positioned with an aluminum contour bar which is attached to the wing casting and the lower front edge of the box. Steel brackets establish door angles. The door chords are 4.75 inches and 6.0 inches, respectively, with spans of 12.2 inches. The doors are contoured to the thickness of the wing lower plate.

(c) 60° Vectoring System 75-001950

The assembly of the 60° vane boxes consists of steel side plates and four formed vanes with a magnesium splitter installed as an integral part of the box. A forward controlled aluminum bar is provided for the installation of the box in the wing casting and attachment of the forward door. Steel brackets are provided to position all doors to the 60° configuration and differentially position the doors 50° and 70° right- and left-hand, respectively.

(d) 90° Vectoring System 75-001950

The assembly of the 90° vane boxes is like the 60° configuration except for the vane shape and door brackets which position all doors to the 90° configuration. Additional brackets are provided to differentially position the right- and left-hand door at 80°, 85°, 95°, and 100°, respectively.

(4) Flaps 75-001951

Four interchangeable flap configurations were fabricated from aluminum with the left-hand components instrumented. The basic (F_1) and two modified flaps (F_2 and F_3) have a leading radius of 1.8 inches. The chord aft of the hinge line of the F_1 and F_2 flap is 5.83 inches. The F_3 flap chord is 8.75 inches and the fourth flap (F_4) with a leading radius

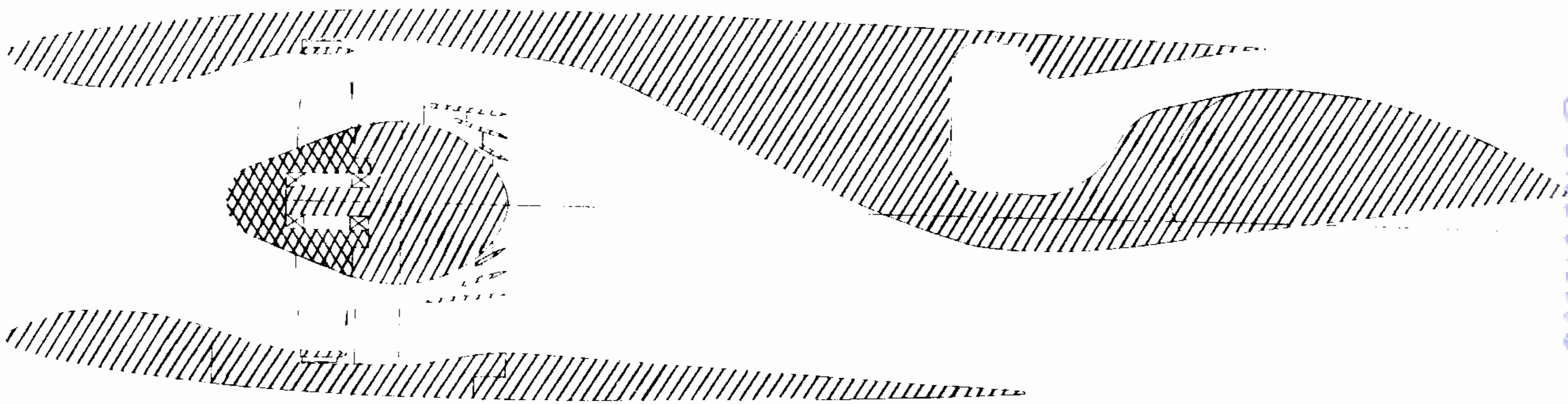


Figure 4. Wing Sections for the 0°, 30°, 60°, and 90° Vectoring Angle

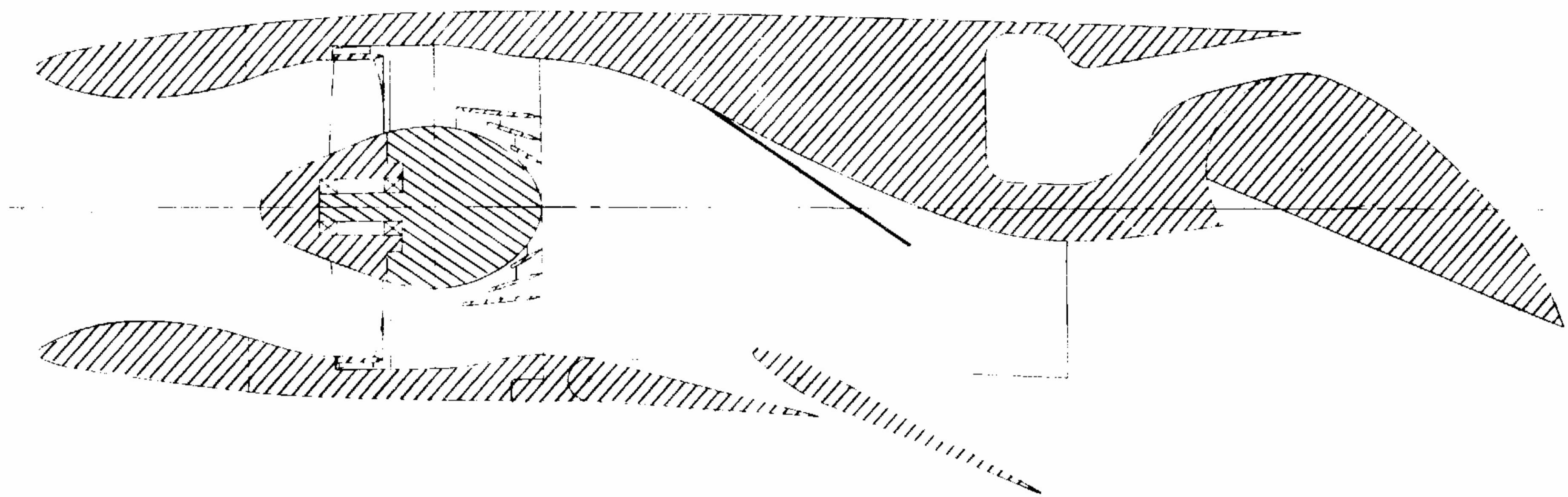


Figure 4. Wing Sections for the 0°, 30°, 60°, and 90° Vectoring Angle (continued)

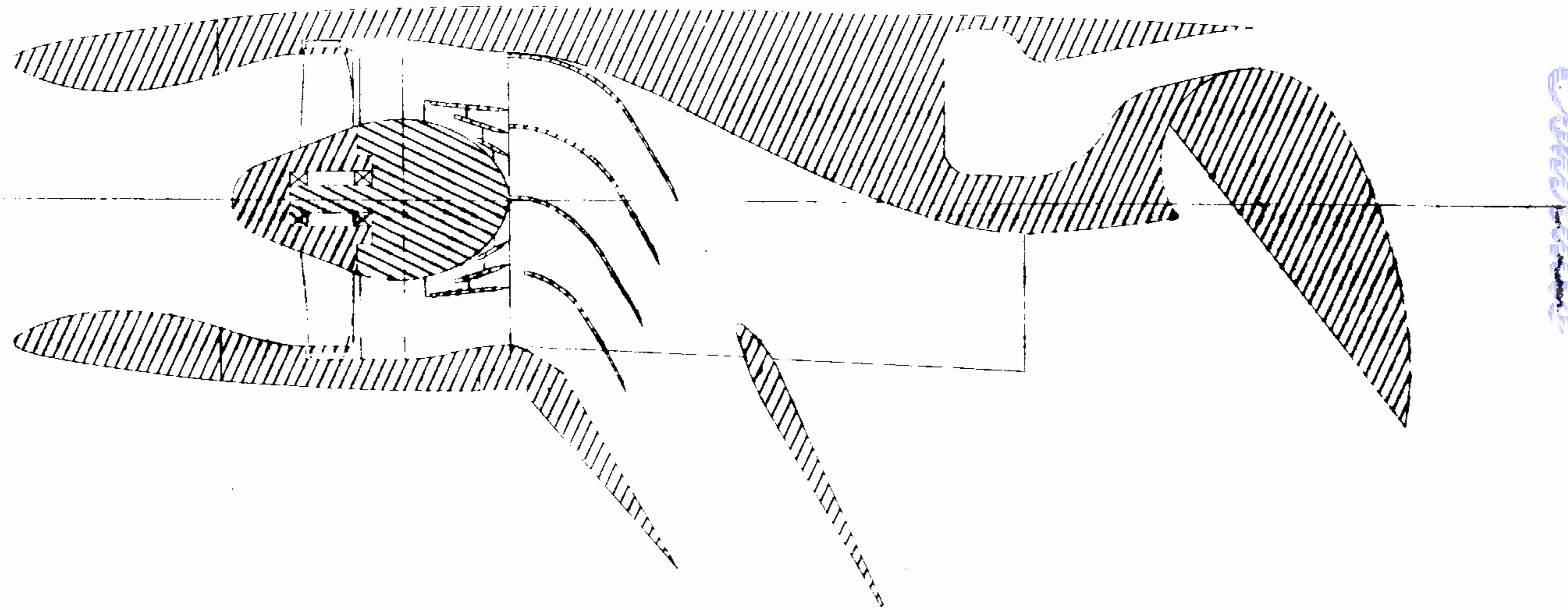


Figure 4. Wing Sections for the 0°, 30°, 60°, and 90° Vectoring Angle (continued)

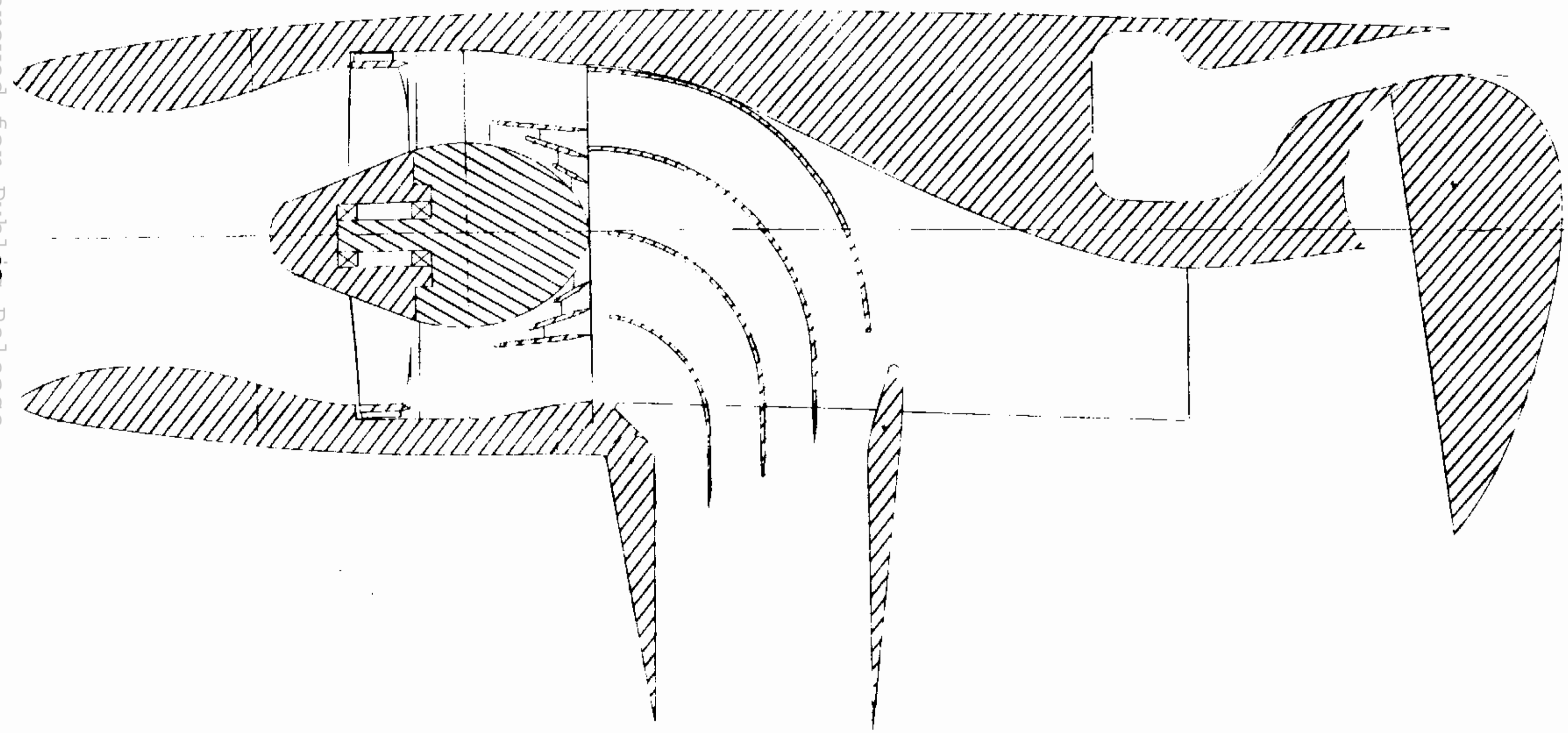


Figure 4. Wing Sections for the 0°, 30°, 60°, and 90° Vectoring Angle (concluded)

of 2.19 inches and chord of 8.75 inches is used to simulate mass flow with trailing edge T_1 configuration. The upper surfaces of F_1, F_3, and F_4 aft of the leading edge radius are circular arc contours; F_2 is flat. The contours of the four flaps are shown in Figure 5. Additional hardware is included to match the flaps to the wing casting and to permit sealed flap rotation through a 120° sector.

(a) Flap Drive 75-001952

The flap drive system consists of two remotely controlled motor-driven gear-box type actuators designed to produce 2,500 inch-pound minimum torque, to be self-locking, to operate during maximum test loadings, and to include a flap position sensing system. The actuators are attached to the wing casting and retain the inboard end of the flaps. All left-hand flap pressure instrumentation passes through the hollow actuator flap drive shaft.

(5) Trailing Edge 75-001956

Two aluminum wing trailing edges, approximately 4 inches chord and 35.6 inches span, are provided to control the airflow from the primary plenum. The external contour of the basic trailing edge (T_1) is the wing contour. The internal surface ejects the primary air parallel with the wing chord plane. The combination of the T_1 trailing edge with the small leading edge radius flaps and large leading edge radius flap simulates volumetric and mass flows, respectively. The alternate trailing edge (T_2) with a modified external and internal contour is used with the small leading edge radius flaps to simulate mass flow. The trailing edges are attached to the wing upper surface with screws.

(6) Tip 75-001955

The wing tips are structural fiberglass fairings contoured to match the wing, inlet, and boom. Two removable panels in each tip permit access to the inlet attach screws, pressure tap instrumentation, and to the air line in the primary plenum. The panels are structural members attached to the wing casting.

(7) Spoiler 75-001959

The spoiler configuration consists of three interchangeable aluminum pieces attached to the wing to provide angles of 15°, 30°, and 60° with the wing chord plane. Each spoiler chord and span is 2.164 inches and 12.200 inches, respectively. Each spoiler is attached to the upper left-hand wing surface with the hinge point at 55.3% wing chord.

b. Body

.he body consists of six fiberglass components; tare nose, fan nose, and two sets of upper and lower aft body shells. The noses are interchangeable on each of the three nose support assemblies required for

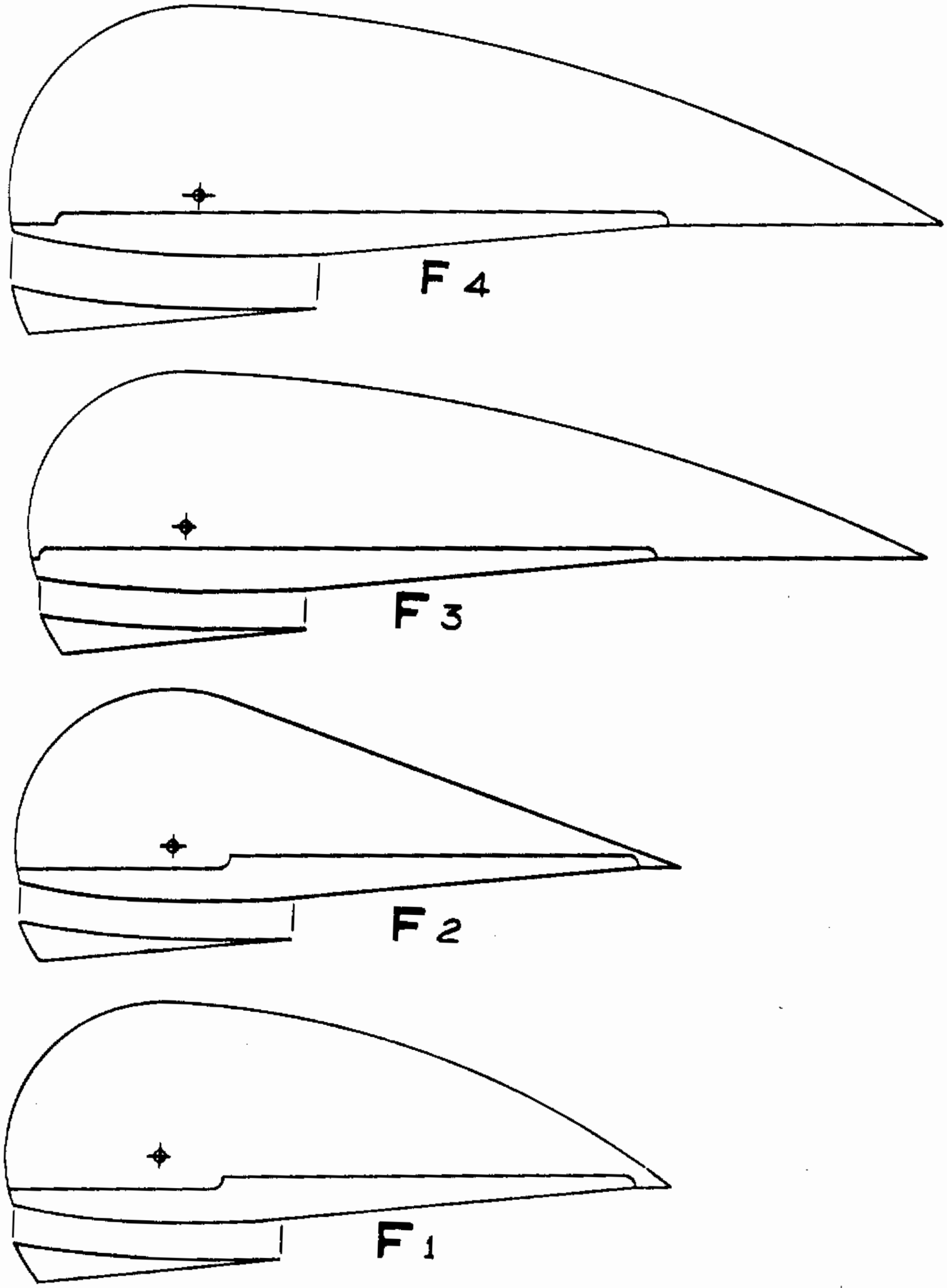


Figure 5. Flap Contours

various tunnel installations. All fiberglass assemblies are reinforced with metal frames or bulkheads as necessary to provide strength, rigidity, and attachment for other components. All metal components are screwed and bonded to the shells. The assembled body size is approximately 10.0 inches wide, 14.0 inches high, and 80 inches long.

(1) Tare Nose

The tare nose consists of a laminated fiberglass monocoque skin with three internal aluminum frames attached to the skin to provide stiffness, and an aluminum ring attached to the aft fiberglass flange to provide structure for joining the nose to the supporting structure. Longitudinal aluminum stiffeners located between the ring and frame to improve local panel strength are screwed and bonded to the skin. The nose dimensions are approximately 10 inches wide by 13 inches high by 31 inches long.

(2) Fan Nose 75-001961

The structural design of the fan nose is similar to that of the tare nose. The fan nose geometry is that of a closed cockpit with a nose inlet, fan installation, and vectoring nozzles. Removable structural side panels are included to provide access to instrumentation, air supply and control systems.

(a) Vectoring Nozzles 75-001961

Two alternate sets of fiberglass nozzles of differing outlet areas are mounted to the fiberglass transition duct. The ducts and nozzles swivel to provide 0° and 90° vectoring. In the 90° position, the nozzles may be rotated 360°.

(b) 0° Vectoring Detach Flap 75-001961

Three interchangeable aluminum flaps with chord and span of 3.00 inches and 8.60, respectively, are fitted to the lower surface of nose shell aft of the 0° nozzle position. The flaps represent a single flap deflected 15°, 30°, and 45°.

(c) Fan Installation 75-001961

A cylindrical-shaped aluminum shell approximately 6.80 inches in diameter and 3.40 inches deep with attaching flanges is used for the installation of the fan and inlet to the nose shell. Provisions for the fan air lines and instrumentation are included.

(3) Aft Body Shell 75-001960

The aft body shell consists of two fiberglass laminated assemblies. Two sets of shells were fabricated to provide a minimum weight assembly for the NASA low speed tunnel and a high strength assembly for the NASA 16-foot transonic tunnel. Sliding pin connections are attached to the shells and mating parts to facilitate rapid shell installation and removal. The aft portion of the shells are reinforced to support the centerline vertical stabilizer V_2.

(4) Nose Support

A nose support assembly was designed for each tunnel installation because of model balance adapter and tunnel support system incompatibility. The minimum weight requirement of the LRC 17-foot test section was also relevant in the design of the respective support. All supports will accommodate either the tare or fan nose assemblies.

(a) Nose Support VAD 7-Foot by 10-Foot LSWT 75-002048

The support consists of a welded steel plate assembly with a forward aluminum extension which supports the propulsion air plenum. Nose mounting provisions are included with the support, also upper and lower frames for the aft body shells. The nose support is aligned by fastening it to the model balance adapter.

(b) Nose Support LRC 17-Foot Test Section 75-002045

Welded steel tube structure was designed to provide a low weight nose support assembly. The aluminum body shell frames used on the 75-002048 assembly are utilized for the support of the upper and lower body shells.

(c) Nose Support LRC 16-Foot Transonic Wind Tunnel 75-002038

The nose support was designed as an integral part of the high speed balance adapter because of the magnitude and distribution of the nose loads and the location of the propulsion air plenum system. The support is a steel plate assembly permanently attached to the adapter. Provisions are included for the air bellows and plenum assemblies, plumbing, access to the balance attach screws, and attachment for the upper and lower body shells.

c. Booms

(1) Boom - High Speed 75-001962

One right-hand and one left-hand aluminum boom, approximately 3.5 inches by 6.0 inches by 14.0 inches were provided. They are externally contoured to match the wing casting and faired to provide supporting and seating surfaces for the horizontal and vertical tails. The vertical stabilizer is retained and positioned by one pin and one screw. The horizontal stabilizer is retained and positioned by a serrated plate and a screw located on the inboard side of each boom. An adapter is provided to support the boom on the model.

(2) Boom - Low Speed 75-001963

One right-hand and one left-hand lightweight boom is provided for the LRC 17-foot test section. The booms are shaped and supported like the high speed booms and utilize similar tail support fittings.

(3) Boom Extension 75-001964

Two aluminum extensions approximately 3.5 inches by 6.0 inches by 11.0 inches are available to provide an aft boom-tail assembly position. The extension contour is constant and matches the wing casting and boom end contours.

(4) Boom Adapter 75-001965

The steel adapters are attached to the wing casting to support the high speed boom-tail assemblies and boom extensions. The adapters are machined to match the internal contours of the booms or extensions. The latter are attached by one screw located in the lower surface. The steel adapters and a duplicate set of lightweight aluminum adapters are used in the LRC 16-foot transonic tunnel and the LRC 17-foot test section, respectively. For structural reasons, the steel adapters must be attached to the wing casting, and the aluminum adapters attached to the extensions when the model is run in the transonic tunnel.

d. Empennage

The empennage includes two horizontal and two vertical stabilizer configurations. High strength and lightweight components are provided to be used in the LRC transonic and low speed tunnels, respectively. Horizontal stabilizer incidence settings range from +10° to -40° in 10° increments. Vertical stabilizers located on the boom move 2.5°, 5.0°, and 10° outboard only. The vertical stabilizer located on body centerline is not adjustable. All empennage surfaces are contoured to the NASA 65010 airfoil section.

(1) Horizontal Stabilizer (H_1) High Aspect Ratio (High Speed) 75-001941

One right-hand and one left-hand tail surface of approximately 11.0-inch span with root and tip chords of 7.0 inches and 3.3 inches, respectively, are fabricated from aluminum. The steel supporting shaft is keyed, pinned, and bonded to the aluminum panel to transfer bending and torsion loads to the boom. Incidence angles are obtained by positioning the radially serrated shaft end in position with a matched plate attached to the boom. One screw through the inboard side of the boom on the shaft centerline locks the tail in position.

(2) Horizontal Stabilizer (H_1) High Aspect Ratio (Low Speed) 75-001942

Two fiberglass lightweight tails fabricated with the contours of the above tails, supported similarly, and positioned with a pin and screw, are provided for testing in the LRC low speed tunnel.

(3) Horizontal Stabilizer (H_2) Low Aspect Ratio (High Speed) 75-001943

One right-hand and one left-hand tail approximately 6.5-inch span with root and tip chords of 12.8 inches and 4.8 inches, respectively, are fabricated in the same manner and are interchangeable with the high speed horizontal stabilizers.

(4) Vertical Stabilizer Outboard (V_1) (High Speed) 75-001944

Two identical vertical stabilizers, approximately 10.0-inch span with root and tip chords of 11.0 inches and 3.8 inches, respectively, were fabricated in the same manner as the high strength stabilizers. To provide angular settings, a pin is installed in the fin base which matches various hole locations in the boom. The stabilizer is retained by a single screw through the bottom of the boom and into the threaded end of the fin support shaft.

(5) Vertical Stabilizer - Body Centerline (V_2) (Low Speed) 75-001934

One steel reinforced wood vertical stabilizer twice the area of V_1 was fabricated for installation on the centerline of the aft body shell. The approximate root and tip chord and span are 16.25 inches, 5.4 inches, and 14.0 inches, respectively.

(6) Vertical Stabilizer - Body Centerline (V_2) (High Speed) 75-001932

One aluminum vertical stabilizer identical in shape to the low speed wood stabilizer was fabricated for installation upon the high speed body shells for testing in the LRC 16-foot transonic tunnel.

e. Model Air Supply Systems

Variations of model installation in three test facilities necessitated separate designs for the model air supply systems in each tunnel. The air from each system passed into a plenum chamber or air case assembly contained within the model and common to all supply systems. Design drive air pressure in the air case assembly was 600 psig at a flow of 10 lb/sec.

(1) VAD 7-Foot by 10-Foot Low Speed Wind Tunnel 75-002046

The model was strut-mounted for installation in the VAD LSWT, with the air supply being brought aboard the model through the front strut. There was no room within the underbelly of the model for the large compressed air supply tube. Hence, the model was mounted in the inverted position in the wind tunnel. The air passed through a pinned ball-and-socket joint as it entered the model. An O-ring sealed the ball-and-socket joint. Pitching the model resulted in pivoting the ball-and-socket joint. A tube conveyed the compressed air from the ball-and-socket joint to the model air case assembly.

(2) LRC 17-Foot Test Section 75-002044

The air supply system to the model is an integral part of the sting assembly furnished by the tunnel and extends within the model to the balance adapter. A short pipe attached to the adapter directs the air to the model air case assembly.

(3) LRC 16-Foot Transonic Wind Tunnel 75-002037

The design of the model air supply system was established by the model test envelope and tunnel sting support system which required 180° rotation about the model centerline and 26° maximum bend at the sting joint. The design of the upstream end of the air system includes a steel fixture with a sealed joint to permit rotation of the sting air line, a 750 psi rupture disc for model protection, and provisions for six flexible lines from the tunnel air source. The thin wall steel air line provides passage through tunnel sting mount and attaches to the sting adapter. Four wedge-shaped steel blocks with seals and air passages are provided to insert into the sting joint in its various settings to direct air through the sting and balance to the model air case assembly.

f. Model Air Control and Metering System

The model propulsion air delivered through the support system is fed to the plenum chamber located within the nose assembly and attached to the balance adapter. The chamber contains pressure and temperature instrumentation, remote control valves for the wing fans and primary air system. The valves are of the motor driven, jack shaft mounted, translating plug type.

(1) Nose Fan Drive Air System 75-002051

The nose fan drive air system is contained in the nose assembly and consists of a remote controlled valve and orifice with pressure instrumentation. A single air line with ports is inserted through sealed openings in the forward and aft face of the plenum and is free to move with the nose to reduce stresses within the system caused by pressure and temperature effect and structural deflection of the nose assembly.

(2) Wing Fan Drive Air System 75-002043

The fan drive air system consists of a remote controlled valve secondary plenum chamber with four trim valves located at the forward and aft plenum faces, respectively. Two air lines are routed from each trim valve through the body and lower inlet lip to each fan. The air is metered by the main valve into the small plenum chamber and is distributed to the wing fans by the manually operated trim valves. The trim valves are of the rotating barrel type with screw drive and spring return.

(3) Primary Drive Air System 75-002052

The control valve and primary air lines are located on the upper forward and aft faces of the plenum chamber, respectively. The system consists of two lines passing over the wing and through the casting into the primary plenum chambers cast into the wing. The air is ejected into the plenum through special perforated tubes tailored to provide approximately uniform flow at the exits. Orifices with pressure instrumentation and manually adjusted trim valves are located in each line and at the primary plenum entrance, respectively.

6. MODEL SUPPORT SYSTEMS

Three model support systems were designed because of incompatibilities of the mountings, air facilities, and test requirements in each tunnel.

a. VAD 7-Foot by 10-Foot LSWT Installation

A tandem support system was used for installation of the inverted model. The forward strut enters the tip of the body in front of the wing casting and is attached to the adapter by a ball-socket joint. The ball is an integral part of the strut and is pinned to restrain the model about the roll and yaw axes. The model air supply passes through the strut and joint to the plenum chamber. The hollow pitch strut is attached to the model at the aft end of the body and is used for instrumentation routing. Figure 6 shows the model in the VAD low speed wind tunnel.

b. LRC 17-Foot Test Section Installation 75-002060 Sheet 2

The model support system including balance and sting was furnished by the wind tunnel. The model was mounted upright and attached to the balance by an adapter which includes provisions for the sting air system. The sting support controlled vertical, pitch and yaw movement of the model. Figure 7 shows the model in the LRC 17-foot test section.

c. LRC 16-Foot Transonic Wind Tunnel Installation 75-002037 Sheet 4

The model support system includes a six-component balance and sting with integral air ducts, four each sting angle setting and air line adapter blocks, and an adapter to fit the sting to the tunnel support system. The balance air bellows assembly extends into a ported shroud attached to the balance adapter. The sealed plenum chamber slips over the shroud and with limited freedom can translate or rotate to prevent loading the balance system. The sting, interchangeable angle setting blocks, and adapter are assembled with three expanding bolts. The air line adapter block applicable to the sting angle is inserted within the sting joint prior to assembly. The remaining air system components extend aft of the sting adapter into the tunnel support system. Model positioning for yaw runs is accomplished by rotating sting assembly with model 90°. The model instrumentation passed through the body along the sting and into the sting support system. Figure 8 shows the model in the LRC 16-foot transonic wind tunnel.

7. MODEL INSTRUMENTATION

Instrumentation of the model is provided to determine the thrust of the primary and secondary air flow and the pressure distribution at the fan duct exits over the wing and fuselage. Permanent rakes are installed at the primary and secondary nozzles of each fan to provide total temperature and total pressures for each fan. Pressure tap locations are shown in Figure 9.

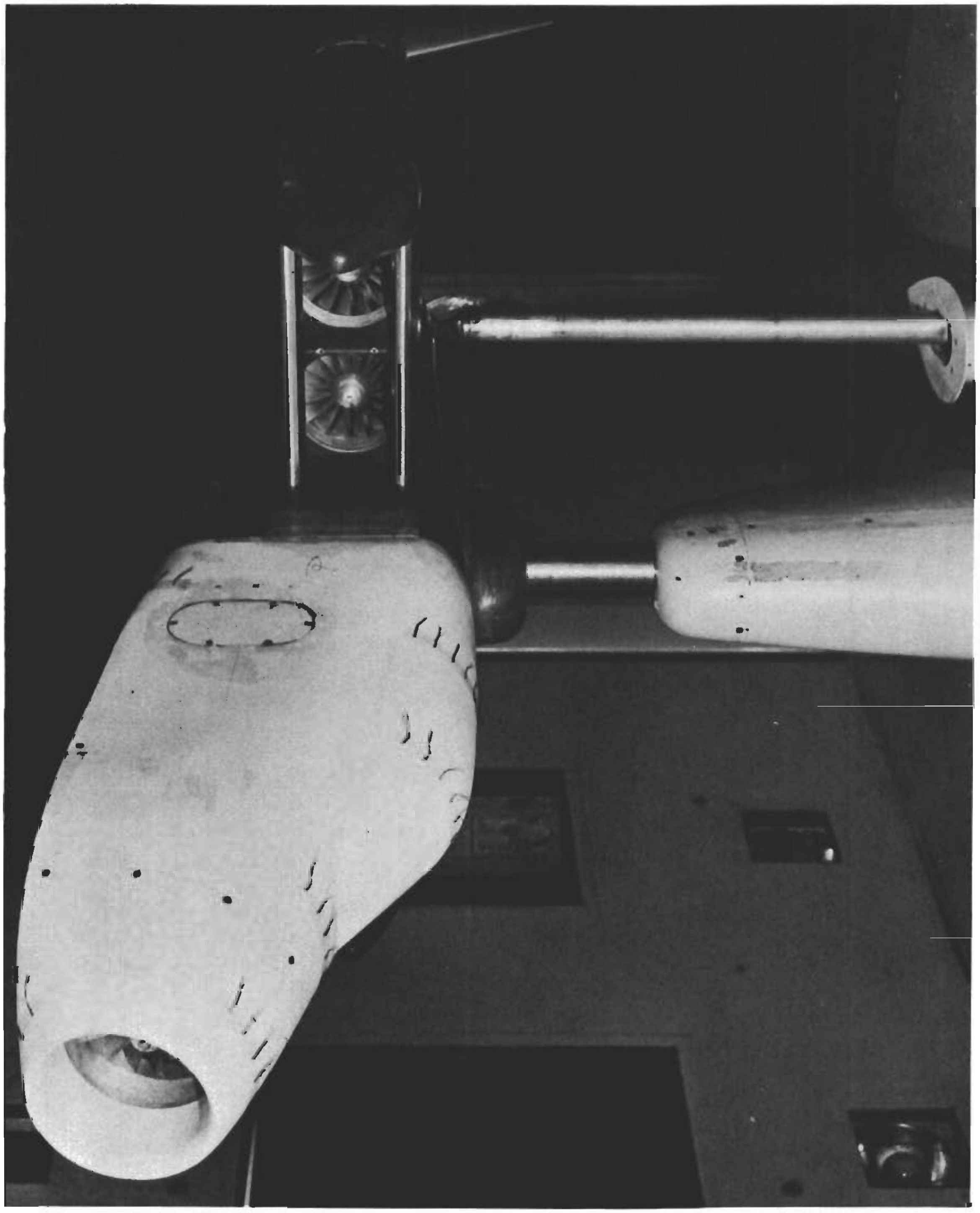


Figure 6. Model in VAD Low Speed Wind Tunnel (Sheet 1)

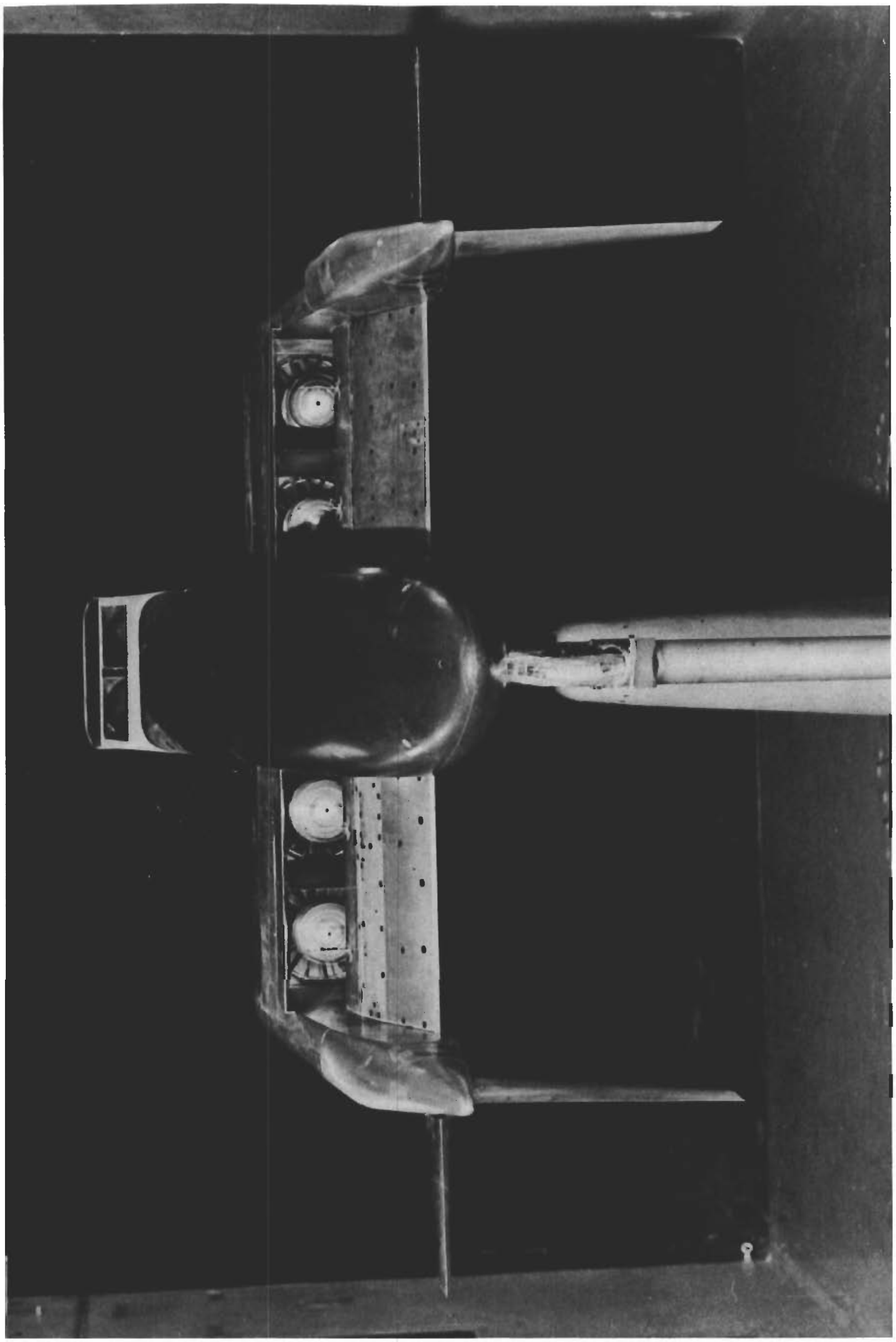
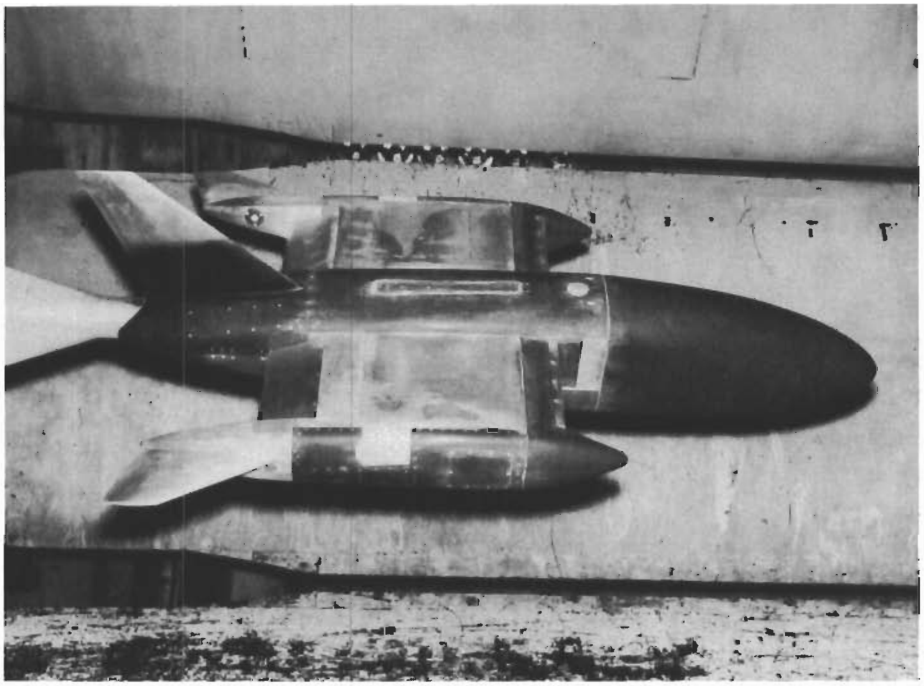


Figure 6. Model in VAD Low Speed Wind Tunnel (Sheet 2)

Figure 7. Model in LRC 17-foot Test Section



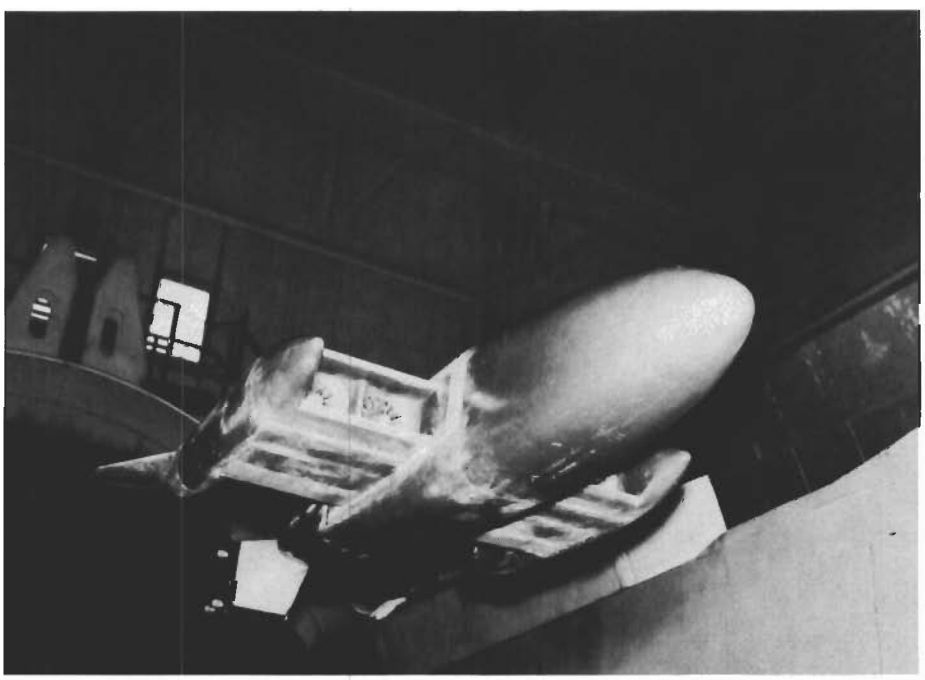
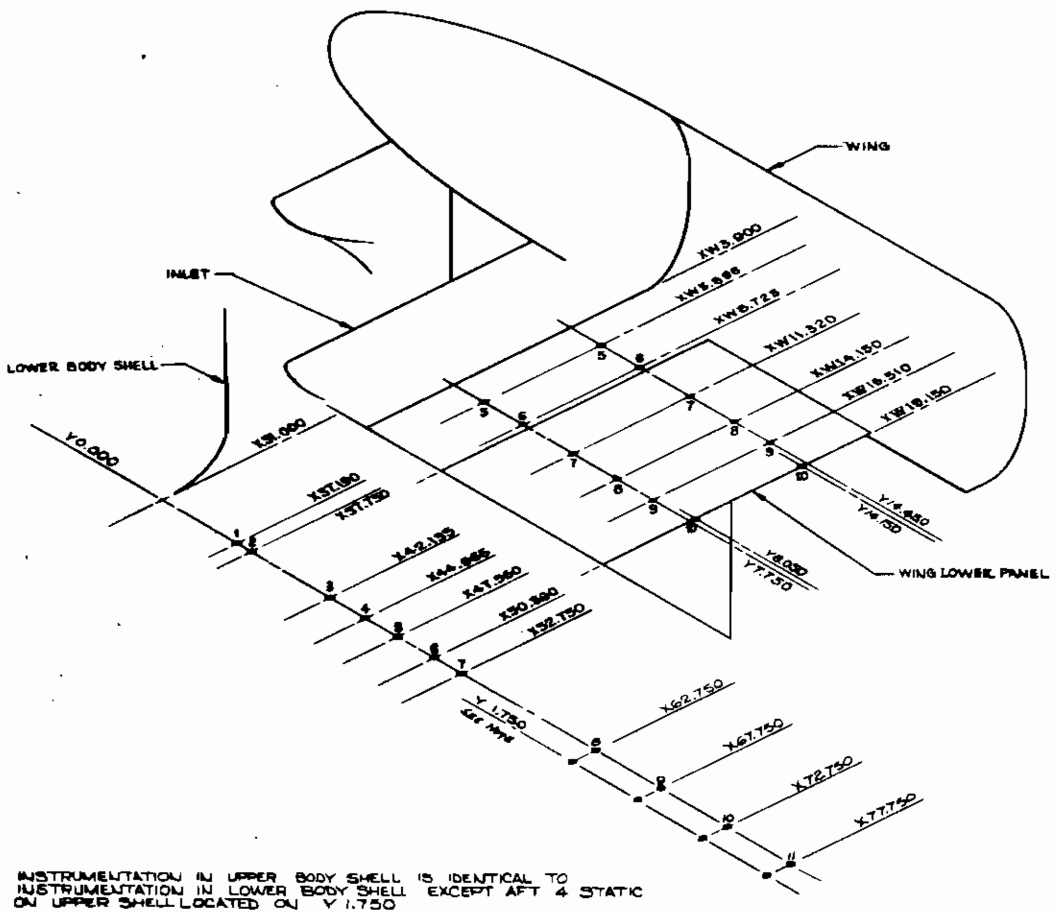


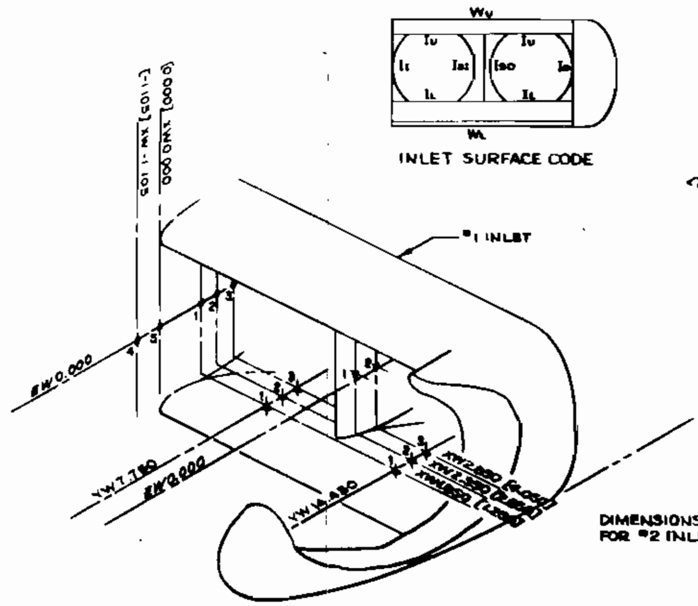
Figure 8. Model in LRC 16-foot Transonic Wind Tunnel



INSTRUMENTATION IN UPPER BODY SHELL IS IDENTICAL TO INSTRUMENTATION IN LOWER BODY SHELL EXCEPT AFT 4 STATIC ON UPPER SHELL LOCATED ON Y 1.750

REFER TO PRINT NO'S
75-001953 WING INSTRUMENTION
75-001957 WING LOWER PANEL
75-001960 BODY SHELL ASSY

Figure 9. Model Pressure Tap Locations (Sheet 1)

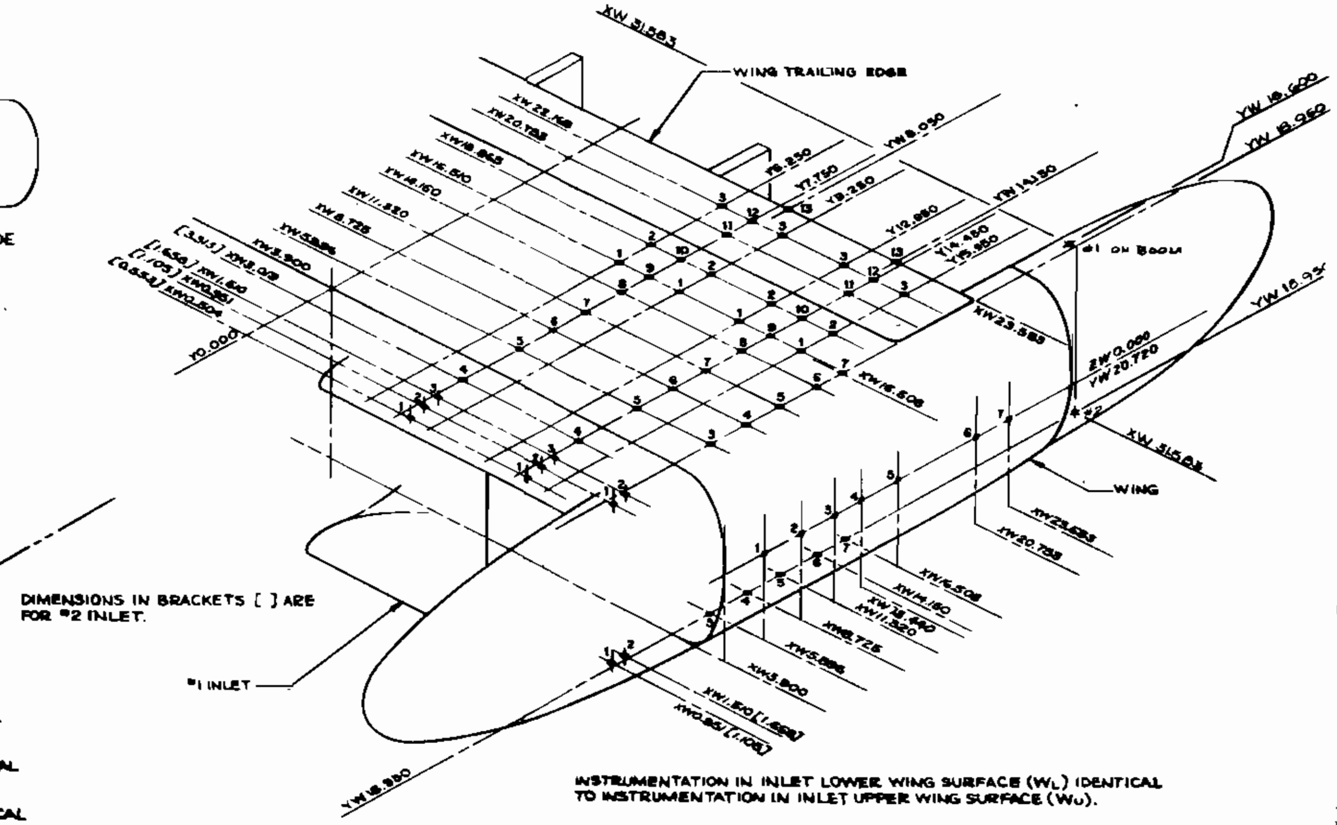


INSTRUMENTATION IN UPPER INLET SURFACE (I_U) IS IDENTICAL TO INSTRUMENTATION IN LOWER INLET SURFACE (I_L).

INSTRUMENTATION IN INBRD. SPLITTER SURFACE (I_{SI}) IDENTICAL TO INSTRUMENTATION IN OUTBRD. SPLITTER SURFACE (I_{SO}).

INSTRUMENTATION IN INBRD. BOOM NOSE SURFACE (I_B) IDENTICAL TO INSTRUMENTATION IN BODY SIDE (I_S).

REFER TO PRINT NO.
75-001947, INLETS #1 & #2 INSTR.



INSTRUMENTATION IN INLET LOWER WING SURFACE (W_L) IDENTICAL TO INSTRUMENTATION IN INLET UPPER WING SURFACE (W_U).

REFER TO FOLLOWING PRINT NO'S FOR FURTHER INFO.
75-001949, WING TIP INSTR.
75-001953, WING INSTR.
75-001956, WING T.E. UPPER DETAIL & INST.
75-001947, INLETS #1 & #2 INSTR.
75-001962 BOOM DETAIL

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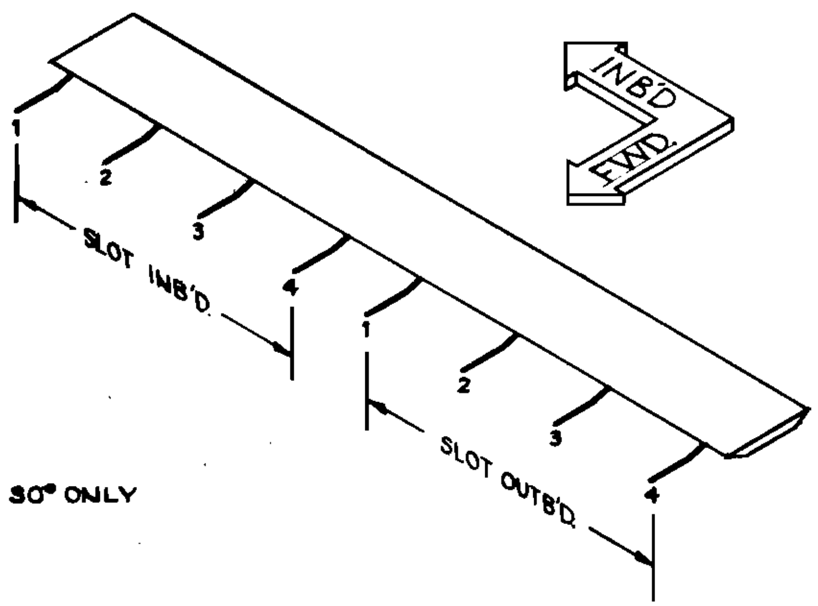
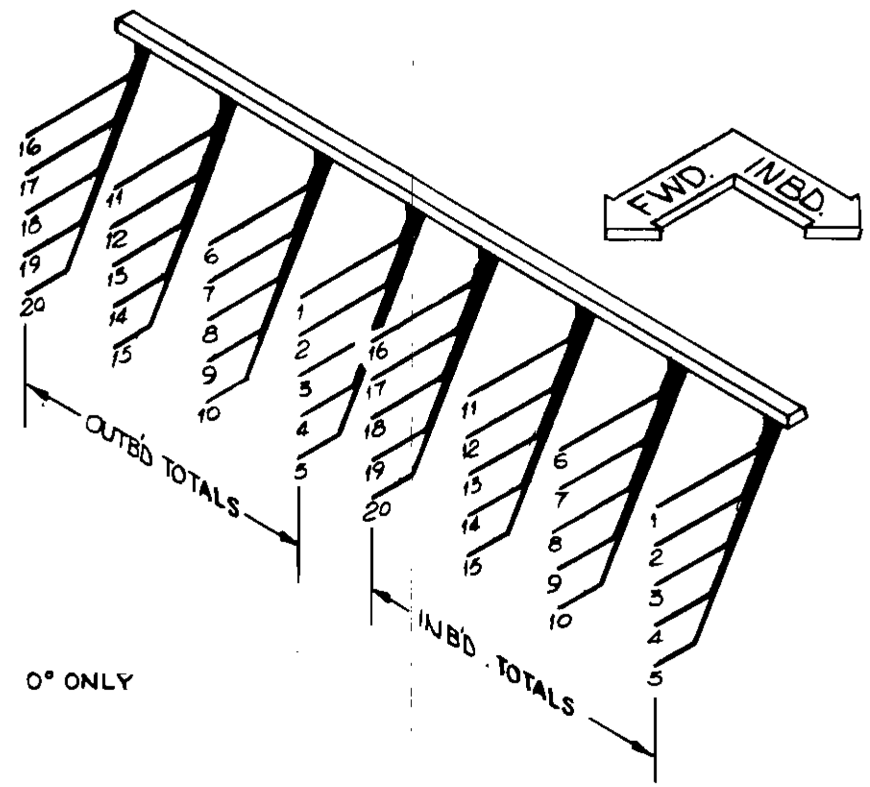
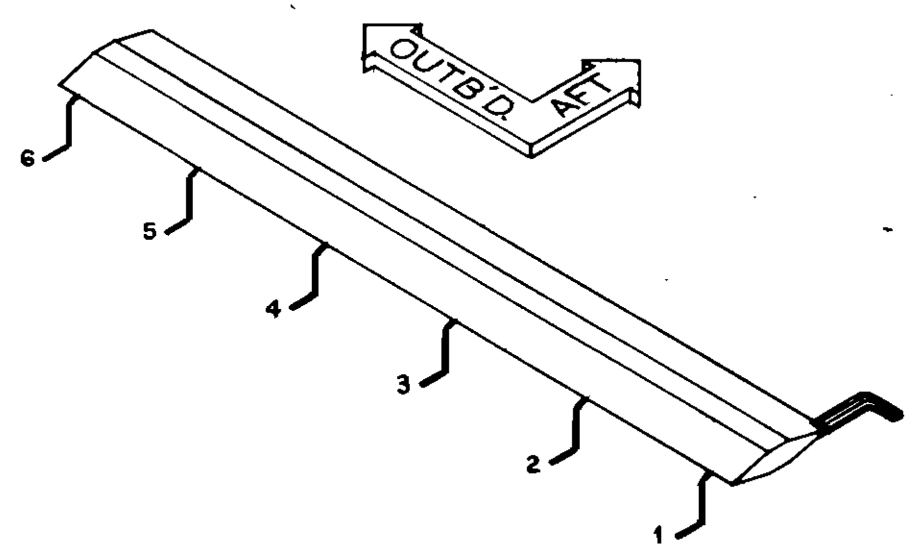


75-002053-7 VANE BOX SECONDARY RAKE



75-002053-1 SECONDARY CALIBRATION RAKE



75-002054-2 PRIMARY NOZZLE CALIBRATION RAKE

Figure 9. Model Pressure Tap Locations (Sheet 2)

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a. Primary Air Flow Instrumentation

Primary air flow instrumentation consists of orifice plates, with applicable transducers, installed in each of the left-hand and right-hand sides of the model at the primary exit.

b. Secondary Air Flow Instrumentation

Secondary air flow instrumentation includes thermocouples in the left-hand inlet splitter and in each permanent secondary rake. Pressure measuring instrumentation includes static taps in the upper surface of the duct and a permanent total pressure rake in each exit. Calibration rakes are provided for calibration of the permanent rakes.

c. Fan Instrumentation

A magnetic pickup is installed in each fan to monitor fan rpm. Two thermocouples are installed in each fan to provide monitoring of the forward and aft fan bearing temperatures.

d. Pressure Tap Installation

Pressure tap instrumentation is provided on the left-hand side of the model only. Pressure taps are located in the upper body shell, lower body shell, nose, inlets, wing, boom, and flaps.

e. Primary Rakes

Permanent rakes are provided in both left-hand and right-hand primary air exits. A calibration rake is provided for the right-hand side only.

(1) Primary Permanent Rake 75-001956

The permanent primary rake consists of three 0.042-inch-diameter tubes installed in the wing trailing edge left hand and right hand. The three tubes are then manifolded together to provide one average pressure readout on instrumentation.

(2) Primary Calibration Rake 75-002054

The primary calibration rake consists of six total pressure tubes spanning the exit area and is used to calibrate the permanent rake. The calibration rake is attached to the boom and to the body and may be raised or lowered as required to satisfy various flap configurations.

f. Secondary Rakes

Permanent rakes are installed for the nose fan and the four wing fan positions. Calibration rakes are provided for each permanent rake.

(1) Wing Secondary Permanent Rake 75-001951

A permanent rake is installed in the fan exit area. The rake consists of three, each, total pressure tubes and thermocouples, alternately spaced. The thermocouples are wired in parallel and the total pressures are manifolded to provide a single temperature and pressure readout from each fan exit area.

(2) Wing Secondary Calibration Rake 75-002053

The wing secondary calibration rake consists of a 40-tube rake which surveys either the left-hand fan exit areas or the right-hand fan exit areas. Brackets are provided to allow positioning of the rake for survey of the exit area with either 30°, 60° or 90° vane boxes in position. A separate rake is provided for cruise condition calibration.

(3) Nose Secondary Permanent Rake 75-001961

A nose secondary permanent rake is located in both right-hand and left-hand nose fan ducts. The rake consists of three, each, total pressure tubes and thermocouples, alternately spaced 60° apart on a 2.48-inch-diameter circle. Thermocouples are wired in parallel and total pressure tubes are manifolded together to provide single temperature and pressure leads to instrumentation.

(4) Nose Secondary Calibration Rake 75-002053

A 32-tube rake is located at the exit area of the nose fan ducts. The rake is attached to the sides of the duct and can rotate 180° with the duct.

g. Flap Coanda Rake

The Coanda rake is a 10-tube rake which is attached to the left-hand botton flap plate. The rake is located on the flap centerline at the trailing edge and perpendicular to the flap upper surface. The tubes are spaced 1.00 inch between centers. The Coanda rake is interchangeable with all four flaps. The pressure tubing is connected to a disconnect inside the flap, taken through the flap actuator to the fuselage, and then run to the Scanivalve instrumentation.

8. MODEL PROPULSION SYSTEM

a. Propulsion System Description

b. Wing Fan Inlet Design

(1) Low Speed Test

The basic wing fan inlet serving two fans was designed for low pressure loss at high mass flow rates. The inlet length, which is equal to one fan diameter, was established from the chord requirements of the propulsive wing. A NASA Series 1 cowl external profile was chosen because of its minimum shape drag characteristics at moderate to high subsonic speeds. The NASA 28E internal profile was chosen because of its high pressure recovery characteristics at high mass flow rates. Transition of the internal geometry from rectangular to circular was begun downstream of the throat. The cross-sectional area decreased slightly with no abrupt area changes.

(2) High Speed Test

The basic inlet, used in the low speed test, was also designed to minimize additive drag at low mass flow ratios. Three variations of this inlet were designed for use in the high speed tests for the purpose of determining lip suction effects. The three variations were (1) upper and lower lip radius of one-half the basic lip radius, (2) zero (sharp edge) upper and lower lip radius, and (3) zero lower lip radius and one-half basic upper lip radius. A fifth inlet which had a length-to-diameter ratio of 1.38 and the same basic lip radius and internal and external profiles was designed. This inlet, having a smaller throat area, was designed for less spillage than the basic inlet at a given flight condition and helped to determine the effect of spillage on external aerodynamic performance.

c. Nose Fan Inlet Design

One inlet was designed for use on the nose fan. It also used the NASA Series 1 cowl external profile and the NASA 28E internal profile and had a length equal to three-quarters of the fan diameter.

d. Wing Fan Exhaust Ducts

(1) Low Speed Design

The fan flow from the wing fan exhaust ducts was separated by a splitter which goes through transition from the fan stator exit station to the wing lower surface trailing edge station and therefore forms one side of each fan duct. This transition of each fan exhaust duct from circular to rectangular begins immediately behind the fan stator and was completed in the model in approximately two inches of axial length. This short transition was required so that the thrust vector for 90° flow deflection could be located in the proper place with respect to airplane center of gravity. The duct doors form the bottom surface of the fan duct for cruise configuration and assists flow turning in all vectoring configurations. Duct cross-sectional area smoothly increased through an equivalent conical diffusion angle of about 24° during transition from circular to rectangular. (The high angle of diffusion was necessary for the model geometry involved. More normal diffusion angles would be used for internal aerodynamics models or an actual airplane.) This diffusion produced a desirably low entrance velocity to accommodate large angle flow deflections. Flow separation from the walls of this diffuser was eliminated or minimized by fan-induced turbulence and exit swirl. Duct cross-sectional area downstream of the transition gradually converged to the required model exhaust area of 18.6 square inches.

Vectoring System. – The fan duct lower surface was formed by the lower wing surface in the cruise configuration. For flow deflections of 30° and greater, the aft portion of the model lower wing surface and the splitter were removed and a vane box with the desired turning angle was inserted. The 60° and 90° vane boxes contained a new splitter which not only divided the two fan flows but also supported the turning vanes which span the fan duct. These vanes were used to provide a more nearly optimum height-to-width ratio for the turn. In addition to the turning vanes, two doors which hinge along a line contained in the plane of the wing lower surface helped to guide the flow. These doors, when closed, formed the aft lower surface of the fan duct in the airplane. Each channel formed by the vanes had the same height-to-width ratio at the entrance to the bend and the ratio gradually converged to the exit plane giving a total exit area of 18.6 square inches. The 30° vane box used no turning vanes because it was felt that, since the turning angle is small, the doors would accomplish efficient flow turning.

(2) High Speed Design

The same design philosophy was used for the high speed configuration. However, fan operating characteristics require a 16.0-square-inch exit area for the high speed cruise tests. This reduction in area was accomplished by changing the plate forming the wing lower trailing edge.

e. Nose Fan Exhaust Ducts

(1) Low Speed Design

The nose fan exhaust duct was bifurcated to minimize the fuselage suckdown problem at hover and transition speeds. The fan afterbody and duct splitter form an aerodynamic means of dividing the flow. Transition from circular cross section behind the fan into two elliptical ducts was accomplished in a smooth manner while diffusing the flow as was done in the wing fan ducts. After transition, the two elliptical ducts decrease in area slightly to reduce turning losses. The total exit area of the two ducts was the same as that of the wing fan ducts, 18.6 square inches.

(2) High Speed Design

There was no change in the nose fan exhaust design philosophy for the high speed tests, but the exit area was reduced to 16.0 square inches as was done in the wing fan ducts.

(3) Vectoring System

The bifurcated duct described in Paragraph 8.e(1) incorporates two swivel joints to facilitate thrust vectoring. Near the end of each of the bifurcated ducts, they transitioned to a circular cross section while continuing the downward bend. At the end of this transition, a 40° segment of a toroid with a mean radius of about 3.0 inches was added, followed by the exhaust nozzle. This nozzle-toroid system had the capability to rotate in a plane containing the joint of the segment and the main duct and/or in a plane containing the joint of the segment and the nozzle. With this arrangement, a large complement of flow angles was obtained.

f. Primary Exhaust System

(1) System Design

The fan turbine drive flows in the airplane, which exhaust on the flap upper surface, were simulated in the model by dumping air into a plenum which extends from the fuselage to the inboard side of the tail boom. Air was supplied to the plenum through two rows of choked holes distributed over the length of a tube running the full length of the plenum. Either mass or volumetric flow could be simulated by changing the primary exhaust area. This area change was accomplished by changing the plate which formed the wing upper trailing edge and the nozzle upper wall.

(2) Vectoring Methods

Turning of the primary jet stream was accomplished by the Coanda effect. The Coanda surface was designed into the wing flap.

g. Air Supply System to the Model

(1) Low Speed

(a) VAD Low Speed Wind Tunnel

Air was supplied to the VAD low speed tunnel at 600 psi where it was regulated and delivered to the model at 450 psi. Air flowed up through a stationary pipe in the tunnel floor, made a turn through a knuckle joint, and dumped into the model plenum. Flow tares in this system are considered negligible since the Mach number in the delivery system is about 0.2.

(b) NASA Langley 7-Foot by 10-Foot Wind Tunnel

Air was supplied at an initial pressure of 5,000 psi and subsequently reduced to approximately 500 psi by two reducers. Air entered the sting through a large housing on the upper end of the sting at the pitch strut and flowed through a double helix contained in this housing. The air flowed from the helix into a tube contained within the sting and, after flowing the length of the sting, dumped into the model main plenum chamber which was the air distribution center. The purpose of the helix was to absorb pressure and thermal forces on the air supply system between the model plenum and the point of air entry to the helix. However, an independent investigation of the air supply system subsequent to completion of the ADAM tests revealed flow tares which were not negligible. Due to the late receipt of this information, no action was taken. Flow losses in the wind tunnel plumbing were excessive; e.g., at a maximum flow of 7.75 pounds/second, a pressure of only 350 psig was attainable in the air case in the model. Propulsive performance of the model was restricted accordingly.

(2) NASA Langley 16-Foot Transonic Wind Tunnel

Supply air at 1,800 psi was reduced to approximately 700 psi and delivered in six separate supply lines to the sting where the air was then manifolded and flowed through the sting and balance before dumping into the model plenum. The model air supply system used a pair of bellows at the plenum in the air line to isolate the tare forces caused by the air supply system and to reduce the stiffness effects of the air supply system on balance measurements. Overpressure of the bellows, which has an approximate 600 psi rating, was prevented by a rupture disk located in the aft portion of the sting and vented to the tunnel through a hole in the sting access door. The model itself was protected from the effects of high pressure air due to bellows rupture or other plumbing failure by a blowout door in the fuselage shell and by the open area around the sting in the fuselage aft section. An investigation of flow tares was conducted before installation of the model in the tunnel; even though the tares were found to be small, they were included in the data reduction. The desired propulsion mass flow rates could not be obtained because the fans could not be operated at design speeds without exceeding permissible fan bearing temperatures.

h. Air Control and Metering System in the Model

(1) Primary Air Supply System

(a) Main Valve

The main valve controlling flow rate from the model plenum to the primary exhaust was attached to the model plenum and was a remote controlled, translating plug valve. The exit region of the valve formed a bifurcated passage so that flow went to both right and left exhaust systems. Flow regulation was limited to 50 percent of maximum primary flow to prevent possible over-pressurization of the model internal plumbing.

(b) Trim Valves

Trim valves are provided at the entrance to the primary exhaust plenum, one on each side of the model. Their purpose was to make allowance for any difference that existed between the right and left supply systems that would cause unequal pressures or flow rates.

(c) Flow Metering Method

Flow was metered in each of the primary supply systems with a standard ASME orifice operating under supercritical conditions. Pressure at each orifice was measured at two corner taps downstream and two pipe taps, one pipe diameter upstream. Measurements at each station were taken 180° apart and manifolded.

(2) Nose Fan Air Supply System

(a) Main Valve

The valve controlling the flow to the nose fan turbine drive air was a remote control translating plug valve with shut-off capability. This valve exhausted into a manifold with dual exits, since two supply sources were required by the fan turbine.

(b) Flow Metering Method

The same type orifice meter and taps were used in the nose fan system as were used in the primary systems.

(3) Wing Fan Air Supply System

The main valve controlling the wing fan turbine drive flow was attached to the model supply plenum and was a remote controlled, translating plug valve with shut-off capability. This valve exhausted into a smaller plenum which provided entry to the trim valves.

(a) Trim Valves

Manually operated trim valves for each of the four wing fans were attached to the small plenum downstream of the main valve. These valves were the rotating cylinder, eyelid type, and provided the capability of trimming all four fans to the same rpm or operating with fans shut down.

(b) Flow Metering Method

The drive air flow rate to the wing fans was determined by summing the flow rates measured in the other systems and subtracting this sum from the known amount of air supplied to the model.

i. Fans

The fans used were 5.5-inch tip turbine fans with an 0.35 hub-tip radius ratio. They were designed to have a referred mass flow rate of 5.55 pounds per second and a 1.25 pressure ratio at 35,800 rpm. Approximately 1.1 pounds per second of turbine drive air was required at design point. The fans will operate satisfactorily for a short time at 110 percent overspeed.

(1) Diffusing Afterbody

To avoid unfavorable pressure gradients behind the fan as a result of abrupt diffusion in the area behind the fan hub, an afterbody was added to the hub. However, the afterbody could be no longer than the duct transition section due to interference with the fan exhaust duct turning vanes. Since a short afterbody still results in a high local diffusion rate and possible oscillating flow separation, circular vanes were added around the afterbody to reduce the local equivalent conical angle of diffusion and local adverse pressure gradient.

9. MODEL PRESSURE TAP LOCATIONS

a. Static Pressure Tap Locations

Static pressure taps were located on the external wing surfaces (46), on the fuselage (14), on the top and bottom of the wing tip (boom) centerline (14), on the boom tip along the chord line (7), inside the wing fan duct (16), inside the wing leading edge inlets (28), on the nose fan exit ramp (4), and on the upper and lower surfaces of the flap (38). A total of 167 static pressure taps were provided; these taps were all on the left side of the model. When alternate components were provided, such as flaps, static pressure taps were installed on all of the alternate components. The locations of the static pressure taps are indicated in Figure 9.

b. Total Pressure Rakes

The locations of total pressure rakes are described in Paragraph 7 and are illustrated in Figure 9.

10. MODEL DIMENSIONS

The significant model physical dimensions are tabulated in Table II.

11. MODEL TEST CONFIGURATION NOMENCLATURE

The model test configuration nomenclature is tabulated in Table III.

TABLE II. MODEL DIMENSIONS

Fuselage	
Length	6.775 ft
Maximum Height	1.327 ft
Maximum Width	0.833 ft
Wing	
Area (basic top surface – tip to tip)	6.7801 sq ft
(basic top surface – boom centerline to boom centerline)	6.2069 sq ft
(basic top surface – boom centerline to boom centerline plus horizontal outboard tails)*	7.2666 sq ft
Span (boom centerline to boom centerline)	3.1583 ft
(tip to tip)*	3.4500 ft
(tip to tip including low aspect ratio horizontal tails)	4.3766 ft
(tip to tip including high aspect ratio tails)	5.2324 ft
Chord (not including flap)*	1.965 ft
(including flap)	2.452 ft
Taper Ratio	1.0
Sweep	0 degree
Twist	0 degree
Aspect Ratio (wing only)	1.755
(including low aspect ratio horizontal tails)*	2.645
(including high aspect ratio horizontal tails)	3.770
Airfoil Section	Not Standard
Incidence Angle	0 degree
Dihedral	0 degree

*Asterisk identifies parameters chosen for data reduction and presentation

TABLE II. MODEL DIMENSIONS (continued)

<u>Flap</u>	
Area (one side)	0.493 sq ft
Span (one side)	1.015 ft
Chord	0.488 ft
<u>Vertical Tail (Outboard - Each Surface)</u>	
Area (exposed)(1.050 sq ft for both surfaces)	0.525 sq ft
Span	0.9308 ft
Taper Ratio	0.308
Sweep (c/4)	43°50'
Aspect Ratio	1.38
Airfoil Section	NACA 65A010
<u>Vertical Tail (Centerline)</u>	
Area (exposed)	1.05 sq ft
Span	1.203
Taper Ratio	0.308
Sweep (c/4)	43°50'
Aspect Ratio	1.38
Airfoil Section	NACA 65A010
<u>Horizontal Tail, Low Aspect Ratio (One Surface)</u>	
Area (exposed)	0.4119 sq ft
(total)	0.52985 sq ft
Span (b/2)	0.6304 ft
Taper Ratio	0.316
Sweep (c/4)	54°10'39"
Aspect Ratio	0.752
Airfoil Section	NACA 65A010
Dihedral	0 degree

TABLE II. MODEL DIMENSIONS (concluded)

Horizontal Tail, High Aspect Ratio (One Surface)	
Area (exposed)	0.4605 sq ft
(total)	0.52985 sq ft
Span (b/2)	1.0583 ft
Taper Ratio	0.385
Sweep (c/4)	28°12'
Aspect Ratio	2.113
Airfoil Section	NACA 65A010
Inlet Capture (Highlight) Areas	
Basic Wing Fan Inlet	0.2222 sq ft
Long Wing Fan Inlet	0.1583 sq ft
Nose Fan Inlet	0.2114 sq ft
Nozzle Exit Areas	
Wing and Nose Fans, per Fan	
Low Speed Nozzles	0.1291 sq ft
High Speed Nozzles	0.1111 sq ft
Primary Nozzles, per Side	
"Mass Flow" Nozzles	0.0284 sq ft
"Volume Flow" Nozzles	0.0612 sq ft

TABLE III. MODEL TEST CONFIGURATION NOMENCLATURE

W — VANE BOXES

Vane boxes provide wing fan thrust (secondary exhaust flow basic deflections) of 0° (cruise mode), 30°, 60°, and 90°

W - Cruise mode

W^{xx}_{yy} - Vane box configuration model part indicated by:

xx - deflection angle for left side
yy - deflection angle for right side

XX configuration only, indicates left- and right-side deflection are equal. Vane boxes providing 60° and 90° deflection have ±10° increments available from the basic deflection angle allowing differential thrust vectoring

N — NOSES

N^{xx}_{1} - Nose fan exits swiveled for the bifurcated configuration

N^{xx}_{2} - Nose fan exits swiveled to simulate single exit

N_3 - Nose fan inlet and exits plugged to simulate the tare nose (for use in low speed tunnel only)

N_4 - Tare nose

XX - indicates nose fan deflection angles

F — FLAPS

All deflection angles from -20° to 100° are available independently on each side by remote control allowing differential deflections for use as ailerons

F^{xx}_{1} - Short chord with circular arc upper surface and small leading edge radius

F^{xx}_{2} - Short chord with flat upper surface and small leading edge radius

F^{xx}_{3} - Long chord with slightly circular arc upper surface and small leading edge radius

F^{xx}_{4} - Long chord with slightly circular arc upper surface and large leading edge radius

TABLE III. MODEL TEST CONFIGURATION NOMENCLATURE (continued)

FLAPS USED AS AILERONS

F_{1A}^{xx} - Flap number one (F_1^{xx}) used as ailerons with the left-side deflection indicated in flap deflection angular directions with the right-side deflection equal in magnitude but opposite direction to produce a positive rolling moment

T – UPPER WING SURFACE TRAILING EDGE

T_1 - Approximately flat trailing edge for simulation of primary exhaust on a volume flow basis when used in conjunction with flaps F_1, F_2, and F_3, or on a mass flow basis when used in conjunction with F_4 flaps

T_2 - Curved trailing edge for simulation of primary exhaust flow on a mass flow basis when used in conjunction with flaps F_1, F_2, and F_3

I – INLET LEADING EDGE CONTOURS

I_1^1 - Short length with large leading edge radius

I_2^2 - Short length with medium leading edge radius

I_3^3 - Short length with small leading edge radius

I_4 - Long length with large leading edge radius

Superscript on I_1, I_2, I_3 indicates these inlet leading edges are constructed in two parts allowing combinations of upper and lower surface inlet contours to define the inlet configuration

B – BOOMS

(Symbols applicable only to Part IV)

B_1 - Short boom

B_2 - Long boom (short boom with boom extension added)

TABLE III. MODEL TEST CONFIGURATION NOMENCLATURE (continued)

V – VERTICAL TAILS

One set of vertical tails for mounting on the wing booms in the vertical plane are constructed for the model. Incidence angles of 0°, 2.5°, 5°, and 10° are available, defined by leading edge outboard from the model center-line. Incidence angles must be individually set manually.

V_1^{XX} - Vertical tail on configuration at forward mounting position with incidence angle indicated by XX

V_2^{XX} - Aft mounting position

V_3 - Centerline vertical tail

H – HORIZONTAL TAILS

Two sets of horizontal tails for mounting in two positions on the wing booms in the horizontal plane are constructed for the model. Incidence angles in 10° increments from +10° (leading edge up) to -40° are available. Incidence angles must be individually set manually. Both sets have the same area. Aft mounting positions required installation of boom extension parts.

H_1^{XX} - Low aspect ratio at forward mounting position (60° leading edge sweep angles)

H_2^{XX} - High aspect ratio at forward mounting position (28.2° quarter-chord sweep angle)

H_3^{XX} - Same as H_2 at aft mounting position

H_4^{XX} - Same as H_1 at aft mounting position

Incidence indicated by XX

HORIZONTAL TAILS USED AS AILERONS

HX_{ZZ}^{YY} - X – indicates horizontal tail set used and position; H_1, H_2, H_3, or H_4. YY – indicates left-side incidence angle. ZZ – indicates right-side incidence angle.

TABLE III. MODEL TEST CONFIGURATION NOMENCLATURE (concluded)

S – SPOILERS

Individual spoiler plates constructed for mounting on the left side only provide deflection angles of 15°, 30°, and 60°

S^{XX} - XX – indicates deflection angle specifying the specific model part installed

D – NOSE FAN DETACHER FLAP

Individual flap plates having angles of 15°, 30°, and 45° will be used for evaluation of suckdown losses to ensure test runs with the nose fan exhaust flow detached from the bottom surface of the model fuselage.

D^{XX} - XX – indicates detacher flap angle specifying the specific model part installed

NOTE:

This test configuration nomenclature is not necessarily consistent with the nomenclature used in describing the design of the model.

SECTION IV

TEST FACILITIES

1. GENERAL

The model was tested in the three test facilities described briefly below.

2. VAD LOW SPEED WIND TUNNEL

The VAD low speed wind tunnel is a continuous flow facility with a 7-foot by 10-foot atmospheric pressure closed test section. The maximum speed is 240 miles per hour. A six-component, external balance is normally used for force measurements, although internal strain gage balances can also be used. The data system provide both tabulated data and punched card output. Pressure measurements utilize either water manometers or strain gage pressure transducers and pressure scanning switches. In operation since 1955, this facility has conducted many thousands of hours of testing of aircraft, missile, and helicopter configurations. This tunnel is illustrated schematically in Figure 10.

3. NASA LANGLEY RESEARCH CENTER 7-FOOT BY 10-FOOT WIND TUNNEL

The Langley 300 miles per hour, 7-foot by 10-foot tunnel is located in Building 1212A and is under the direction of the Full-Scale Research Division. This tunnel is illustrated schematically in Figure 11.

a. 7-Foot by 10-Foot Test Section

Speed is variable from 0 to 300 miles per hour in this continuous-flow, fully closed test section. The stagnation pressure and temperature are atmospheric. The tunnel Reynolds number per foot ranges from 0 to 2.5×10^6. Model support systems include semispan, sting, and strut. The test section is 15 feet long.

b. 17-Foot Test Section

Speed is variable from 0 to 80 miles per hour in this continuous-flow test section which is 17 feet wide and 15.8 feet high. A moving belt can be installed for ground-effects tests. Tunnel stagnation pressure and temperature are atmospheric. Tunnel Reynolds number per foot ranges from 0 to 0.7×10^6. Model support systems include semi-span and sting. The test section is 15 feet long. This section is designed to test V/STOL models through high angles of attack with minimum tunnel wall effects.

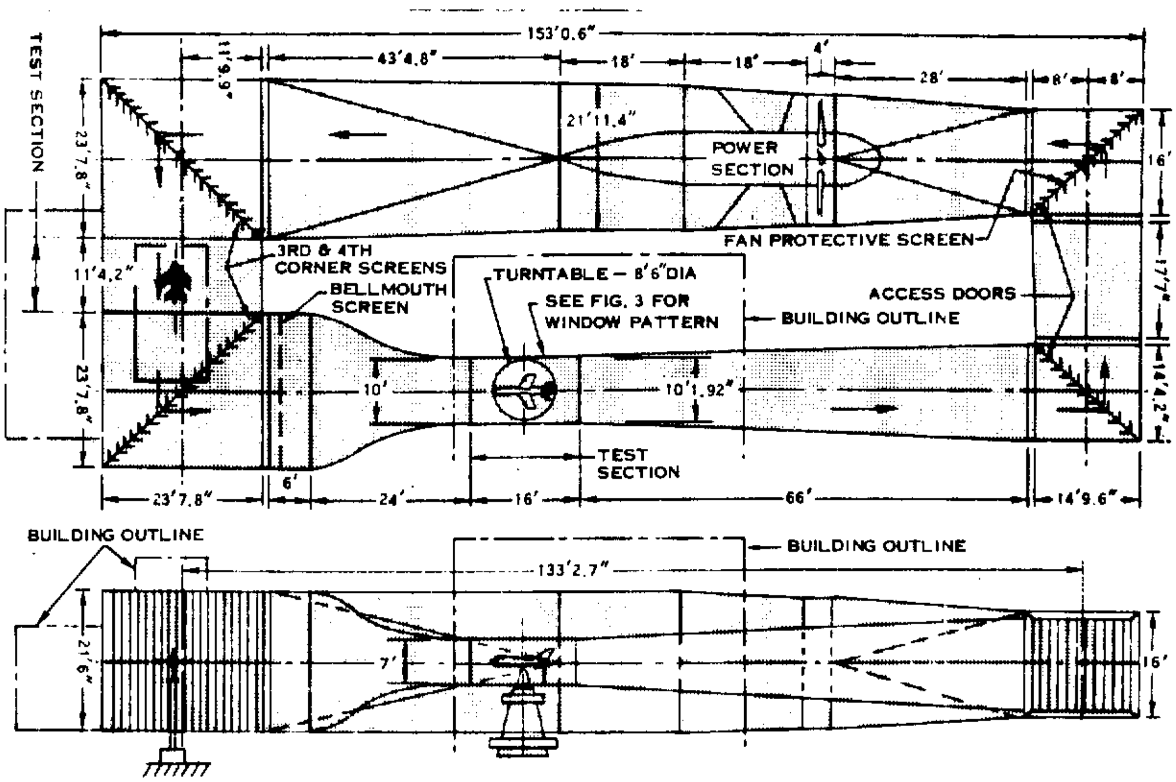


Figure 10. VAD Low Speed Wind Tunnel

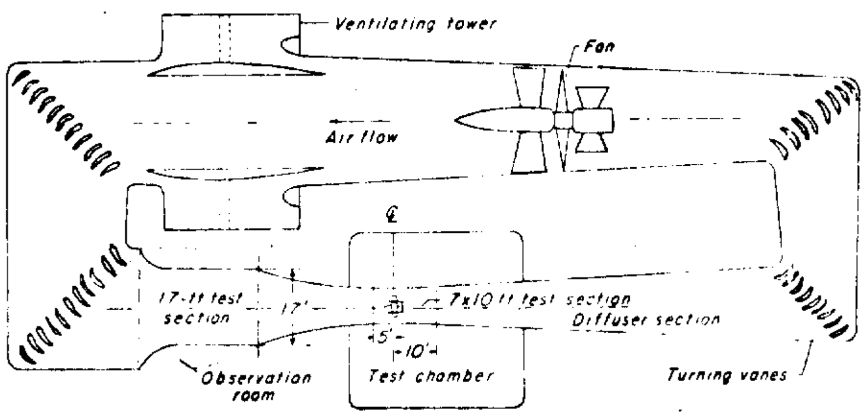


Figure 11. LRC 300-mph 7-Foot by 10-Foot Wind Tunnel

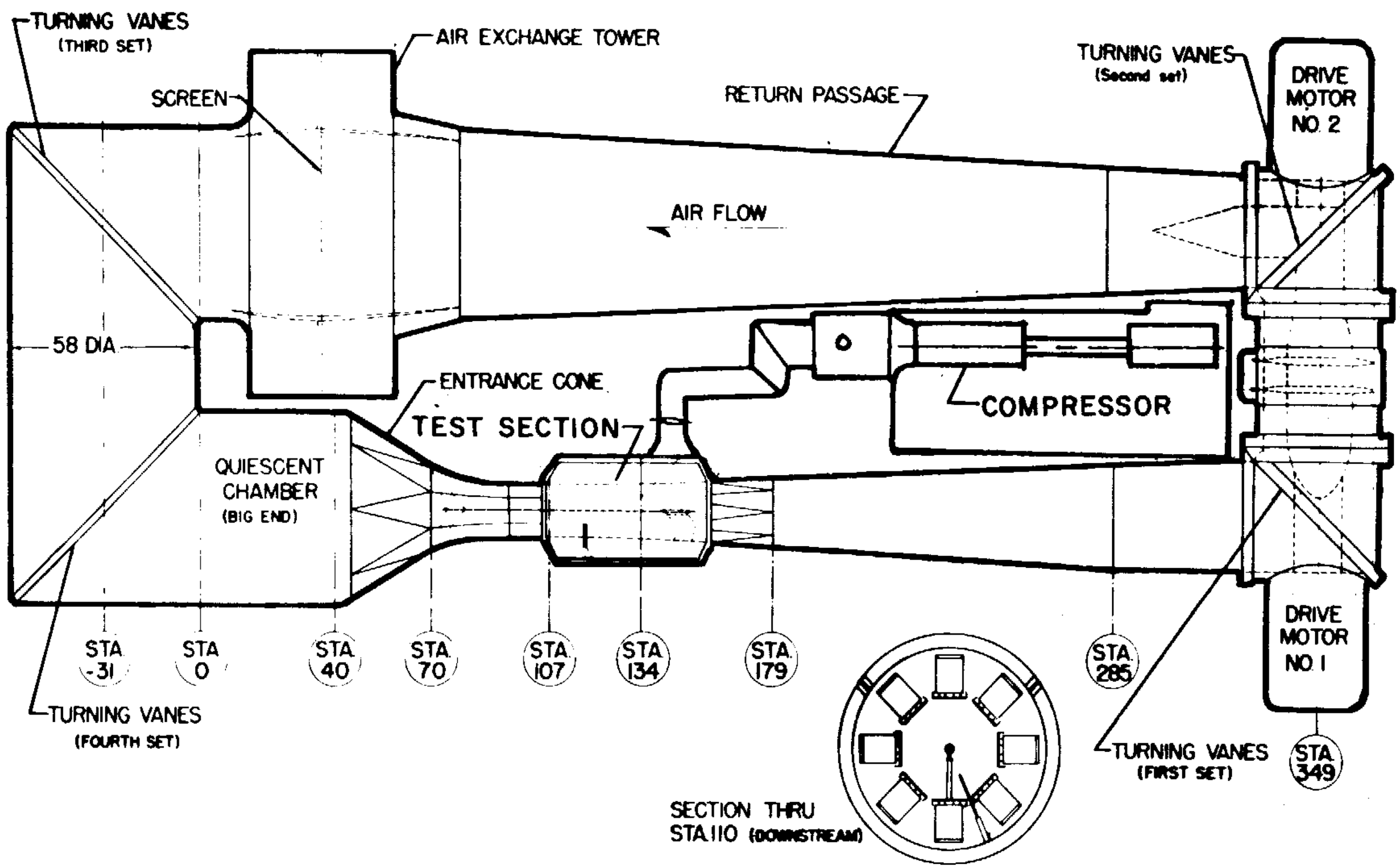


Figure 12. LRC 16-Foot Transonic Tunnel

4. NASA LANGLEY RESEARCH CENTER 16-FOOT TRANSONIC WIND TUNNEL

The Langley 16-foot transonic wind tunnel, illustrated in Figure 12, is located in Building 1146 and is under the direction of the Full-Scale Research Division. The test medium is air with air exchange for cooling. Model mounting consists of wall, sting, and strut supports. This is a continuous, single-return, atmospheric tunnel with slotted octagonal throat and test section which is 15.5 feet wide and 22 feet long. It can be used for propulsion tests with 90 percent hydrogen peroxide or compressed air. Examples of operating conditions are as follows:

Stagnation pressure .... Atmospheric

Stagnation temperature, °R .... 510 to 650

Reynolds number per foot .... 1.2×10^6 to 3.7×10^6

Mach number .... 0.2 to 1.3

Dynamic pressure, lb/sq ft .... 58 to 830

SECTION V

BALANCES

1. GENERAL

Different balances were used in each of the three wind tunnels. These balances are described below.

2. VAD LOW SPEED WIND TUNNEL

The tunnel's standard six components, pyramidal, external balance was used for the shakedown testing in this tunnel.

3. LRC 17-FOOT TEST SECTION

In the 17-foot test section at LRC, the model was tested with the NASA No. 1612 six-component balance. This balance was rated for the following loads:

NF	1,200 lb	PM	2,000 in-lb
SF	500 lb	YM	2,000 in-lb
AF	250 lb	RM	1,000 in-lb

This balance is quite accurate and is within 0.5% on all components. It does not provide for an integral compressed air supply. The centerline of the balance was offset 1.875 inches from the centerline of the air supply tube. Flow tares were not measured during this program.

4. LRC 16-FOOT TRANSONIC WIND TUNNEL

a. Decision to Use a New Balance

A new internal strain gage balance was provided for the testing in the LRC 16-foot transonic wind tunnel. This balance was given the designation VTB-3. The transonic testing of the 0.167-scale model presented unique balance problems. The anticipated loads were quite large, as compared to the capacity of the majority of balances. The model required 10 pounds/second air supply, which preferably should flow through the center of the balance. There was no balance in use, or being developed, that would meet these requirements. Therefore, the scope of the originally proposed program was enlarged to include design, fabrication, calibration, and evaluation of a balance and its related equipment within the scope of work of the subject contract.

b. Design Concept

(1) Air Supply

From VAD's experience in designing systems to supply air and hydraulic fluid to powered models, the externally pressurized bellows system was chosen as having the best all-around characteristics. The bellows assembly is compact and affords a very "soft" structural bypass in parallel with the balance. With the properly assembled and attached bellows, hysteresis and nonrepeat (due to having a load path other than the balance) will be insignificant. And lastly, the bellows and balance can be designed as a unit for ease of model buildup.

(2) Balance

As is the usual case in designing a balance for a particular model, it was necessary to compromise between the long length desired for high resolution and the short length desired to leave space for the many components in the model. Because of the 2-inch-diameter air passage through the balance, (1) VAD's conventional moment-type design was the most suitable, and (2) one-piece construction was found to be preferable. (Initially, the balance was a one-piece machining with axial force elements attached. Removal of these elements will be discussed subsequently.) Taper joints were chosen for secure attachment at both ends, as VAD had found tapers to be reliable. However, VAD's experience did not include thin-wall tapers, and the tapers did prove to be a source of nonrepeatability.

c. Problem Areas and Solutions

Two basic problems came to light during the calibration of the balance: (1) nonrepeat and zero shift in the bending moment bridges, and (2) slippage of the axial force elements.

(1) Nonrepeatability and Zero Shift of Bending Moment Bridges Adjacent to the Attach Tapers

Extensive experimentation proved that erratic data were the result of slipping and redistribution at the taper joint, caused by the thin walls. This affected the two bridges that generate normal force and pitching moment, and the two that generate side force and yawing moment. The normal force-pitching moment problem was solved by relocating the bridge in an area between the roll flexures, well removed from the taper joints. These bridges proved to be most satisfactory from all points of consideration. The side force-yawing moment bridge could not be relocated because there was no other area having adequately high strain. These bridges proved to have an accuracy of 3.0% for side force and 2.0% for yawing moment. The conclusions were that: (1) taper walls should be thicker, and (2) tapers should be farther removed from the strain gage bending section (balance should have been 2 to 4 inches longer). Neither of these revisions could be made without machining a new balance and making extensive modifications to the model.

(2) Slippage of Axial Force Elements

The separately machined axial force elements were chosen because this made it possible to leave more structure at the center of the balance. Initially, the elements were attached with dowel pins and screws, which were found to be inadequate. Then the elements were welded in place by electron-beam welding. This appeared satisfactory until it was found that a certain combination loading would cause unacceptable slippage. Repetition of that loading would not cause slip. To reproduce the shift, it was necessary to repeat a certain sequence of loadings ending with the culprit combination. The conclusion was that the welds were yielding due to the high flexural shear induced by normal force and pitching moment. The solution for this problem then was to remove the axial elements and locate axial force bridges on the roll flexures. Satisfactory data were obtained after this change, even though the gages were located where total strain was greater than was desirable.

d. Description of Final Balance Configuration

The VTB-3 balance is a one-piece machining, 4.25 inches in diameter and just under 24 inches overall length (Figure 13). The inside diameter is 2.25 inches, which leaves one-eighth-inch clearance all around the air supply tube (Figure 14).

The one-piece machining may be considered as a thick-wall tube which is cut diagonally through the center, but with the two parts left integrally connected by flexures machined at each end of the diagonal cut. These flexures are gaged to measure rolling moment and axial force. One normal force bridge is located near the upper outer beam fibers of the forward section, approximately 5 inches from the aft rolling flexures. The other normal force bridge is located similarly on the aft section of the balance. Beyond each of the flexure joints is a bending section which is gaged to measure bending moments due to side force. Outside of the bending sections are the fore and aft tapers for attaching to the model and sting respectively. The forward taper is drawn into the model adapter by angled screws, and the aft taper is drawn by a contra-nut.

The balance was temperature compensated for the range of 75°F to 175°F, and modulus compensated for the decrease of Young's modulus with an increase of temperature in the 0°F to 400°F range. Wiring is routed through holes in the bending sections and in grooves along the outer surface, and then potted in place. The wires are gathered into two bundles ahead of the aft taper and are routed along either or both sides of the sting.

e. Sensitivity, Accuracy, and Pressure-Flow Tares

The following sensitivity values are from calibrations performed by VAD. The accuracy values are comparisons of the printout from the data reduction routine, with loads actually applied in calibration. The calibrations used for generating the routine (and tunnel use) were performed under the supervision of the Instrument Research Division (IRD) of NASA-LRC by a NASA contractor. The routine was generated by IRD personnel. The calibrations were single- and multicomponent loadings up to and including 100% of all six components. The accuracy shown is not a statistical value, but is the greatest error shown for each component in all of the loadings performed.

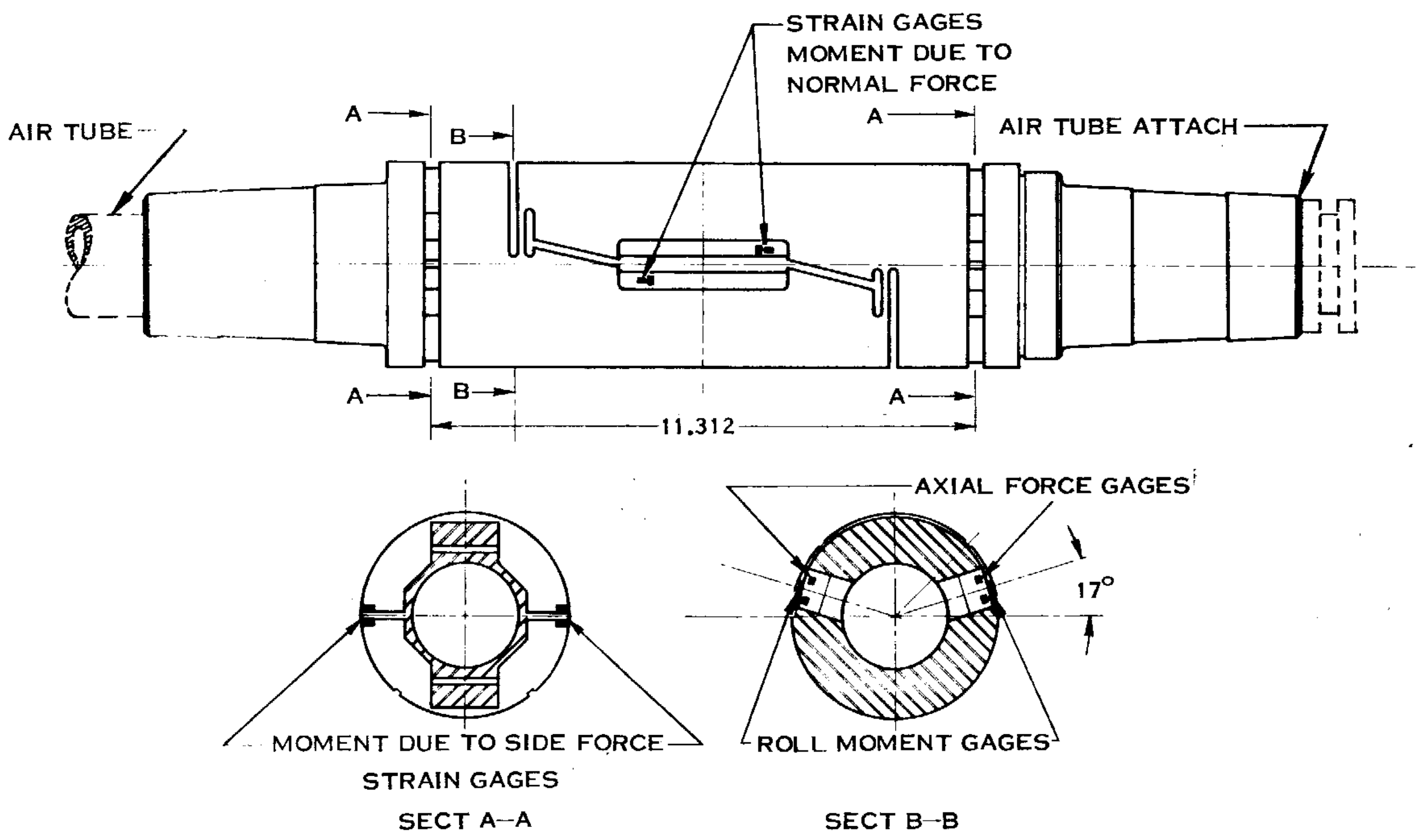


Figure 13. Functional Schematic – VTB-3 Balance

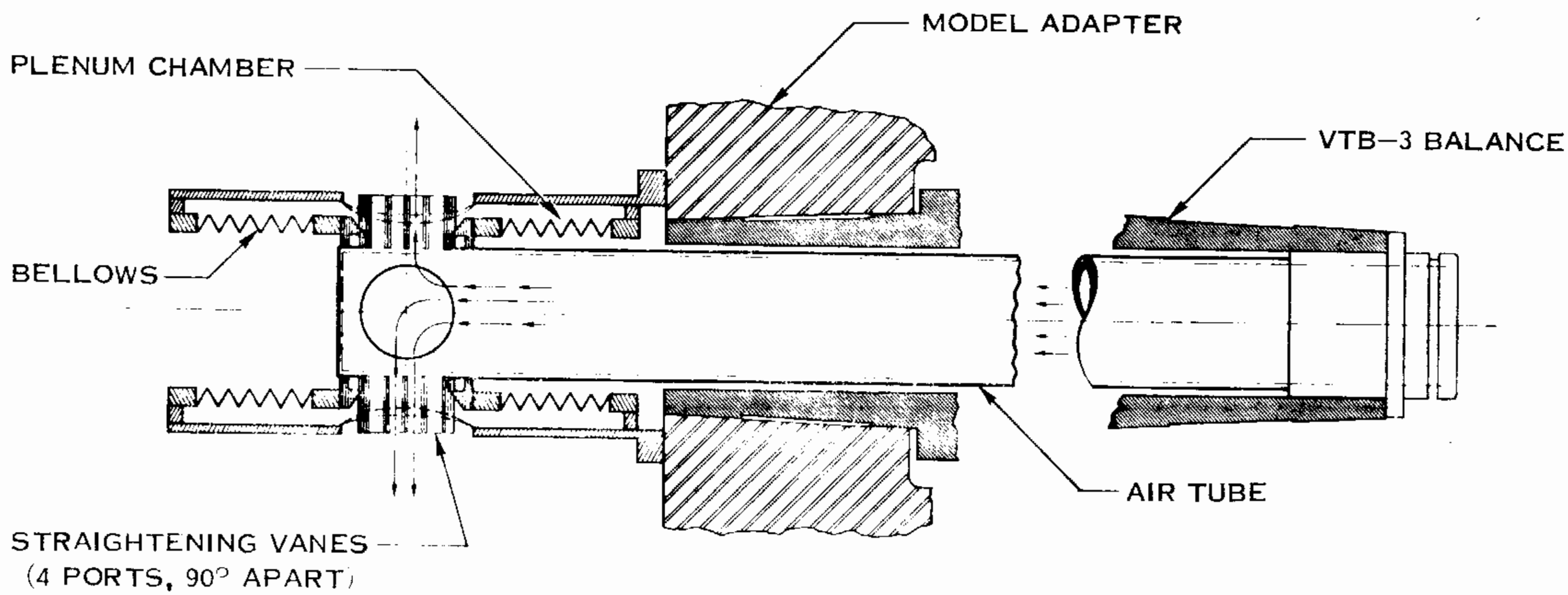


Figure 14. Functional Schematic – VTB-3 Integral Air Supply System

SENSITIVITY

Component	Value	Bridge	μ In./In.	MV/V	Accuracy (Percent Full Scale)
Normal Force	6,000 lb	R_1	240	0.48	
		R_2	211	0.42	0.05
Pitching Moment	36,000 in-lb	R_1	350	0.72	
		R_2	340	0.68	0.02
Side Force	1,500 lb	R_3	336	0.67	
		R_4	356	0.71	3.0
Yawing Moment	11,500 in-lb	R_3	352	0.70	
		R_4	376	0.75	2.0
Rolling Moment	14,000 in-lb	R_5	625	1.25	0.8
Axial Force	500 lb	R_6	376	0.75	1.2

The normal force and pitching moment accuracies are equivalent to those of the best balances in service. The side force-yawing moment accuracies are unsatisfactory for general testing, but for this particular program those two components were the least important, and the accuracies were adequate. The rolling moment accuracy is quite sufficient, although not nearly as good as was anticipated. Actually, the roll component was as good as NF and PM, but the thin-wall tapers permitted enough shift of the calibration body that normal force sometimes applied a small roll moment. The axial force accuracy was also less than desired, but actually it was quite good when considering that the 500-pound axial force was measured within 6 pounds, in the presence of 6,000 pounds normal force, 36,000 inch-pounds pitching moment, and 14,000 inch-pounds rolling moment.

Pressure and flow tares were either undetectable or insignificant on all components except axial force, which is as expected. With design pressure of 600 psi and flow of 11.0 pounds/second, which was 10% above design flow, the combined pressure and flow tare was 11.0 pounds, or 2.2% of full scale.

f. Comparison of VAD VTB-3 Balance with NASA No. 1612 Balance

(1) Flow Tares

Investigation conducted independently of this contract has indicated that the NASA No. 1612 balance with the separate air supply had significantly greater flow tares than the VTB-3 balance with its integral air supply.

(2) Load Ratios

The VTB-3 balance had to be designed to much more difficult load ratios.

	No. 1612	VTB-3
PM/NF	1.67	6.0
NF/AF	4.8	12.0
PM/AF	8.0	72.0
RM/AF	4.0	28.0

As the ratio increases, it becomes more difficult to separate normal force from pitching moment. As any of the other ratios increase, the roll flexures become more rigid, and it becomes difficult to obtain axial force sensitivity and accuracy.

SECTION VI

DATA REDUCTION

1. VAD LOW SPEED WIND TUNNEL

Data reduction was accomplished by the VAD low speed wind tunnel group. Force and moment data from the external pyramidal balance were presented in coefficient form and tabulated in wind axes and stability axes systems. The moment coefficients were about the 36% wing chord on waterline 10.

2. LRC 17-FOOT TEST SECTION

Data reduction was accomplished by the Langley Research Center Data Processing group and is not detailed here. Force and moment data recorded from the internal balance were converted to coefficient form and tabulated in stability axes and body axes systems. The balance center was located on waterline 8.125 at the 36% chord station inside the model fuselage. The moment coefficients were transferred to the wing quarter chord on waterline 10 for tabulation.

3. LRC 16-FOOT TRANSONIC WIND TUNNEL

Data reduction was accomplished by the Langley Research Center Data Processing group and is not detailed here. Force and moment data recorded from the internal balance were converted to coefficient form and tabulated in stability axes and body axes systems. The balance center was at the wing quarter chord on waterline 10, and moment coefficients were presented about this point.

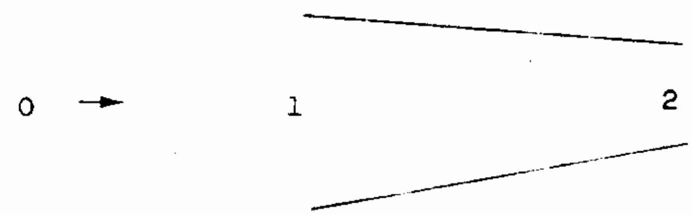
4. CALIBRATION OF THE PROPULSION SYSTEM

Propulsion data reduction was accomplished in two steps: the first step was propulsion system calibration, and the second was reduction of test data obtained based on calibration results. To calibrate the nose fan exhaust system, a set of 16 tubes was installed in the exit plane of each exhaust nozzle. Wing fan calibration rakes had 20 tubes per fan when calibrating the 0°, 60°, and 90° vane boxes, and 24 tubes when calibrating the 30° vane box. Measurements were made only on the right-hand wing exhausts. Flow conditions on the left side were assumed equal to those on the right with the left-hand fans operating at the same point. The calibration rake for the primary exhaust system had six total pressure tubes, and again, data were taken only on the right side of the model. RPM of all fans, fan inlet total temperature, right-hand primary orifice upstream pressure, and temperature and downstream pressure were also recorded. In the cruise configuration, each of the model exits had a set of three permanent total pressure tubes which were manifolded to give an indicated total pressure. Each of the fan exits had a set of three total temperature probes connected in parallel to give an average temperature. The primary exit had one total temperature probe located in the left-hand nozzle. For configurations of wing fan efflux vectoring greater than 0°, no data were available from permanent wing fan instrumentation. Wing fan efflux temperature was assumed equal to the nose fan efflux total temperature. The data thus obtained were used in a computer program which integrates the pressures from the calibration rakes over the areas involved. These integrated pressures were then used with the measured temperatures and model geometry to calculate thrust and mass flows from the calibrated nose and wing fan exhaust areas. Primary weight flow was calculated from orifice measurements and geometry. Finally, listings were made which show the relation between exhaust total to ambient static pressure ratio calculated from readings from the permanent pressure tubes and the integrated pressures for the cruise configuration, and between wing fan rpm and integrated exhaust total to ambient static pressure ratio for wing fan vectoring angles greater than 0°.

5. REDUCTION OF PROPULSION TEST DATA

To carry out the second step of data reduction, separate tables of indicated and integrated exhaust total to ambient static pressure ratio as a function of Mach number were made for the nose fan, inboard and outboard wing fans, and primary exhaust systems. For vectoring angles of 30°, 60°, and 90°, the wing fan tables contained fan rpm rather than indicated pressure ratio. To find the results of a data run, Mach number and rpm or indicated pressure ratio were used to enter these tables and find the actual exit pressure ratio. These pressure ratios, together with other data from model permanent instrumentation, were used in a computer routine to calculate gross thrust, ram drag, weight flows, and momentum coefficient, C_μ. Tables of gross thrust from the calibrated exits in the cruise configuration were made as a function of Mach number and fan rpm. These tables were stored in the computer with the other tables and provided thrust values in the event of permanent pressure tube malfunction so that not all the run value would be lost.

6. CALIBRATION AND DATA REDUCTION THRUST AND AIRFLOW EQUATIONS



Zero subscript denotes stagnation conditions

Energy:

$$V_1^2/2gJ + h_1 = V_2^2/2gJ + h_2 = h_0$$

$$V_2 = \sqrt{2gJ(h_0 - h_2)}$$

$$V_2 = \sqrt{2gJC_pT_2(T_{02}/T_2 - 1)}$$

$$V_2 = \sqrt{2gJC_pT_2\left[(P_{02}/P_2)^{(\gamma - 1/\gamma)} - 1\right]}$$

$$V_2 = \sqrt{2gJC_P\left(T_{02}(P_2/P_{02})^{(\gamma - 1/\gamma)}\right)\left[(P_{02}/P_2)^{(\gamma - 1/\gamma)} - 1\right]}$$

$$V_2 = \sqrt{2g(\gamma/\gamma - 1)(RT_{02})(P_2/P_{02})^{(\gamma - 1/\gamma)}\left[(P_{02}/P_2)^{(\gamma - 1/\gamma)} - 1\right]} \qquad \text{Eq (1)}$$

$$V_2 = \sqrt{2g(\gamma/\gamma - 1)RT_2\left[(P_{02}/P_2)^{(\gamma - 1/\gamma)} - 1\right]}$$

Continuity:

$$\dot{m} = \rho_1A_1V_1 = \rho_2A_2V_2 = P_2A_2V_2/RT_2$$

$$\dot{m} = (P_2A_2/RT_{02})(P_{02}/P_2)^{(\gamma - 1/\gamma)} \sqrt{2g(\gamma/\gamma-1)(RT_{02})(P_{02}/P_2)^{(\gamma-1/\gamma)}\left[(P_{02}/P_2)^{(\gamma-1/\gamma)} - 1\right]}$$

$$\dot{m} = P_2A_2\sqrt{(2g/RT_{02})(\gamma/\gamma-1)(P_{02}/P_2)^{(\gamma-1/\gamma)}\left[(P_{02}/P_2)^{(\gamma-1/\gamma)} - 1\right]} \qquad \text{Eq (2)}$$

Thrust:

$$F_G = \dot{m}V_2/g \qquad (\Delta P \cdot Aex \text{ term negligible})$$

Combining equations (1) and (2) in the thrust equation gives:

$$F_G = 2P_2A_2(\gamma/\gamma-1)\left[(P_{02}/P_2)^{(\gamma-1/\gamma)} - 1\right]$$

Ram Drag:

$$D_{RAM} = \dot{m}V_0/g$$

Momentum Coefficient:

$$C_\mu = F_G/qS$$

Orifice Equation:

$$\dot{W} = KYA_O P_1 \sqrt{2g(1-r)/RT_1}$$

where:

$K = .608 + .415\beta^4$

A_O = Orifice Area

P_1 = Upstream Static Pressure

T_1 = Upstream Static Temperature

P_2 = Downstream Static Pressure

$r = P_2/P_1$

β = Diameter Orifice/Diameter Pipe

$Y = f(r)$

$$Y = 1.005 - (.41 + .35\beta^4)(1-r/1.4) \qquad (1)\ r > .63$$

$$Y = 1.015 - (.4244 + 1.28\beta^4 - 1.6\beta^6)(1-r-.1r^2)/1.4 \qquad (2)\ r < .63$$

REFERENCES

1. R.T. Stancil; L.J. Mertaugh, Jr.: "Analysis of a Low Speed Wind Tunnel Test of a High Rate Vectored Propulsion Flow Model," (U), LTV/VAD Report No. 2-53310/4R-2166, ARO-D Project No. 5360E, Contract No. DA-31-124-ARO-D-262, dated 15 February 1965. Unclassified

2. LTV/VAD Report No. 2-55400/5R-50303, "Technical Proposal – Design Fabrication Testing and Data Analysis of ADAM II Concept (Propulsive Wing)," dated 5 August 1965.

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1. ORIGINATING ACTIVITY *(Corporate author)* LTV Aerospace Corporation	2a. REPORT SECURITY CLASSIFICATION Unclassified 2b. GROUP
3. REPORT TITLE Design, Fabrication, Testing, and Data Analysis of ADAM II Concept - Part I	
4. DESCRIPTIVE NOTES *(Type of report and inclusive dates)* Final Report 2 December 1966 - 7 July 1967	
5. AUTHOR(S) *(Last name, first name, initial)* Part I - James G. McClure; Justin F. Shumway; Glen P. Cragin, Jr.	

6. REPORT DATE	7a. TOTAL NO. OF PAGES	7b. NO. OF REFS
May 1968		

8a. CONTRACT OR GRANT NO. AF33(615)-3293 b. PROJECT NO. 1366 c. Task No. 136617 d.	9a. ORIGINATOR'S REPORT NUMBER(S) 9b. OTHER REPORT NO(S) *(Any other numbers that may be assigned this report)* AFFDL-TR-68-31, Part I

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13. ABSTRACT

This report contains in four parts, the details of design, fabrication, low and transonic speed wind tunnel testing of a powered propulsive wing model of the LTV ADAM II, V/STOL Aircraft Concept. Part I of this report contains the details of the design and fabrication of the model and includes a description of a new type of "flow-thru" internal strain-gage balance that was developed specifically for testing the model at transonic speeds. Parts II, III, and IV contain analyses of the results from three separate wind tunnel tests. Results presented in these volumes concern the hover, transition, and cruise flight modes. Adequate low speed pitch control power is demonstrated with the use of a vectored thrust nose fan. Cruise mode tests indicate that satisfactory flying qualities can be achieved. A high drag rise Mach number is verified. Requirements for further wind tunnel testing are indicated.

14. KEY WORDS	LINK A		LINK B		LINK C	
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b. Powered Model Testing						
c. "Flow-Thru" Balance						
d. High Bypass Ratio Fans						
e. Vectored Thrust						
f. Propulsive Interactions						
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h. Jet Flap						
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