NASA TECHNICAL NOTE

NASA TN D-3748

NASA TN D-3748

GPO PRICE $ ____________

CFSTI PRICE(S) $ 2.50

Hard copy (HC) ____________

Microfiche (MF) .75

ff 653 July 65

FACILITY FORM 602

N67 11960	
(ACCESSION NUMBER)	(THRU)
75	1
(PAGES)	(CODE)
	11
(NASA CR OR TMX OR AD NUMBER)	(CATEGORY)

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by William C. Moseley, Jr., and Joseph C. Martino

Manned Spacecraft Center
Houston, Texas

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • DECEMBER 1966

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Springfield, Virginia 22151 – Price $2.50

ABSTRACT

An important phase of the Apollo spacecraft development is the analysis and modification resulting from the application of aerodynamic data acquired through the Apollo wind tunnel testing program. A brief history of the aerodynamic development of the Apollo configurations is presented. Basic vehicle components and the purpose and the scope of the Apollo wind tunnel testing program are discussed, with an introduction to models, facilities, and methods of testing. Problem areas that were encountered in the design evaluation and the methods of their solution are introduced.

CONTENTS

Section | Page

TABLES

FIGURES

Figure | Page

APOLLO WIND TUNNEL TESTING PROGRAM -

HISTORICAL DEVELOPMENT OF GENERAL CONFIGURATIONS

By William C. Moseley, Jr., and Joseph C. Martino*
Manned Spacecraft Center

SUMMARY

In 1959 feasibility studies were initiated for an advanced spacecraft system capable of manned circumlunar and earth-orbital flights. The project, assigned the name Apollo, was reoriented in the spring of 1961 to include manned lunar exploration. One important phase in the development of the Apollo spacecraft was the analysis and subsequent modification resulting from the application of input aerodynamic data derived from the Apollo wind tunnel testing program (AWTTP). A multifacility and model program was initiated to obtain aerodynamic data necessary for the evaluation of the theoretical Apollo design, define any problem areas encountered, and provide input information necessary for the solution of these problems. Through the proper application of those data derived from the wind tunnel investigation there was a systematic development of the basic configuration to the present production model.

INTRODUCTION

Project Apollo is one step in man's arduous journey into space. This project was begun in late 1959 when personnel from several National Aeronautics and Space Administration (NASA) centers recommended a circumlunar flight and earth-orbiting laboratory program. The program was initiated and assigned to the NASA Space Task Group.

The aerospace industry was invited in September 1960 to recommend, define, and substantiate the most feasible approach to an advanced spacecraft and systems capable of manned circumlunar and earth-orbital flight. The plan was to include a program for research and development in all necessary areas. Feasibility proposals were submitted by Republic Aviation Corporation,

*ITT/Federal Electric Corporation.

Vought Astronautics, Goodyear Aircraft Corporation, and Boeing Airplane Company. Each company submitted a plan for the management of program control and guidelines for a technical approach. Technical aspects covered by these organizations included:

1. Practical systems and analytical techniques defining objectives and the integration of subsystems that would lead to a final definition of subsystem characteristics.

2. Statement of results.

(a) Recommendations for the Apollo system.

(b) Specifications of system and subsystem performance.

(c) Reliability goals.

(d) Development programs including costs, test programs, and facility requirements.

(e) Areas demanding research.

The proposed feasibility studies and combined NASA efforts formulated the texts of actual study contracts awarded General Electric, General Dynamics/Convair, and Martin Company. Results of the study contracts, NASA Space Task Group study results, and independent company-funded studies (made by Grumman Aircraft Engineering Corporation and North American Aviation) were used in formulating design requirements and specifications for the initial Apollo spacecraft.

On May 25, 1961, Project Apollo was reoriented to achieve a lunar landing before 1970. Before the end of 1961, North American Aviation became the prime spacecraft contractor, and the goals of Project Apollo were clearly defined.

Project Apollo is a multimission effort with each mission serving to qualify objectives of subsequent missions. Each mission aids in developing the technology of manned space flight to the state-of-art, enabling man to land on and explore the moon and return safely to earth. The Apollo spacecraft will be the vehicle making the journey.

The practicality of efficient usage of weight and the economics of time and money necessitated minimum change to the spacecraft in meeting specific mission requirements. In the formulation of design criteria, the guidelines established by NASA were developed around certain stipulations. These stipulations were:

1. Three-man crew.

2. Fourteen-day mission.

3. Weight control in relation to launch vehicle capability.

4. Progressive step-by-step evaluation and training for earth-orbit flights, to circumlunar flights, to lunar orbit and lunar landing, and a return to earth.

5. Rendezvous of system components in earth orbit, later modified to lunar-orbit rendezvous.

Considering the above factors, the separable module principle of the spacecraft conformed most closely to the requirements. With progressive modifications the spacecraft components were defined originally as the:

1. Command module (CM), the spacecraft command center where all crew-initiated functions are exercised. As the inflight command center, the CM contains equipment for communication, navigation, guidance, control, computing, and display.

2. Launch escape vehicle (LEV), providing a means of adequate escape capability in the event of a malfunction of the booster or spacecraft. The LEV is used only during the atmospheric flight of the ascent trajectory. The LEV is composed of an escape rocket, escape tower, and CM. After the atmospheric portion of the ascent trajectory is completed, the escape tower and rocket are jettisoned.

3. Service module, unmanned and provided with propulsion system and stores. The service module contains other systems that do not require crew access for operation or maintenance during flight.

4. Mission module, or lunar excursion module (LEM), the vehicle destined to land on the moon. The LEM contains the necessary scientific equipment and stores for planned activities on the moon surface, and it has the ability to return the crewmen to the CM and service module.

Naturally, with these considerations and the aerodynamic limitations, many possible types of configurations were considered. Winged gliders, and symmetrical and unsymmetrical lifting bodies were studied. The basic configuration chosen for development was the one that was determined to be most practical for the development of the state-of-art at that time. This configuration met the necessary volumetric demands and satisfied the theoretical aerodynamic requirements.

The basic design of the Apollo spacecraft had to be thoroughly evaluated. Each designated module had to be tested to determine if it was capable of functioning properly for its specific mission requirements and if it was capable of functioning relative to the composite spacecraft. It was known that the spacecraft must function over a large angle-of-attack range from Mach number 0 to perhaps as high as Mach number 30. The spacecraft also was to be subjected to extreme variations of temperature and pressure.

One means of evaluating the basic design was through the AWTTP. This program was devised to obtain aerodynamic stability, force, and heating data throughout the atmospheric flight regimes of the Apollo mission and the abort trajectories. New test procedures and new test techniques had to be developed to obtain the necessary design data through the use of existing wind tunnel facilities. The size of the flight article projected a large variation in the anticipated Reynolds number. A test program had to be broad enough to define the desired information for the large range of variables. No one tunnel could handle all of the possible parameters, and it was therefore necessary to use many models and groups of facilities to define the needed design information. The models were to be designed for multifacility use whenever possible. Simulation of wind tunnel conditions and models to actual flight conditions was paramount.

The Apollo development program called for a systematic demonstration and a qualification flight program using Saturn and Little Joe II launch vehicles. Aerodynamic development of the Saturn launch vehicles was the responsibility of the Marshall Space Flight Center.

This paper gives (1) a brief history of the early development of the basic configuration of the Apollo spacecraft, (2) an introduction to the basic vehicle components and the purpose and scope of the AWTTP, (3) the types of testing and the models and facilities, (4) the problem areas (and a brief discussion of their solution) of the design evaluation, and (5) a systematic development of the transition from the basic spacecraft configuration to the production model, resulting from the evaluation of wind tunnel data.

SYMBOLS

The aerodynamic coefficients are referred to both the body and stability systems of axes. The body system of axes is shown in figure 1.

C_A axial-force coefficient, $\frac{\text{axial force}}{qS}$

C_D drag coefficient, $\frac{\text{drag}}{qS}$

C_L lift coefficient, $\frac{\text{lift}}{qS}$

C_l rolling-moment coefficient, $\frac{\text{rolling moment}}{qSd}$

C_m pitching-moment coefficient, $\frac{\text{pitching moment}}{qSd}$

C_{m_α} $\frac{\partial C_m}{\partial \alpha}$, per rad

$C_{m_q} + C_{m_{\dot{\alpha}}}$ damping-moment coefficient

C_N normal-force coefficient, $\frac{\text{normal force}}{qS}$

C_n yawing-moment coefficient, $\frac{\text{yawing moment}}{qSd}$

C_Y side-force coefficient, $\frac{\text{lateral force}}{qS}$

d reference length (maximum diameter of the CM, 154 in.)

I moment of inertia

$\frac{L}{D}$	lift-to-drag ratio
M	free-stream Mach number
q	pitching velocity
q_{∞}	free-stream dynamic pressure
R	Reynolds number (based on maximum diameter of CM)
S	maximum cross-sectional area of CM, $\pi\left(\frac{d}{2}\right)^2$
V	free-stream velocity
$\frac{X}{d}$	longitudinal location of center of gravity from theoretical apex of the CM
X, Y, Z	model axes
$\frac{Z}{d}$	vertical location of center of gravity measured from axis diameter of symmetry
α	angle of attack of model center line, deg
θ	angular displacement
$\dot{\theta}$	angular velocity
$\ddot{\theta}$	angular acceleration
ω	oscillatory frequency

Subscripts:

fs	full-scale vehicle
m	model or test conditions

PURPOSE AND SCOPE OF THE APOLLO WIND TUNNEL TESTING PROGRAM

It was necessary to evaluate thoroughly the basic design of the Apollo spacecraft originating from the theoretical analysis. A broad test range was necessary to cover the extremes in Mach number, Reynolds number, and variations in temperature and pressure that were expected to be encountered under flight conditions. The AWTTP was designed to obtain the necessary aerodynamic data throughout the atmospheric flight regimes of mission and abort trajectories. Through the proper use of applicable data it was possible to evaluate the basic design, to overcome problem areas encountered, and to modify the design to meet mission requirements. The program also provided data for detailed flight planning and flight analysis.

The Apollo spacecraft will employ a low lift-to-drag $\left(\frac{L}{D}\right)$ ratio for flight path control during entry into the earth atmosphere. The lift-to-drag requirement for this entry trajectory control will be provided by center-of-gravity management. Proper center-of-gravity management during the design of the entry module provides a trimmed angle of attack during entry and also provides the associated $\frac{L}{D}$. Aerodynamic input data for entry studies were obtained through the AWTTP.

The need for an abort can occur at almost any time during the mission. Time consideration is of more or less consequence depending on the type or time of abort. Some aborts may be delayed and studied for the most favorable abort time or condition, while those aborts necessary in the event of a launch vehicle in danger of exploding must be made quickly to prevent a catastrophic loss of the spacecraft. The AWTTP incorporated tests that were designed to gather data necessary for the study of atmospheric abort situations involving the LEV from the launch pad through the atmospheric flight. The LEV must have a rocket-forward trim point during its powered and coasting flight. It then must be jettisoned for deployment of the earth landing system. However, the CM proved to have an undesirable apex-forward trim point, and, should an abort occur, and the earth landing system be deployed with the CM in its apex-forward trim position, the possibilities of fouling the parachute and even cutting the parachute support lines would be great. To eliminate this undesirable abort characteristic, it was necessary to turn the CM so that it would descend heat-shield forward for deployment of the earth landing system. It is also necessary to assure that the CM enters the atmosphere in a heat-shield-forward attitude in the event of a high altitude abort to avoid excessive g loads on the crew. Several modifications for reorienting the CM were investigated. The necessary data for the definition of this problem were acquired through wind tunnel testing, as were the data used for the determination of its solution.

Heat transfer data also were programed in the wind tunnel testing. The purpose of these tests was to obtain heat shield design information and to provide input data for empirical calculations. This kind of testing called for specially equipped, thin-skinned models designed to measure rapid temperature changes. Heat sensitive coatings were also used, as well as oil-flow photographs to define flow patterns or distributions. Thermocouples were attached to the inner surface of the model at given points, and temperature-time histories were taken with the angle of attack, Reynolds number, and Mach number as variables. These thermocouples defined the temperature distributions and the stagnation-point heating. Closely related pressure tests defined pressure distribution for the same test conditions. Shadowgraph and schlieren pictures of the flow patterns were useful in defining several necessary parameters, such as shock-standoff distance and boundary-layer flow conditions.

Pressure tests to define the load distribution were incorporated in the AWTTP. The purpose of these tests was to provide data to the structural design, and to appraise the effect of protuberances on the configuration. These data were obtained through the use of pressure models.

There are two kinds of pressure measurements, static and transient (fluctuating). Static pressures are those forces with average pressure during a given length of time on any given point. Transient pressures are those indicated by the magnitude and spectral distribution of broadband, randomly fluctuating pressures on a selected point of the flight vehicle under given flight conditions.

Through the use of two-balance data, component loads were obtained for any given portion of the LEV. For instance, a balance on the rocket and one on the CM gave measurements which permitted a calculable difference showing loads on the escape tower. Two-balance data were also used in apex cover-separation tests. The relationship of the jettisoned apex cover to the CM had to be established to assure clean separation.

The choice of facility for any given test was determined by selecting the one which could most closely approximate flight conditions. Flight conditions were approached through control of the geometric similarity of properly instrumented models and tunnel control of:

1. Reynolds number, that is, the inertia force divided by the viscous force. The mechanical similarity between a model and prototype is realized when the dimensionless Reynolds number for the model equals the Reynolds number for the prototype. Off-nominal conditions require large Reynolds

number ranges. In some instances, it is the design of a particular wind tunnel test to show that the effect of varying the Reynolds number is a negligible factor.

2. Mach number, that is, the ratio of the relative velocity of model and velocity of medium to the speed of sound in the medium.

3. Angle of attack, that is, the attitude of model in relation to the velocity vector.

4. Other parameters such as thrust coefficients, reduced frequencies, and model stiffness distribution that had to be simulated for specific tests.

At times it was not feasible to fill all of the requirements of a particular flight simulation by the testing of just one model in a specific tunnel. Some configurations were expected to encounter a complete 360° angle-of-attack range during mission performance. These configurations could be expected to encounter Mach number ranges from near 0 to as high as Mach number 30 while experiencing wide Reynolds number ranges. Due to this broad range of test conditions, it was necessary, in many tests, to use combinations of facilities and models to obtain data over the wide range of test conditions.

The AWTTP called for many types of testing. One of these was static stability testing. In the area of static stability there were two kinds of testing - thrusting and nonthrusting. The effects of the jet plumes from the launch escape rocket were determined by using thrusting data. Initial studies using solid bodies to represent the predicted shapes of the jet plumes, indicated that an interference on the flow field over the CM existed even though the simulation was somewhat crude. A major design problem was also associated with the simulation of the rocket exhaust. Study in this area disclosed that the decomposition products of concentrated hydrogen peroxide could be so utilized that scaled-thrust values and reasonable simulation of jet interference and exhaust impingement could be obtained in the transonic speed range ($M = 0.5$ to 1.3). A compromised simulation with high-pressure cold air was used in testing at supersonic speeds ($M = 1.5$ to 3.5).

In the consideration of static stability, three-component or pitch-plane data were taken when testing symmetrical bodies. Six-component data were taken when roll, yaw, and side-force coefficients were considered significant.

There are no tests or combination of tests that could account for all variables such as Reynolds number, Mach number, angle of attack, model size, wind tunnel, or balance choice that could be made in such a manner as to provide all desired data for the determination of aerodynamic characteristics. Therefore, testing was pointed to those areas considered most important and practical. With limited data it is possible to predict or estimate aerodynamic characteristics in untested areas.

The development of the launch vehicle configurations was not included in the AWTTP. However, brief tests to define the static stability of the Apollo-Saturn I vehicle were made. The investigation of the buffet response for this launch configuration was also made. Heating ratio and pressure measurements were also conducted on the launch configurations using models of the foreportion of the launch configuration only.

This program also called for extensive studies in the area of dynamic stability. These are the moments that are developed due to the angular velocity of a vehicle as it oscillates about the center of gravity. An indication of the dynamic stability is given by the damping moment coefficient, which may be either stabilizing (when values of $\left(C_{m_q} + C_{m_{\dot{\alpha}}}\right)$ are negative) or destabilizing (when values of $\left(C_{m_q} + C_{m_{\dot{\alpha}}}\right)$ are positive).

Dynamic stability data were obtained through three different techniques:

1. Forced oscillation

2. Limited free oscillation

3. Free-to-tumble

A discussion of the test techniques and data reduction procedures for the forced oscillation test is given in reference 1. A discussion of the apparatus, test procedures, and data reduction for the limited free oscillation test is given in reference 2.

The free-to-tumble method of obtaining data for damping moment investigations was to design the model to oscillate about its center of gravity without restraint. It called for statically balanced models mounted on a transverse rod that passed through the model center of gravity and permitted it to tumble freely. The development of this type of testing presented problems in the mounting of the model on the transverse rod. The desirable situation was to obtain a system with minimum friction and interference. A gas bearing was designed to support the model. Gas bearings were used successfully in some limited free-oscillation tests. However, in some cases the load changes were quite severe and eventually lead to galling of the bearing. Finally, precision ball bearings were used and proved satisfactory after it was determined that the friction damping was negligible compared to the total damping moment.

Tare damping (friction) was factored out at Mach numbers where tare damping was considered to be a proportionate amount of the total moment being measured.

In both the free-oscillation and the limited free-oscillation technique, the input data were acquired for calculating the damping moment coefficient using the oscillatory angle-of-attack time history.

Utilizing the single-degree-of-freedom equation of motion

$$\left[\left(I\ddot{\theta} + \left(C_{m_q} + C_{m_{\dot{\alpha}}}\right)\left(\frac{qSd^2}{2V}\dot{\theta}\right) + C_m \; dS = 0\right)\right]$$

where C_m was available from static tests, and I was measured beforehand, it was found that the only variable is $\left(C_{m_q} + C_{m_{\dot{\alpha}}}\right)$. From this, integrate using the equation of motion with assumed $\left(C_{m_q} + C_{m_{\dot{\alpha}}}\right)$ until the measured θ-time histories are assimilated.

Model simulation of the full-scale vehicle is most important in obtaining dynamic stability data, as in other wind tunnel testing. The dynamic similarity between the full-scale flight vehicle and the wind tunnel model was achieved through the reduced frequency parameter k.

$$k = \left(\frac{\omega d}{V}\right)_m = \left(\frac{\omega d}{V}\right)_{fs}$$

where ω is the frequency of oscillation, d is reference length, V is the velocity, m is the model or tunnel test conditions, and the subscript fs indicates the full-scale vehicle.

$$\left(\frac{\omega d}{V}\right)_m = \left(\frac{\omega d}{V}\right)_{fs} \quad \text{where } \omega = \left(C_{m_\alpha} \frac{q_\infty \, Sd}{I}\right)^{\frac{1}{2}}$$

so that

$$\left[\frac{d}{V}\left(\frac{C_{m_\alpha} \, q_\infty \, Sd}{I}\right)^{\frac{1}{2}}\right]_m = \left[\left(\frac{C_{m_\alpha} \, q_\infty \, Sd}{I}\right)^{\frac{1}{2}} \frac{d}{V}\right]_{fs}$$

If static operating temperatures are the same $\left(T_m = T_{fs}\right)$ then, by definition $V_m = V_{fs}$, reducing the above identity to

$$\left(\frac{q_\infty \, d^5}{I}\right)_m = \left(\frac{q_\infty \, d^5}{I}\right)_{fs}$$

However, if the velocities are not equal, the identity reduces to

$$\left(\frac{\rho d^5}{I}\right)_m = \left(\frac{\rho d^5}{I}\right)_{fs}$$

Therefore at times, the density form of reduced frequency parameter may be the most accurate index of dynamic similarity. The density method was used in pressure tunnel testing, while the inertia variation method was used in atmospheric tunnels where there was no control over density.

TEST FACILITIES

Table I lists the wind tunnel facilities used in the AWTTP, and it also indicates the test section size, Mach number range, and Reynolds number range for each tunnel.

Many facilities were required to simulate the particular pattern of flight conditions that were necessary to evaluate design and provide input aerodynamics for studies leading to modifications and final design of the Apollo spacecraft. These data had to be accumulated over the entire range of flight conditions that were encountered through the launch and trajectories. The selection of a tunnel for any particular test was based on the tunnel capabilities to simulate required conditions, and selection was based also on its convenience and operational economy when there were possible alternate choices.

There were three primary types of wind tunnels used in the AWTTP.

1. Continuous flow tunnels: These tunnels permit uninterrupted testing until all data points are obtained. Models and test conditions of Mach number, Reynolds number, and angle of attack are set up, and the air flow is recirculated for indefinite periods when operating within nominal operating ranges.

2. Intermittent tunnels (blowdown tunnels): These tunnels have an operating time of from several seconds to a few minutes. They have storage tanks charged with pressurized air that is suddenly released, and data are taken over a short blowdown time span.

3. Impulse tunnels: These tunnels are designed for the simulation of high Mach number range. They have a very short run time. Instrumentation of the models for this type of tunnel is set up to enable complete and almost instantaneous recording of necessary data.

The Apollo wind tunnel tests were conducted in 25 different tunnels having Mach number range capabilities from near Mach number 0 to Mach number 20, with Reynolds number capabilities from 0.0001×10^6 to 14×10^6 per foot. Tests were also made in ballistic ranges and free-flight facilities. Photographic data were usually obtained to evaluate the motions of the test model.

Figure 2 illustrates Mach number and Reynolds number ranges expected in a normal launch and reentry trajectory. It can be seen that there are conditions during the launch trajectory where there is no facility capable of expected flight simulation. However, off-nominal conditions that may arise from abort situations can, for the most part, be adequately simulated by the proper choice of model and wind tunnel.

TEST MODELS

Considering the large range of flight simulation that was necessary in gathering data to evaluate the design and the modification of the Apollo spacecraft, it is understandable that a detailed model program had to be developed. Figure 3 illustrates the initial Apollo spacecraft design that was dictated primarily by theoretical estimates of the aerodynamic characteristics. Through testing and modification a final configuration was developed from the initial configuration.

The model scale ranges were from 0.02 to 0.15 of the full-scale counterpart. Model size was determined by the testing parameters and effective tunnel size for attaining flight simulation with minimum tunnel interference. Acceptable wind tunnel models were geometrically-scaled and machined to extremely close tolerances in order to obtain data that were within accuracy limitations. The test models and the facilities in which they were tested are shown in table II. These models were constructed mostly of aluminum and stainless steel. Some plastic and wooden models were used for tests of specific configurations.

The type of data being sought determined the internal composition and symmetry of the models as well as the materials from which they were made. Dynamic models, for instance, were dynamically balanced enabling the gathering of data in atmospheric tunnels for a study of dynamic stability. By control of model inertia, a range of reduced frequency parameter could be tested. Pressure models had internal accommodations for instrumentation of transducers to selected pressure taps. Generally, heat-transfer models were thin-shelled and instrumented with thermocouples designed to record rapid temperature changes. Some of these models were instrumented with thin-film resistance thermometers, thin-wafer calorimeters, and thin-film platinum heat-transfer gages. The structural dynamic (SD) model was flexible with a scaled-stiffness distribution and a variable mass distribution for simulating given conditions. This model was spring-mounted to allow bending in the first and second free-bending modes as well as pitch oscillation about the center of gravity, and was instrumented with bending moment strain gages, accelerometers, and transducers for measuring transient pressures. This model was also equipped with an electromagnetic shaker installed between the sting and the model to excite the model to obtain aerodynamic damping in pitch characteristics.

NARRATION

Theoretical Studies

The basic configuration chosen for development was determined to be the most practical for the development of the state-of-art at that time and for conforming most closely to mission requirements. It was determined that the separable module principle of a spacecraft would be the one developed. The modular concept for a spacecraft enabled a design plan to be initiated that would aid in solving a critical weight problem. Excess weight imposed severe penalties in reaching maximum booster capability. Separable modules are one means of discarding excess weight after a system has completed its respective phase of the mission.

The theoretical studies predicted the anticipated volumetric requirements for mission completion. They also defined the extremes in Mach number range (Mach number 0 to approximately 30) that the spacecraft or component of the spacecraft might be expected to encounter. The studies presented the necessity of design detail to accommodate structural parameters and severe heating problems expected in atmospheric flight regimes.

After establishing a theoretical base from which to evolve the proposed flight article (see fig. 3 for illustration of an early configuration), the AWTTP was developed to accomplish the following:

1. Substantiate or verify design estimates

2. Provide design data

3. Evaluate effects of modifications

4. Provide detailed aerodynamic characteristics for use in studies, mission planning, and flight analysis.

The test program was initiated in early 1962, shortly after the selection of the prime contractor. The program, as previously stated, consisted of many tests and various facilities. An indication of complexity of the program can be seen in table III, which shows a schedule of the wind tunnel tests from 1962 through 1964.

Parametric Studies

Theoretical estimates were made of the aerodynamic characteristics of the basic vehicle. The parametric study, the first of the wind tunnel tests, determined the effects of systematically varying some of the geometric dimensions of the vehicle components. The program was designed to verify estimates and to assure that the original design was proper.

The variables tested on the CM in the parametric study were corner radius, afterbody angle, heat-shield radius, and nose radius (fig. 4). Six-component data were taken on many combinations of these variables using precision-scaled static-force models in prescribed wind tunnels. By applying these data the aerodynamic behavior of the configuration was predictable.

Parametric studies of the LEV were also made. These studies were designed to obtain data demonstrating the effect of variations in the escape tower lengths and rocket lengths. Escape tower lengths were varied to find the optimum length to remove the escape rocket from the immediate vicinity of the CM to minimize the adverse effect of the rocket exhaust on the structure and aerodynamic stability of the vehicle. The LEV was also varied by changing the flared skirt at the rocket base and by modification of the rocket nose in an attempt to improve the vehicle static stability (fig. 5). It was necessary to generate loads data for the LEV for a high angle-of-attack range. It was also necessary to define the static and dynamic stability throughout subsonic, transonic, and low supersonic Mach number ranges. See tables II and IV for information on test facilities, ranges, and models used in the parametric studies of the CM and LEV.

Data from the parametric studies resulted in the selection of the final shape of the CM and gave a good working base from which to develop the production configuration. Continued testing resulted in the acceptance of a final configuration (fig. 6).

Flow Separator Investigation

With a defined configuration selected, the next task was to make it function properly. In an attempt to improve the stability of the LEV, there were extensive wind tunnel tests involving flow separators. Flow separators are collars attached near the base of the escape rocket in the vicinity of the rocket flare (fig. 7). It was thought that by governing the relative size or shape of the flow separator, an optimum flow pattern could be developed that would improve the stability of the LEV. Testing indicated flow separators did add to the stability in the transonic speed range; however, the net effect was not enough to warrant their addition to the configuration. See tables II and V for information on test facilities, ranges, and models used in gathering data that were applicable in determining the effects of flow separators.

Thrusting LEV

The Apollo LEV provides for the immediate removal of the CM from the launch pad in case of pad abort, and the LEV also provides for the removal of crewmen from a malfunctioning booster during launch. The launch escape rocket is mounted ahead of the CM, and experience with a similar Mercury escape system indicated the stability of the Apollo LEV would be reduced by the firing of the escape rocket. Therefore, it was necessary to investigate the effect of escape rocket exhaust plumes on the stability of the Apollo LEV.

Solid bodies, representing the predicted shapes of jet plumes from the launch escape rocket, were used in an early attempt to simulate the effect of rocket exhaust plumes on the Apollo LEV. This method was not a good simulation of the rocket exhaust, since it did not illustrate the effects of jet impingement on the CM that might occur at high angles of attack, nor did it account for the pressure gradient in the mixing region of the jet plumes. The solid body study, however, did indicate a reduction in the static stability and indicated that the jet plume bending, due to the free-stream velocity, could result in impingement on the CM at the higher angles of attack. Further study was necessary to define these effects.

There were two methods of simulation used in obtaining the data for the investigation of jet effects, a hot-jet simulation, in the subsonic and transonic ranges, and a cold-jet simulation in the supersonic Mach number range. There are many variables that were considered in this simulation that enabled an acceptable one. Some of the more important variables are Reynolds number, Mach number ratio, velocity ratio, temperature, density ratio, mass flow, and ratio of specific heats. Of course, several possible combinations of tunnels and propellants could closely duplicate some of these variables if the facilities were so equipped to handle necessary propellants. However, problems such as quantities of propellant, supply pressures needed to acquire given conditions, or problems in handling due to chemical properties, both of the propellant and the exhaust products, made the selection of a tunnel and propellant a most difficult one. It was found that concentrated hydrogen peroxide, upon decomposition, with presence of a catalyst increases its volume many times. The specific heat of the resulting products of decomposition closely match the specific heat of the exhaust products of the escape rocket on the LEV. The Langley 16-Ft Transonic Tunnel is equipped to handle concentrated hydrogen peroxide, and by means of a unique arrangement of injecting the hydrogen peroxide under 2000 pounds pressure through a silver catalyst pack, and by controlling the escape of the decomposition products through contoured nozzles, an acceptable simulation of the LEV under thrusting conditions was attained through subsonic speed ranges to Mach number 1.3.

Problems were encountered in obtaining an acceptable simulation in the supersonic Mach number range as there were no tunnels equipped to handle the specialized system that was required. By knowing the relative effect of alternate variable control (effects of Reynolds number, temperature variations, and other factors), compromises were possible, and they were made in the testing performed in this area. A compromise simulation was made in the Arnold Engineering Development Center, von Karman Gas Dynamics Facility Tunnel A, by using high-pressure cold air escaping through the properly shaped nozzles in testing the Apollo LEV in the Mach number range from 1.5 to 3.5. Those data obtained in cold-jet testing were considered acceptable even though several parameters such as temperature and pressure ratio were not simulated. They were considered acceptable because data resulting from tests run at Mach number 0.7 compared favorably with hot-jet data at the same Mach number. The cold-jet tests did satisfactorily match the jet plume shapes and expansions that indicated that the stability changes are due primarily to shielding. Temperature proved to have the lesser effect on obtaining usable aerodynamic data than did all other variables.

Results from the hot-jet tests indicated that the escape rocket exhausts increased the axial force and decreased the static stability of the Apollo LEV. Associated pressure tests were run and disclosed that jet impingement caused some high local pressures on the lower surface of the CM at higher angles of attack. Corresponding LEV and service module separation studies were made to assure clean separation in the event of an abort. These tests were made both power-on and power-off. See tables II and VI for models, tunnels, and ranges on power-on and solid body tests of the LEV.

Dynamic Stability Tests

Tests were conducted to define the damping parameters for the Apollo configurations. The initial tests were run over a limited angle-of-attack range using a forced-oscillation technique and oscillation amplitudes of $\pm 5^\circ$ or less. Difficulties in properly locating the test models on the model center of gravity were encountered and were due primarily to the incompatibility between geometric shapes of the configurations and the existing test facilities. Attempts to define the effect of testing at centers of gravity other than nominal were inconclusive. The usefulness of the data in flight planning programs is in the simulation of the dynamics of the full-scale vehicle. Test facility capabilities, however, do not permit testing at high oscillation amplitudes matching those of the full-scale vehicle around a given angle of attack. A series of tests were also made using a limited free-oscillation test technique that enabled testing over a higher oscillation range, but allowed testing only about a stable trim point. The damping also had to be stable or limited to a limit cycle oscillation within the amplitude range $\pm 25^\circ$ of the oscillation system. This method was particularly effective in obtaining damping information on the CM in its reentry attitude.

Later in the test program, a free-to-tumble test technique was developed wherein the model was mounted on a transverse rod and allowed to tumble as necessary. The system had limitations imposed by the interferences associated with the transverse rod and supports. Definition of the friction of the system is required. Where possible, the friction was held to a minimum. A gas bearing was used in both the limited free-oscillation and the free-to-tumble tests. Excessive loads led to problems with the gas bearings in the studies of these tests. The use of precision ball bearings was found to be acceptable since the friction damping was found to be a negligible portion of the total damping. Tare corrections were made to account for the bearing friction. The usefulness of the data obtained by the free oscillation test techniques was further limited by the data being obtained as a function of the oscillation amplitude rather than the more useful angle of attack. The development of a technique to convert the damping data as a function of oscillation amplitude to a function of angle of attack made the free-to-tumble technique very attractive. A description of the method is given in reference 3. A summary of the dynamic stability models and tests is given in tables II and VII.

Apollo-Saturn I

Determination of the aerodynamics of the Apollo spacecraft in conjunction with the launch vehicle is necessary for use in mission planning. The development of the Saturn launch vehicles was the prime responsibility of the Marshall Space Flight Center. However, there were some brief tests that were made in support of the program. It was necessary to determine the aerodynamics for the total launch configuration, as well as for the components, for use in studies of normal or abort separation during flight. The necessary data for this study were obtained in a group of wind tunnel tests using the FSL-1 static force model (table II). It was necessary to know exactly what each component of the system would do or where it would go in the case of normal separation or separation in the event of abort during any given sequence. After separation, launch vehicle components could become a menace in the event of contact with the manned spacecraft continuing the mission or going through an abort recovery sequence.

The Apollo-Saturn I launch vehicle was tested with the Apollo spacecraft (fig. 8). Tests were also made for the various flight configurations dictated by both nominal and abort separation during the launch trajectory. Static stability characteristics of the complete Apollo-Saturn launch vehicle, with and without the command module, were determined at subsonic, transonic, and supersonic speeds to Mach number 3.5 in the Ames 14-foot, 9- by 7-foot, and 8- by 7-foot wind tunnels. These configurations were also tested at high Reynolds numbers in the North American Aviation Trisonic Wind Tunnel, and at large angles of attack up to 60°, in the North American Columbus Aeronautical Laboratory Wind Tunnel. Further tests were made in the Arnold Engineering Development Center, von Karman Facility, Wind Tunnels A and B to gather data to determine the static stability characteristics of the Apollo-Saturn I launch vehicle with and without the CM at or near flight Reynolds numbers for Mach number 3.05 to Mach number 8. Also the static stability characteristics of the Apollo second stage configurations (after booster separation) were determined for Mach numbers 6 and 8. These configurations consisted of:

1. The complete S-IV stage forward

2. Same as 1. above, with tower removed

3. Same as 2. above, with CM and service module removed.

The later configurations, Apollo-Saturn I-B and Apollo-Saturn V, were defined by studies conducted by the Marshall Space Flight Center.

Keels, Spoilers, and Strakes

Initial tests determined that the CM had a secondary and undesirable trim point, in the 60° to 70° angle-of-attack region, that had its strongest influence in the subsonic Mach number range. For proper deployment of the earth-landing system drogue parachute, the CM must be oriented heat-shield forward to eliminate the possibilities of fouling the parachute or cutting parachute support lines. Several exploratory investigations were made in an effort to eliminate this apex-forward trim point. It was desirable, if at all possible, that modifications should be "passive," that is, a simple physical addition to or deletion from the external aerodynamic shape of the configuration. Protuberances, called strakes or spoilers, were tried on the CM (fig. 9). Again, the length, size, and shape were varied, numbers were varied, as were their relationship to one another. In some cases keels were extended around the heat-shield corner of the CM (fig. 10).

Further studies were made into the effect of flaps at or near the nose of the CM configuration. The flaps were varied in shape, size, and location (fig. 11). See tables II and VIII for models and facilities used in these studies.

Static and Transient Pressures

The AWTTP included pressure tests to define load distribution. Testing was done to provide data necessary for structural design, and to appraise the effect of protuberances on the configuration. Pressure distributions were also used in making heat transfer evaluations. The static pressure models used in this investigation were the PS models and are listed in table II. Table IX lists the facility and range of tests involving the static pressure models.

It was known that there would be a transient (fluctuating) pressure or noise-pressure level associated with the launch trajectory. This condition is highly dependent on Mach number (near $M = 1$), and the most critical points of investigation are corners, shoulders, and external protuberances. A definition of these loadings was necessary for structural design. The data were also useful in the determination of the buffet response of the vehicle. Tests were made using as large a model as permitted by facility limitations to accommodate the necessary instrumentation and to alleviate the problem of instrumentation selection. The scaled frequencies of the spectrum had to be matched. Tests were run on the PSTL-1 model, and later, on the PSTL-2 model (due to a change in the ramp angle of the adapter housing). These tests were done at the North American Aviation Trisonic Wind Tunnel, the Ames 14-Ft Transonic Wind Tunnels, and the Ames Unitary Plan Wind Tunnel (figs. 12(a) and 12(b)), providing the data necessary for transient pressure investigation. Static tests were also made on the models (tables II and IX).

Structural Dynamics Tests

A structurally scaled model of the Apollo-Saturn I vehicle was designed to investigate further the buffet response of the launch vehicle. The model had a scaled stiffness distribution and a variable mass distribution in order to simulate the proper bending modes. Instrumentation included strain gages, accelerometers, and pressure transducers at critical points. The model was self-excited to determine the buffet response.

Heat Transfer Investigation

The AWTTP incorporated those tests necessary to thoroughly investigate heating phenomena. Tests were run to obtain heat-shield design information and to provide input data for empirical calculations. Specially equipped thin-skinned models were designed to measure rapid temperature changes. Some plastic models were used with heat sensitive coatings to verify the thermocouple measurements. The use of oil flow patterns, shadowgraphs, and schlieren photography helped to define the flow field patterns around the test models. Heat transfer and pressure distribution tests were first made on the configurations without protuberances. Later, these data were compared to those data resulting from testing configurations with protuberances added. As anticipated, holes and protuberances resulted in high local heating areas that had to be defined and accommodated.

Tests of the launch vehicle configurations were made using scaled models of the foreportion (spacecraft) of the vehicle. Thermocouple measurements were made on the tower structure, the CM, the service module, and the spacecraft adaptor. The models used in heat transfer investigations are shown in table II, and the associated wind tunnel testing and the test ranges are shown in table X.

Tower Web Changes

An analysis of the jet plume shapes during the development of hot- and cold-jet testing of the LEV indicated impingement with the launch escape tower with a resulting loss in stability and structural integrity. The application of a suitable heat protection for the affected members was prohibitive because of a weight penalty. As a result of these studies, the leg bracing was modified to provide the production configuration with an hour-glass webbing arrangement that eliminated the effects of jet impingement (fig. 13).

Lift-to-Drag Improvement Modifications

The pitching moment of the vehicle may be varied to provide a trimmed angle of attack, and an associated lift-to-drag ratio $\left(\frac{L}{D}\right)$ results from the trimmed angle-of-attack flight. The basically symmetrical CM was to be trimmed at angle-of-attack by center-of-gravity management. The initial specifications defined a nominal value of 0.5 $\frac{L}{D}$ as a requirement. Studies indicated that center-of-gravity management would not provide this nominal value. In an attempt to establish some simple, passive modification that would supplement the center-of-gravity management method of $\frac{L}{D}$ control, tests were made that involved canting the heat shield, changing the corner radius, and changing other corner modifications of the CM. No modifications were adopted as a result of this study. See figure 14 for $\frac{L}{D}$ improvement modifications tested.

Additional Studies on the Apex-Forward Trim Point Problem

The use of strakes to correct the undesirable secondary apex-forward CM trim point was decided upon. They were expected to become a part of the final configuration; however, they were abandoned because:

1. The strake surfaces were not large enough to turn the vehicle.

2. Protection of the strake surfaces with necessary heat protective material caused an additional weight penalty.

3. Strakes proved to be dynamically unstable.

Solutions to the apex-forward secondary trim point problem were studied after the abandonment of strakes. A destabilizing pitching moment had to be introduced to properly orient the CM in a heat-shield-forward position for deployment of the earth-landing system. Tests were run on the FS-2 in the North American Aviation Trisonic Wind Tunnel to define the effect of an apex cover strake. Data were acquired at a Mach number ranging from 0.4 through 3.5 and an α range of from -15 to 137°. Results of preliminary testing proved the beanie cap strake to be dynamically unstable (fig. 15).

Tower flaps were the next modification investigated. The flap configurations were obtained by adding plates in the tower bracing. The jettison of the escape rocket would expose the plates to the free stream and would provide a destabilizing pitching moment to reorient the CM heat-shield forward. The flaps were also to provide the damping moment necessary to stabilize the configuration. It was found that the tower flap effectiveness was severely reduced at high supersonic speeds due to an unfavorable shock pattern interaction. Dynamic stability tests also indicated that the tower flap configuration was dynamically unstable at both subsonic and supersonic speeds and therefore unacceptable. For information on models used in this study see table II. Test facilities and ranges concerning tower flap investigations are shown in table XI. Figure 16 shows several typical tower flap configurations.

An independent study was made by personnel of the Aerodynamics Branch, Manned Spacecraft Center, Houston, Texas. This study explored the possibility of using deployable canard surfaces near the escape rocket nose in an effort to overcome the secondary apex-forward trim point problem. After burnout of the launch escape rocket the canard surfaces would be deployed to provide the destabilizing pitching moment necessary to reorient the vehicle heat-shield forward. The surfaces would also provide the damping moments required to stabilize the configuration. Preliminary tests indicated that the canard surfaces would provide the pitching moment and the damping moment necessary. Tests of a full-scale launch escape rocket nose with deployable surfaces demonstrated that the canard mechanism would function under flight conditions. Problems, such as the angle of the canard opening and the total area of canard surfaces, were worked out in the preliminary investigations; however the mechanical sophistication was left up to the contractor. This, of course, led to further wind tunnel tests concerning detailed static stability, and dynamic stability (table XII). Models used in this investigation are listed in table II. The input data from wind tunnel testing of canard surfaces were used in developing an operational method for aborts at all altitudes. The sequencing for canard deployment has to be built around requirements necessary for any abort situation, from an on-pad abort to a high altitude abort. Figure 17 illustrates the demonstration model used in the initial study of canard surfaces. Figure 18 shows a sketch of the production LEV with deployed canard surfaces.

Two-Balance Tests

An apex cover is used on the CM to protect the earth-landing system and its associated components. This cover is eventually jettisoned for proper deployment of earth-landing system. Such a design necessarily called for investigation into its particular separation characteristics, since a potential catastrophic damage to the earth-landing system would exist if the cover recontacted the CM after separation. Through use of a two-balance method of measuring loads, the apex cover loads and those loads on the CM were measured at specific separation distances. This enabled a definition of the post-separation flight path of the apex cover and CM. This study was made in wind tunnel testing using the static force models FS-10 and FS-10A (tables I and II) at the North American Aerodynamics Laboratory. The test covered a Mach number range of 0.40 to 0.55 and an α range of 15° to 179° with a Reynolds number range of 6.1×10^6 to 13.7×10^6. The two balance measurements were also used for load distribution for any given portion of the LEV.

Detailed CM Evaluation

Tests were designed for a detailed evaluation of the refined CM configuration. All refinements, such as tower leg wells and various protuberances, had been defined by this time, and the incremental effects of adding these refinements had to be sought, as well as an accurate determination of hypersonic trim lift-to-drag ratio. It was concluded that proper center-of-gravity management could reduce the protuberance roll effects, but that it would require highly accurate rolling moment data. The data obtained in these tests showed the overall effects of the protuberances on trim $\frac{L}{D}$, trim angle of attack, and general effect on the vehicle to be small. This detailed evaluation was run with the FS-12 static force model in the Arnold Engineering Development Center Tunnels B and C. Test parmeters were: a Mach number range from M = 6.0 to M = 10.0, an α range from 145° to 165°, and a Reynolds number range from 0.519×10^6 to 4.25×10^6.

Block II CM

An evaluation followed and determined the reentry static stability characteristics of the block II CM. Block II is the final configuration of the CM returning from the lunar mission. It houses a docking mechanism accommodating the LEM and provides a means of personnel transfer. These tests were made with a modified FS-2 static force model at Ames Unitary Plan Wind Tunnels 11- by 11-foot, 9- by 7-foot, and 8- by 7-foot. Mach number ranges tested were from M = 0.7 to 3.4, α ranges were from 105° to 175°, and Reynolds number ranges were from 3.94×10^6 to 2.68×10^6.

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Manned Spacecraft Center
National Aeronautics and Space Administration
Houston, Texas, July 29, 1966
914-50-10-03-72

TABLE I. - TEST FACILITY CAPABILITIES

Facility	Test-section size	Mach number range	Reynolds number range, $\times 10^{-6}$/ft
Continuous tunnels			
Ames 2-Ft Transonic Wind Tunnel	2 ft^2	0 to 1.4	2 to 8.4
Ames 14-Ft Transonic Wind Tunnel	13.5 ft^2	0.6 to 1.2	2.8 to 4.2
Ames Unitary Plan Wind Tunnel	8 by 7 ft 9 by 7 ft 11 by 11 ft	2.4 to 3.5 1.5 to 2.6 0.7 to 1.4	0.5 to 5 1 to 7 1 to 10
Ames 12-Ft Pressure Tunnel	12-ft diameter	0 to 0.95	0.5 to 9
Arnold Engineering Development Center, von Karman Facility 40-In. Tunnel A	40-in. diameter	1.5 to 6	0.3 to 9
Arnold Engineering Development Center, von Karman Facility 50-In. Tunnel B	50-in. diameter	8	0.25 to 3.3
Arnold Engineering Development Center, von Karman Facility 50-In. Tunnel C	50-in. diameter	10	0.29 to 2.5
Jet Propulsion Laboratory, 20-In. Supersonic Wind Tunnel	18 by 20 in.	1.3 to 5	0.4 to 6
Jet Propulsion Laboratory, 21-In. Hypersonic Wind Tunnel	21 by 15 to 28 in.	5 to 9.5	0.25 to 3.6
Langley Unitary Plan Wind Tunnel	Two 4 by 7 ft	1.47 to 2.86 2.29 to 4.63	0.56 to 7.83
Langley 20-Ft Free-Spinning Tunnel	20-ft diameter	0 to 0.9	0 to 0.62
Langley 12-Ft Low-Speed Tunnel	12-ft octagonal		
Langley 8-Ft Transonic Pressure Tunnel	7.1 ft^2	0.2 to 1.3	0.3 to 4.2
Langley 16-Ft Transonic Dynamics Tunnel	16 ft^2	0 to 1.22	0.04 to 9
Langley 16-Ft Transonic Wind Tunnel	15.5-ft diameter	0.2 to 1.3	1.2 to 3.7
Lewis 8- by 6-Ft Supersonic Wind Tunnel	8 by 6 ft	0.4 to 2.1	2.5 to 5.05
North American Aerodynamics Laboratory, 7- by 10-Ft Low-Speed Wind Tunnel	7 by 10 ft	0.2	1.44
North American Columbus Division, Aerodynamics Laboratory Subsonic Wind Tunnel	7.75 by 11 ft	0.05 to 0.39	2.7
Impulse tunnels			
Arnold Engineering Development Center, von Karman Facility 100-In. Tunnel F	100-in. diameter	9 to 22	0.032 to 0.30
Arnold Engineering Development Center, von Karman Facility 50-In. Hot-Shot II, Tunnel H	50-in. diameter	16 to 21	
Cornell Aeronautical Laboratory, 24- and 48-In. Shock Tunnels	24 and 48 in.	5 to 18	0.03 to 10

TABLE I.- TEST FACILITY CAPABILITIES - Concluded

Facility	Test-section size	Mach number range	Reynolds number range, $\times 10^{-6}$/ft
Cornell Aeronautical Laboratory, 6-Ft Shock Tunnel	72-in. diameter	10 to 30	0.0002 to .05
North American Aviation, 12-In. Shock Tunnel	12-in. diameter	7 to 22	0.0001 to 3
Intermittent tunnels			
North American Aviation, 7- by 7-Ft Trisonic Wind Tunnel	7 ft^2	0.2 to 3.5	5 to 14
North American Aviation Supersonic Aerophysics Laboratory	16 $in.^2$	0.7 and 1.56 to 3.75	

TABLE II. - TEST MODELS AND FACILITIES USED IN WIND TUNNEL PROGRAM

Model (a)	Test facilities	Scale	Model description
FS-1	North American Aviation Supersonic Aerophysics Laboratory 16-In. Wind Tunnel Jet Propulsion Laboratory 21-In. Hypersonic Wind Tunnel Jet Propulsion Laboratory 20-In. Supersonic Wind Tunnel Ames 2- by 2-Ft Transonic Wind Tunnel	0.02	CM with several detachable launch escape system configurations including means for simulation of jet plume from escape motor.
FS-2	Ames Unitary Plan Wind Tunnel 9 by 7 ft 11 by 11 ft 8 by 7 ft North American Aviation 7- by 7-Ft Trisonic Wind Tunnel North American Aviation Aerodynamics Laboratory 7- by 10-Ft Wind Tunnel Ames 12- by 12-Ft Pressure Tunnel	0.105	CM and LEV with several detachable escape tower configurations. Large scale of model provides means of obtaining high Reynolds number data.
FS-3	Arnold Engineering Development Center, von Karman Facility 40-In. Tunnel A Arnold Engineering Development Center, von Karman Facility 50-In. Tunnel B Arnold Engineering Development Center, von Karman Facility 50-In. Tunnel C	0.045	CM and LEV with several detachable escape tower configurations; model designed for high-temperature flow.
FS-4	Arnold Engineering Development Center, von Karman Facility 50-In. Hot-Shot II	0.04	CM of lightweight construction designed for testing in impulse tunnels.
FS-7	Jet Propulsion Laboratory 20-In. Hypersonic Wind Tunnel Jet Propulsion Laboratory 20-In. Supersonic Wind Tunnel	0.02	CM with parametrically varied shapes.
FS-8	Cornell Aeronautical Laboratory 48-In. Shock Tunnel	0.05	CM of lightweight construction.
FS-9	North American Aviation 7- by 7-Ft Trisonic Wind Tunnel	0.105	CM with apex drogue chute cover removed.
FS-10, FS-10A	North American Aviation 7- by 10-Ft Subsonic Wind Tunnel	0.125	CM with and without apex cover.
FS-11	North American Aviation 7- by 7-Ft Trisonic Wind Tunnel	0.15	Forward section of launch escape rocket including canard surfaces.

[a]Force, static.

TABLE II. - TEST MODELS AND FACILITIES USED IN WIND TUNNEL PROGRAM - Continued

Model	Test facilities	Scale	Model description
FD-3[b]	Arnold Engineering Development Center, von Karman Facility 40-In. Tunnel A	0.045	CM and LEV detachable.
	Arnold Engineering Development Center, von Karman Facility 50-In. Tunnel C		
FD-4[b]	Langley 12- by 12-Ft Low-Speed Tunnel	0.10	CM and detachable LEV.
FD-5[b]	Arnold Engineering Development Center, von Karman Facility 40-In. Tunnel A	0.05 0.059	CM and LEV with and without strakes.
	Ames Unitary Plan Wind Tunnel 11 by 11 ft		
FD-6[b]	Ames 12- by 12-Ft Pressure Tunnel	0.10	CM and LEV with strakes on and off.
FD-9[b]	North American Aviation 7- by 7-Ft Trisonic Wind Tunnel	0.059	CM and launch vehicle with and without canards.
FD-9[b]	Lewis 8- by 6-Ft Supersonic Wind Tunnel	0.059	CM and launch vehicle with and without canards.
SD-1[c]	Langley 16- by 16-Ft Transonic Dynamics Wind Tunnel	0.08	Launch configuration, Saturn I; dynamically similar model to determine response to transonic buffeting.
PS-1[d]	Jet Propulsion Laboratory 20-In. Supersonic Wind Tunnel	0.02	CM with detachable LEV. Model is instrumented with pressure tips for obtaining pressure distributions on command with and without the escape tower installed.
	Jet Propulsion Laboratory 21-In. Hypersonic Wind Tunnel		
	Ames 2- by 2-Ft Transonic Wind Tunnel		
PS-3[d]	Arnold Engineering Development Center, von Karman Facility 40-In. Tunnel A	0.045	CM with detachable LEV; model is instrumented with pressure tips for obtaining pressure distributions on the escape tower, CM, service module, and flow separation.
	Arnold Engineering Development Center, von Karman Facility 50-In. Tunnel B		
	Arnold Engineering Development Center, von Karman Facility 50-In. Tunnel C		
PS-3[d]	North American Aviation 7- by 7-Ft Trisonic Wind Tunnel	0.045	CM and detachable launch vehicle.
	Langley 8- by 8-Ft Transonic Pressure Tunnel		
PS-4[d]	Arnold Engineering Development Center, von Karman Facility 50-In. Hot-Shot II	0.04	CM with miniature pressure transducers.

[b]Force, dynamic.
[c]Structure, dynamic.
[d]Pressure, static.

TABLE II. - TEST MODELS AND FACILITIES USED IN WIND TUNNEL PROGRAM - Continued

Model	Test facilities	Scale	Model description
FS-12[a]	Arnold Engineering Development Center, von Karman Facility 50-In. Tunnel B	0.09	CM with protuberances.
	Arnold Engineering Development Center, von Karman Facility 50-In. Tunnel C		
FSC-1[e]	North American Aviation 7- by 7-Ft Trisonic Wind Tunnel	0.10	CM with various parachute modifications; model includes a drag balance.
FDC-1[f]	North American Aviation 7- by 7-Ft Trisonic Wind Tunnel	0.10	Dynamically similar CM with variable drogue chute diameters, porosities, riser lengths, and elasticity.
	Langley 16- by 16-Ft Transonic Dynamics Wind Tunnel		
FSJ-1[g]	Langley 16- by 16-Ft Transonic Wind Tunnel	0.045	LEV using hydrogen peroxide gas generator in some tests.
FSJ-3[g]	Arnold Engineering Development Center, von Karman Facility 40-In. Tunnel A	0.045	LEV using cold-air jet.
FSL-1[h]	Ames Unitary Plan Wind Tunnel 8 by 7 ft 9 by 7 ft 11 by 11 ft	0.02	Saturn C1 Apollo launch configuration; provisions for detaching the escape tower, the CM, and the service module are included to obtain the characteristics of the C-1 booster alone.
	Arnold Engineering Development Center, von Karman Facility 40-In. Tunnel A		
	Arnold Engineering Development Center, von Karman Facility 50-In. Tunnel B		
	North American Aviation 7- by 7-Ft Trisonic Wind Tunnel		
	North American Aviation 7- by 10-Ft Subsonic Wind Tunnel		
FD-1[b]	Jet Propulsion Laboratory 21-In. Hypersonic Wind Tunnel	0.03	CM with center of gravity on center line and with offset center of gravity; models are of lightweight construction and are mounted on air bearings.
	Jet Propulsion Laboratory 20-In. Supersonic Wind Tunnel		
FD-2[b]	Langley Unitary Plan Wind Tunnel 4 by 4 ft	0.055	CM and LEV with detachable escape tower; model is of lightweight and simple construction to permit early testing.
	Langley 8- by 8-Ft Transonic Pressure Tunnel		

[a]Force, static.

[b]Force, dynamic.

[e]Force, static, parachute.

[f]Force, dynamic, parachute.

[g]Force, static, jet effects.

[h]Force, static, launch.

TABLE II. - TEST MODELS AND FACILITIES USED IN WIND TUNNEL PROGRAM - Concluded

Model	Test facilities	Scale	Model description
PS-5[d]	Cornell Aeronautical Laboratory 48-In. Shock Tunnel	0.05	CM with miniature pressure transducers.
PS-6[d]	North American Aviation 12-In. Shock Tunnel	0.018	CM with miniature pressure transducers.
PS-9[d]	Arnold Engineering Development Center, von Karman Facility 50-In. Tunnel C		A 7.4-in. diameter hemisphere model.
PSTL-1[i]	Ames 14- by 14-Ft Transonic Wind Tunnel	0.055	Launch configuration, Saturn I.
	North American Aviation 7- by 7-Ft Trisonic Wind Tunnel		
	Ames Unitary Plan Wind Tunnel 11 by 11 ft 9 by 7 ft 8 by 7 ft		
PSTL-2[i]	Ames Unitary Plan Wind Tunnel 11 by 11 ft 9 by 7 ft	0.055	Launch configuration, Saturn IB and Saturn V.
H-1[j]	Jet Propulsion Laboratory 21-In. Hypersonic Wind Tunnel	0.02	CM and launch (LEV plus service module); service module instrumented with pressure taps.
	Arnold Engineering Development Center, von Karman Facility 50-In. Tunnel C		
H-2[j]	Arnold Engineering Development Center, von Karman Facility 50-In. Tunnel C	0.045	CM and launch (LEV and service module); models are thin skinned and instrumented with thermocouples to obtain heat transfer notes.
	Arnold Engineering Development Center, von Karman Facility 50-In. Tunnel B		
	Langley Unitary Plan Wind Tunnel 4 by 4 ft		
H-4[j]	Cornell Aeronautical Laboratory 48-In. Shock Tunnel	0.050	CM instrumented with thin-film resistance thermometers.
H-6[j]	North American Aviation 12-In. Shock Tunnel	0.018	CM instrumented with thin-film, platinum-resistant heat-transfer gages.
H-7[j]	Arnold Engineering Development Center, von Karman Facility 50-In. Tunnel F	0.040	CM made of stainless steel instrumented with thin wafer calorimeters.
H-9[j]	Arnold Engineering Development Center, von Karman Facility 50-In. Tunnel C		A 7.4-in. diameter hemisphere for study of cutsphere theory.
H-11[j]	Arnold Engineering Development Center, von Karman Facility 50-In. Tunnel C	0.09	CM to define heat transfer with protuberances.
HL-1[k] HL-1B[k]	Langley Unitary Plan Wind Tunnel 4 by 4 ft	0.045	Launch configuration.
	Arnold Engineering Development Center, von Karman Facility 50-In. Tunnel C	0.09	CM to define heat transfer with protuberances.
HL-1C	Langley Unitary Plan Wind Tunnel 4 by 4 ft	0.045	Launch configuration

[d]Pressure, static.

[i]Pressure, static, transient, launch.

[j]Heat transfer.

[k]Heat transfer, launch.

TABLE III. - APOLLO WIND TUNNEL TESTING SCHEDULE [1]

Scale	Model	Feb		Mar				Apr					May				June				July					Aug				Sep				Oct					Nov				Dec			
		19	26	5	12	19	26	2	9	16	23	30	7	14	21	28	4	11	18	25	2	9	16	23	30	6	13	20	27	3	10	17	24	1	8	15	22	29	5	12	19	26	3	10	17	24
	Force models																																													
.02	FS-1[4]	B	B	r[2]	40 (56)			q	B	(24)	B	(16)	B[3]				B	40 (10)	B	B	B	80 (56)	a	40 (37)					40 (38)	40 (54)			r 40 (16)	a	a	a	80 (65)		q	q	30 (32)					
.105	FS-2		80 (54)			d	e	40 (40)	c	40 (23)	A	A	80 (52)							c	40 (38)		40 (35)	z	z	A	40 (42)	A	d	e	c	40 (47)									A	24 (24)				
.045	FS-3					40 (80)	40 (53)															60 (43)	k	k	l	40 (25)		40 (42)	m				m	20 (10)												
.013	FS-6																																						j	j						
.02	FS-7				r	40 (24)	q	40 (16)																																						
.05	FS-8																										p	p	40 (80)																	
.105	FS-9																								z	20 (6)																	80 (83)			
.10	FSC-1[5]																																						z	z	z	z				
.10	FDC-1[6]																																			33	33	33	33	33	33	z	80 (36)	v	80 (55)	
.085	FSJ-1[7]																																			w	w	w	w	w	w	w	160 (232)			
	FDC-2 and FDC-3																																				x	x	x	x	x	x	x	x	x	x
	FBR-1																											40 (31)	40 (8)	i 40 (7)	i	i	i	40 (11)	40 (16)											
																																						10	20	15	40 (45)		40 (20)			
.02	FSL-1[8]																											e	d	c				l	k			A	A	A		c				
.03	FD-1[9]																				40 (32)	r	q	40 (32)																						
.055	FD-2					s	40 (44)						u	40 (39)		s	40 (40)																s	40 (27)			u	40 (20)			s 40 (27)	40 (10) k				
.045	FD-3																																											m	40 (24)	
.143	FD-4																																				t									
.08	SD-1[10]																																								v	v	v	240 (240)		
	Pressure models																																													
.02	PS-1[11]				q	40 (40)		r	r	80 (80)															20 (58) k	20 (58) l	a	a	40 (40)																	
.045	PS-3																													m	20 (8)	m	20 (16)		A	A	80 (75)					m	(8)			
.05	PS-5												32	32	32	32	32																			p	40 (32)									
.019	PS-6[12]												D	D	D	D	D (160)																													
.055	PSTL-1																		A	A	A	120 (108)				g	g	g (144)			e 40 (40)	c 40 (24)		d	d	80 (70)										
	Heat transfer models																																													
.02	H-1[13]									r	r	80 (68)																																		
.045	H-2																	l	40 (40)			m	40 (30)																			m	(8)			
.013	H-3												C																								C	C								
.05	H-4																	32	32	32	32	32															p	40 (32)								
.019	H-6																	D	D	D	D	D (160)																								
.045	HL-1 and HL-1B[14]																																								80	m	40 (24)			
.040	H-7																																							n	(64)	n				

[1]Tests done in 1962.
[2]Scheduled hours are indicated adjacent to test period.
Numbers in parentheses indicate actual charged time.
[3]31 hrs. in 11- by 11-ft unitary, plus 22 hrs. in 9 - by 7-ft unitary.
[4]Force, static.
[5]Force, static, parachute.
[6]Force, dynamic, parachute.
[7]Force, static, jet effects.
[8]Force, static, launch.
[9]Force, dynamic.
[10]Structure, dynamic.
[11]Pressure, static.
[12]Pressure, static, transient, launch.
[13]Heat transfer.
[14]Heat transfer, launch.

NASA-S-66-4899 MAY 20

TABLE III. - APOLLO WIND TUNNEL TESTING SCHEDULE[1] - Continued

Scale	Model	Jan					Feb			Mar			Apr			May		June				July					Aug				Sep		Oct			Nov			Dec		
		31	7	14	21	28	4	11	18	4	11	18	1	8	22	20	27	3	10	17	24	1	8	5	22	29	5	12	19	26	9	16	14	21	28	4	18	25	2	9	16
	Force models																																			63	48		61		
.105	FS-2[3]		e	e	c	d	60 (51)[2]															A	A	80					A	40 (41)						(48) c	(48) d		(60) e		
.045	FS-3			90 (143)	60 (58)	k	k	k	m	20 (18)	160												(45)			(43)		k	8		m	k				k					
.085	FSJ-1[4]					40	40	40		w	w	w				40	40	40							w	w	80	(8)		W	16	24				24					
.045	FSJ-3						(80)				(88)					k	k	k								k	80				(16)	(32)				(35)					
.045	FD-3[5]	k	40 (15)														(22)									(64)															
.100	FDC-1[6]				z	z	40 (60)								40																										(100)
.050	FD-5														k					e																					120
.100	FD-6														(38)					(80)																					f
.100	FD-8																							b																	
	Pressure models												60	20					40	40				(16)																	
.045	PS-3[7]												k	m	(39)				u	u	(160)																				
.050	PS-5												(49)							p	(48)																		96	48	
.055	PSTL-2[8]																			30														72					(60) d	(60) e	
	Ringsail																																	h (80) h							
	Vent tests																										B							B				B			
																											(40)							(40)				(40)			
	Heat transfer models																																								
.020	H-1[9]	m	10 (8)			40							20																												
.045	H-2		10		s	(80)							m																										24		
7.4"D	H-9 (sphere)	m	(8)										(30)									30																	(24)		
.050	H-4				40																	p																	p		
.045	HL-1 and HL-1B[10]			s	(80)																	(96)																			

[1]Tests done in 1963.
[2]Scheduled hours are indicated adjacent to test period.
Numbers in parentheses indicate actual charged time.
[3]Force, static.
[4]Force, static, jet effects.
[5]Force, dynamic.
[6]Force, dynamic, parachute.
[7]Pressure, static.
[8]Pressure, static, transient, launch.
[9]Heat transfer.
[10]Heat transfer, launch.

NASA-S-66-4918 MAY 20

TABLE III. - APOLLO WIND TUNNEL TESTING SCHEDULE[1] - Concluded

Scale	Model	Jan 6	Jan 20	Feb 3	Feb 10	Feb 17	Mar 2	Mar 9	Mar 16	Mar 23	Mar 30	Apr 6	May 18	June 1	June 8	June 15	June 22	June 29	July 6	Aug 10	Aug 17	Sep 14	Sep 21	Sep 28	Oct 5
	Force models										40	80 (40)		60	60	60	(240) 60								
.105	FS-2[3]										f	f		c	d	e	e								
.045	FS-3			m	(19) 24													k	(47) 40						
.125	FS-10[4]					A	20 (51)		A	(44) 40															
.085	FSJ-1								w	w	w		w												
.045	FSJ-3[5]						20 (51)	k	40	40 (72)	40		40 (40)												
.10	FD-6		(56) 80 f																						
	Ringsail no. 2				h	64 (80)																			
.125	FS-10A	(23) 16	A														40	40	(93)						
.15	FS-11																A	A	A	40					(120)
.059	FD-9																			A	A	y	y	y	y
	Heat transfer models																								
.09	H-11[6]						(56) 56										m	(32) 40							
.045	HL-1C[7]						s																		

[1]Tests done in 1964.
[2]Scheduled hours are indicated adjacent to test period. Numbers in parentheses indicate actual charged time
[3]Force, static.
[4]Force, static, jet effects.
[5]Force, dynamic.
[6]Heat transfer.
[7]Heat transfer, launch.

Symbols for Table III

Ames Research Center

a 2-Ft Transonic Wind Tunnel
b 7- by 10-Ft Low-Speed Wind Tunnel
c 8- by 7-Ft Unitary Plan Wind Tunnel
d 9- by 7-Ft Unitary Plan Wind Tunnel
e 11-by 11-Ft Unitary Plan Wind Tunnel
f 12-Ft Pressure Wind Tunnel
g 14-Ft Transonic Wind Tunnel
h 40- by 80-Ft Wind Tunnel
i Prototype Hypersonic Free-Flight Facility
j 6-In. Arc Jet Tunnel

Arnold Engineering Development Center, von Karman Facility

k Tunnel A
l Tunnel B
m Tunnel C
n Tunnel F
o Hot-Shot Tunnel II

Cornell Aeronautical Laboratories

p 48-In. Shock Tunnel

Jet Propulsion Laboratory

q 20-In. Supersonic Wind Tunnel
r 20-In. Hypersonic Wind Tunnel

Langley Research Center

s Unitary Plan Wind Tunnel
t 12-Ft Low-Speed Tunnel
u Transonic Pressure Tunnel
v 16-Ft Transonic Dynamics Tunnel
w 16-Ft Transonic Wind Tunnel
x Spin Tunnel

Lewis Research Center

y 8- by 6-Ft Supersonic Wind Tunnel

North American Aviation

z Aerodynamics Laboratory
A Trisonic Wind Tunnel
B Supersonic Aerophysics Laboratory
C Columbus Aerodynamics Laboratory
D 12-In. Shock Tunnel

TABLE IV.- MODELS, FACILITIES, RANGES - PARAMETRIC INVESTIGATION

Model (a)	Facility	Mach number range	α range, deg	Reynolds number range, $\times 10^{-6}$
FS-7	Jet Propulsion Laboratories 21-In. Hypersonic Wind Tunnel	5.0, 7.3, 9.0	-15 to 195	0.29 to 0.844
	Jet Propulsion Laboratories 20-In. Wind Tunnel	1.48 to 5.01	-25 to 205	0.079 to 1.08
FS-1	North American Aviation Supersonic Aerophysics Laboratory 16-In. Wind Tunnel	1.57, 1.88, 2.48, 2.77, 3.27	LEV -10 to 90. CM -10 to 190	0.539 to 1.16
	Jet Propulsion Laboratories 20-In. Supersonic Wind Tunnel	1.48 to 5.01	-25 to 205	0.079 to 1.078
	North American Aviation Supersonic Aerophysics Laboratory 16-In. Wind Tunnel			
FS-2	Ames Unitary Plan Wind Tunnel 9 by 7 ft 11 by 11 ft 8 by 7 ft	0.7, 0.9, 1.1, 1.2, 1.35, 1.55, 1.7, 2.0, 2.4, 2.6, 3.0, 3.4	CM -15 to 195 LEV -15 to 90	3.4 to 3.6
	North American Aviation 7- by 7-Ft Trisonic Wind Tunnel	0.7 to 3.50	-10 to 51	14.4
	Ames Unitary Plan Wind Tunnel 11 by 11 ft 9 by 7 ft	0.7 to 1.35 1.55 to 2.40	-15 to 35	3.4 to 5.2

[a]Force, static.

TABLE V. - MODELS, FACILITIES, RANGES - FLOW SEPARATOR INVESTIGATION

Model	Facility	Mach number range	α range, deg	Reynolds number range, $\times 10^{-6}$
FS-2[a]	North American Aviation Trisonic Wind Tunnel	0.7, 0.9, 1.2, 1.5, 2.0 and 3.5	-10 to 51	14.4
	Ames Unitary Plan Wind Tunnel	0.7 to 2.4	-15 to 35	3.4 to 5.2
PSTL-1[b]	North American Aviation Trisonic Wind Tunnel	0.7 to 3.5	-4 to 15	5.2 to 6.9
FS-1[a]	North American Aviation Supersonic Aerophysics Laboratory	0.7 to 3.5	-4 to 15	5.2 to 6.9
FS-1[a]	North American Aviation Supersonic Aerophysics Laboratory	0.7 to 3.25	-10 to 10	0.98 to 0.54
FD-2[c]	Langley Research Center Unitary Plan Wind Tunnel	1.6 to 2.75	-16 to 8	0.62 to 3.98
FS-2[a]	North American Aviation, North American Aviation Laboratory	0.18 to 0.26	-10 to 90	2.49 to 1.77
FSL-1[d]	Arnold Engineering Development Center Tunnels A and B	3.5 to 8.0	-4 to 16	0.2 to 2.01
PSTL-1[b]	Ames Unitary Plan Wind Tunnel	0.7 to 3.5	-4 to 15	1.49 to 4.14

[a]Force, static.
[b]Pressure, static, transient, launch.
[c]Force, dynamic.
[d]Force, static, launch.

TABLE V. - MODELS, FACILITIES, RANGES - FLOW SEPARATOR INVESTIGATION - Concluded

Model	Facility	Mach number range	α range, deg	Reynolds number range, $\times 10^{-6}$
PS-3[e]	North American Aviation Trisonic Wind Tunnel	0.40 to 1.36	0 to 60	6.2 to 7.5
FSL-1[d]	North American Aviation, North American Columbus Division Laboratory	0.31	-4 to 60	0.878
FSL-1[d]	North American Aviation Trisonic Wind Tunnel	0.40 to 3.5	-4 to 12	3.54 to 6.4
FD-2[c]	Langley Research Center Unitary Plan Wind Tunnel	1.5 to 2.8	-18 to 2	2.45 to 3.67
FS-2[a]	Langley Research Center Unitary Plan Wind Tunnel	0.7 to 3.4	-3 to 55	3.86 to 2.63
FD-2[c]	Ames Unitary Plan Wind Tunnel	0.3 to 1.2	-12 to 6	1.75 to 3.76
FD-2[c]	Ames Unitary Plan Wind Tunnel	0.5 to 3.4	-15 to 55	7.2 to 3.6
FD-3[c]	Arnold Engineering Development Center Tunnel A	1.5 to 6.0	-15 to 164	1.0 to 6.0
FS-3[a]	Arnold Engineering Development Center Tunnel A	0.5 to 1.11	-4 to 8	None

[a]Force, static.

[c]Force, dynamic.

[d]Force, static, launch.

[e]Pressure, static.

TABLE VI. - MODELS, FACILITIES, RANGES - HOT- AND COLD-JET THRUSTING STUDIES

Model	Facility	Mach number range	α range, deg	Reynolds number range, $\times 10^{-6}$
FS-1[a]	North American Aviation Supersonic Aerophysics Laboratory 16-In. Wind Tunnel	0.70 to 3.25	±10	1.16 to 0.54
FSJ-1[b]	Langley Research Center 16-Ft Transonic Wind Tunnel	0.70 to 1.30	-5 to 31	3.9 to 4.4
	Langley Research Center 16-Ft Transonic Wind Tunnel	0.50 to 1.3	-5 to 61	3.1 to 4.4
	Langley Research Center 16-Ft Transonic Wind Tunnel	0.50 to 1.3	-5 to 31	3.0 to 4.4
FSJ-3[b]	Arnold Engineering Development Center Tunnel A	1.48 to 6.0 0.7 to 4.97 1.48 to 3.99	0 to 50 25 to 80	0.66 to 2.05 1.3 to 1.7

[a]Force, static.

[b]Force, static, jet effects.

TABLE VII. - MODELS, FACILITIES, RANGES - DYNAMIC STABILITY STUDIES

Model (a)	Facility	Mach number range	α range, deg	Reynolds number range, $\times 10^{-6}$
FD-1	Jet Propulsion Laboratories 21-In. Hypersonic Wind Tunnel	5.0 to 9.0	±15 about trim	0.067 to 0.673
	Jet Propulsion Laboratories 20-In. Supersonic Wind Tunnel	2.0 to 3.99	±15 about trim	0.067 to 0.673
FD-2	Langley Research Center Unitary Plan Wind Tunnel	2.4 to 4.65	0 to 25 160 to 180	1.05 to 3.39
	Langley Research Center 8-Ft Transonic Pressure Tunnel	0.30 to 1.20	CM 136 to 154 LEV 16	0.61 to 3.49
	Langley Research Center Unitary Plan Wind Tunnel	1.5 to 2.8	CM 134 to 158 LEV -16 to 8	0.628 to 3.98
	Langley Research Center Unitary Plan Wind Tunnel	1.60 to 2.75	LEV -18 to 2 CM 140 to 162	2.45 to 3.67
	Langley Research Center 8-Ft Transonic Pressure Tunnel	0.30 to 1.20	LEV -12 to ±6 CM 136 to 164	1.75 to 3.76
	Langley Research Center Unitary Plan Wind Tunnel	3.00 to 4.65	-18 to 4	0.91 to 2.63
FD-3	Arnold Engineering Development Center Tunnel A	1.5 to 6.0	-5 to 15 -15 to 164	1.0 to 6.00
	Arnold Engineering Development Center Tunnel C	10.0	-5 to 15 -15 to 164	1.00

[a]Force, dynamic.

TABLE VII. - MODELS, FACILITIES, RANGES - DYNAMIC STABILITY STUDIES - Concluded

Model (a)	Facility	Mach number range	α range, deg	Reynolds number range, $\times 10^{-6}$
FD-4	Langley Research Center 12-Ft Low Speed Tunnel	0.77	CM -30 to 60, tower + CM 0 to 180	0.68
FD-5	Arnold Engineering Development Center Tunnel A	1.5 to 6.0	147 ± 18 0 ± 18	0.31 to 3.56, 0.36 to 5.28
	Ames 11- by 11-Ft Unitary Plan Wind Tunnel	0.7 to 1.4	147 ± 18 60 ± 18 2 ± 18	
FD-6	Ames 12-Ft Wind Tunnel	0.3 to 0.8	Free-to-tumble	2.2 to 7.6
	Ames 12-Ft Wind Tunnel	0.3 to 0.8	Free-to-tumble	1.7 to 7.8
FD-9	North American Aviation Trisonic Wind Tunnel	0.50, 0.70, 0.80	Free-to-tumble	3.14 to 4.86
	Lewis 8- by 6-Ft Supersonic Wind Tunnel	1.59 and 1.98	Free-to-tumble	3.57 to 3.88

[a]Force, dynamic.

TABLE VIII. - MODELS, FACILITIES, RANGES - KEEL, STRAKE, AND CM FLAP INVESTIGATION

Model (a)	Facility	Mach number range	α range, deg	Reynolds number range, $\times 10^{-6}$
FS-1	North American Aviation Supersonic Aerophysics Laboratory 16-In. Wind Tunnel	0.70 to 3.24	-10 to 50	0.98 to 1.024
	Jet Propulsion Laboratories 21-In. Hypersonic Wind Tunnel	7.33	123 to 179	0.84
	Ames 2 by 2 Ft Transonic Wind Tunnel	0.7 to 1.35	-8 to 88	2.12 to 0.93
	Jet Propulsion Laboratories 20-In. Supersonic Wind Tunnel	0.7 to 5.01	50 to 80	1.17 to 0.77
FS-2	North American Aviation 7- by 7-Ft Trisonic Wind Tunnel	0.4 and 0.7	40 to 80	11.1 to 13.2
	Ames Unitary Plan Wind Tunnel 11 by 11 ft 8 by 7 ft 9 by 7 ft	0.5 to 3.4	-15 to 190	7.2 to 3.6
FS-3	Arnold Engineering Development Center, von Karman Facility 40-In. Tunnel A	4.0 to 6.0	75 to 180 -5 to 90	0.52 to 2.45
	Arnold Engineering Development Center, von Karman Facility 50-In. Tunnel C	10.0	75 to 180	0.52 to 2.45

[a]Force, static.

TABLE VIII. - MODELS, FACILITIES, RANGES - KEEL, STRAKE, AND CM FLAP INVESTIGATION - Continued

Model	Facility	Mach number range	α range, deg	Reynolds number range, $\times 10^{-6}$
FD-5[b]	Arnold Engineering Development Center, von Karman Facility 40-In. Tunnel A	1.5 to 6.0	147 ± 18 0 ± 18	0.31 to 3.56 0.36 to 5.28
	Ames 11- by 11-Ft Unitary Plan Wind Tunnel	0.7 to 1.4	147 ± 18 60 ± 18 2 ± 18	
PS-3[c]	Langley Research Center 8-Ft Transonic Pressure Tunnel	0.4 to 1.2	0 to 180 0 to 120	1.3 to 2.4
FS-1[a]	North American Aviation Supersonic Aerophysics Laboratory 16-In. Wind Tunnel	0.70 to 3.24	-10 to 50	0.98 to 1.024
FS-1[a]	Ames 2- by 2-Ft Transonic Wind Tunnel	0.7 to 1.35	-8 to 88	2.12 to 0.93
	Jet Propulsion Laboratories 20-In. Supersonic Wind Tunnel	0.7 to 5.01	50 to 80	1.17 to 0.77
FS-2[a]	Ames 8- by 7-Ft Unitary Plan Wind Tunnel	2.6 to 3.4	75 to 180	3.4 to 3.6
	North American Aviation 7- by 7-Ft Trisonic Wind Tunnel	0.4 and 0.7	40 to 80	11.1 to 13.2

[a]Force, static.
[b]Force, dynamic.
[c]Pressure, static.

TABLE VIII. - MODELS, FACILITIES, RANGES - KEEL, STRAKE, AND CM FLAP INVESTIGATION - Concluded

Model (a)	Facility	Mach number range	α range, deg	Reynolds number range, $\times 10^{-6}$
FS-2	Ames Unitary Plan Wind Tunnel 11 by 11 ft 8 by 7 ft 9 by 7 ft	0.5 to 3.4	-15 to 190	7.2 to 3.6
FS-3	Arnold Engineering Development Center, von Karman Facility Tunnel A	4.0 to 6.0	75 to 180 -5 to 90	0.53 to 2.45
	Arnold Engineering Development Center, von Karman Facility Tunnel C	10.0	75 to 180	0.53 to 2.45

[a] Force, static.

TABLE IX. - MODELS, FACILITIES, RANGES - STATIC PRESSURE INVESTIGATION

Model (a)	Facility	Mach number range	α range, deg	Reynolds number range, $\times 10^{-6}$
PS-1	Jet Propulsion Laboratories 20-In. Supersonic Wind Tunnel	1.5 to 5.0	0 to 180 -1 to 90	0.75 to 1.71
	Jet Propulsion Laboratories 21-In. Hypersonic Wind Tunnel	5.06 to 9.08	0 to 180 -1 to 90	0.059 to 0.807
	Ames 2- by 2-Ft Pressure Tunnel	0.4 to 1.34	-1 to 180	0.077
PS-3	Arnold Engineering Development Center Tunnel A	1.5 to 5.0	0 to 180 0 to 60	0.46 to 4.5
	Arnold Engineering Development Center Tunnel B	8.0		
	Arnold Engineering Development Center Tunnel C	10.0	0 to 180	0.24 to 1.1
	North American Aviation 7- by 7-Ft Trisonic Wind Tunnel	0.40 to 1.36	0 to 140	6.2 to 7.5
	Arnold Engineering Development Center Tunnel C	10.0	0 to 180	0.24 to 1.1
	Arnold Engineering Development Center Tunnel A	1.5 to 3.0	0 to 180	0.58 to 3.12
	Arnold Engineering Development Center Tunnel C	10.0	0 to 120	
	Langley Research Center 8-Ft Transonic Pressure Tunnel	0.4 to 1.2	0 to 180 0 to 120	1.3 to 2.4

[a]Pressure, static.

TABLE IX. - MODELS, FACILITIES, RANGES - STATIC PRESSURE INVESTIGATION - Concluded

Model	Facility	Mach number range	α range, deg	Reynolds number range, $\times 10^{-6}$
PS-4[a]	Arnold Engineering Development Center Hot Shot II	19.5	120 to 180	0.084
PS-5[a]	Cornell Aeronautical Laboratory 48-In. Shock Tunnel	12.0 to 17.3	0 to 50	0.03 to 7.5
	Cornell Aeronautical Laboratory 48-In. Shock Tunnel	6 to 16	0 to 33	0.03 to 7.6
PS-6[a]	North American Aviation 12-In. Shock Tunnel	15.5 to 18.3	140 to 180	0.39 to 0.64
PS-9[a]	Arnold Engineering Development Center Tunnel C	10.0	180	0.616 to 1.48
PSTL-1[b]	North American Aviation 7- by 7-Ft Trisonic Wind Tunnel	0.70 to 3.5	-4 to 15	5.2 to 6.9
	Ames 14- by 14-Ft Transonic Wind Tunnel	0.5 to 1.11	-4 to 8	
	Ames Unitary Plan Wind Tunnel 11 by 11 ft 9 by 7 ft 8 by 7 ft	0.7 to 3.5	-4 to 15	1.49 to 4.14
PSTL-2[b]	Ames Unitary Plan Wind Tunnel 11 by 11 ft 9 by 7 ft	0.5 to 2.4	-4 to 15	2.38 to 7.20

[a]Pressure, static.

[b]Pressure, static, transonic, launch.

TABLE X. - MODELS, FACILITIES, RANGES - HEAT TRANSFER INVESTIGATION

Model (a)	Facility	Mach number range	α range, deg	Reynolds number range, $\times 10^{-6}$
H-1	Jet Propulsion Laboratories 21-In. Hypersonic Wind Tunnel	6.07 to 9.04	CM 130 to 180 LEV 0 to 20	0.6 to 0.8
	Arnold Engineering Development Center Tunnel C	10.0	180	0.616 to 1.48
H-2	Arnold Engineering Development Center Tunnel B	8.0	CM 130 to 180 LEV 0	0.19 to 2.08
	Arnold Engineering Development Center Tunnel C	10.0	CM 130 to 180 LEV 0 to 20	0.19 to 2.08
	Arnold Engineering Development Center Tunnel C	10.0	130 to 180	0.19 to 1.39
	Langley Research Center Unitary Plan Wind Tunnel	2.5 to 3.71	0 to 180	2.08 to 3.66
	Arnold Engineering Development Center Tunnel C	10.0	130 to 180	0.19 to 1.39
H-4	Cornell Aeronautical Laboratory 48-In. Shock Tunnel	6.13 to 17.3	0 to 50	0.03 to 7.5
	Cornell Aeronautical Laboratory 48-In. Shock Tunnel	6 to 16	0 to 33	0.03 to 7.6
H-6	North American Aviation 12-In. Shock Tunnel	15.5 to 18.3	140 to 180	0.39 to 0.64

[a]Heat transfer.

TABLE X. - MODELS, FACILITIES, RANGES - HEAT TRANSFER INVESTIGATION - Concluded

Model	Facility	Mach number range	α range, deg	Reynolds number range, $\times 10^{-6}$
H-7[a]	Arnold Engineering Development Center Tunnel F	18.95 to 20.20	0 to 50	0.06 to 0.07
H-9[a]	Arnold Engineering Development Center Tunnel C	10.0	180	0.616 to 1.48
H-11[a]	Arnold Engineering Development Center Tunnel C	10.0	143 to 180	0.33 to 2.2
HL-1,[b] HL-1B[b]	Langley Research Center Unitary Plan Wind Tunnel	2.5 to 3.71	0 to 180	2.08 to 3.66
HL-1C[b]	Langley Research Center Unitary Plan Wind Tunnel	2.98 to 4.44	±5	1.16 to 3.5

[a]Heat transfer.

[b]Heat transfer, launch.

TABLE XI. - MODELS, FACILITIES, RANGES - TOWER FLAP INVESTIGATION

Model	Facility	Mach number range	α range, deg	Reynolds number range, $\times 10^{-6}$
FS-2[a]	North American Aviation 7- by 7-Ft Trisonic Wind Tunnel	0.4 to 3.5	-15 to 137	5.0 to 18.75
	North American Aviation 7- by 7-Ft Trisonic Wind Tunnel	0.4, 1.2, 3.5	-2 to 157	11.4 to 16.6
FS-3[a]	Arnold Engineering Development Center Tunnel A	3 to 65	0 to 360	0.19 to 2.63
	Arnold Engineering Development Center Tunnel C	1.0	±118	0.19 to 2.63
FD-6[b]	Ames 12-Ft Pressure Tunnel	0.3 to 0.8	Free-to-tumble	2.2 to 7.6
	Ames 12-Ft Pressure Tunnel	0.3 to 0.8	Free-to-tumble	1.7 to 7.8

[a]Force, static.
[b]Force, dynamic.

TABLE XII. - MODELS, FACILITIES, RANGES - CANARD INVESTIGATION

Model	Facility	Mach number range	α range, deg	Reynolds number range, $\times 10^{-6}$
FS-1[a]	Jet Propulsion Laboratories 20-In. Supersonic Wind Tunnel	1.3 to 5.6	Free-to-tumble -50 to 135	1.2 to 2.7
	Jet Propulsion Laboratories 21-In. Hypersonic Wind Tunnel	6 and 9	4.1 to 11	-10 to 50
FS-2[a]	Ames 12-Ft Pressure Tunnel	0.25 and 0.50	10 to 320	1.2 to 3.6
FS-3[a]	Arnold Engineering Development Center, von Karman Facility 40-In. Tunnel A	1.5 to 6.0	0 to 360	1.73 to 4.37
FD-9[b]	North American Aviation 7- by 7-Ft Trisonic Wind Tunnel	0.50, 0.70, and 0.80	Free-to-tumble	3.57 to 3.88
	Lewis 8- by 6-Ft Supersonic Wind Tunnel	1.59 and 1.98	Free-to-tumble	3.47 to 3.88

[a]Force, static.

[b]Force, dynamic.

NASA-S-66-9524 OCT 6

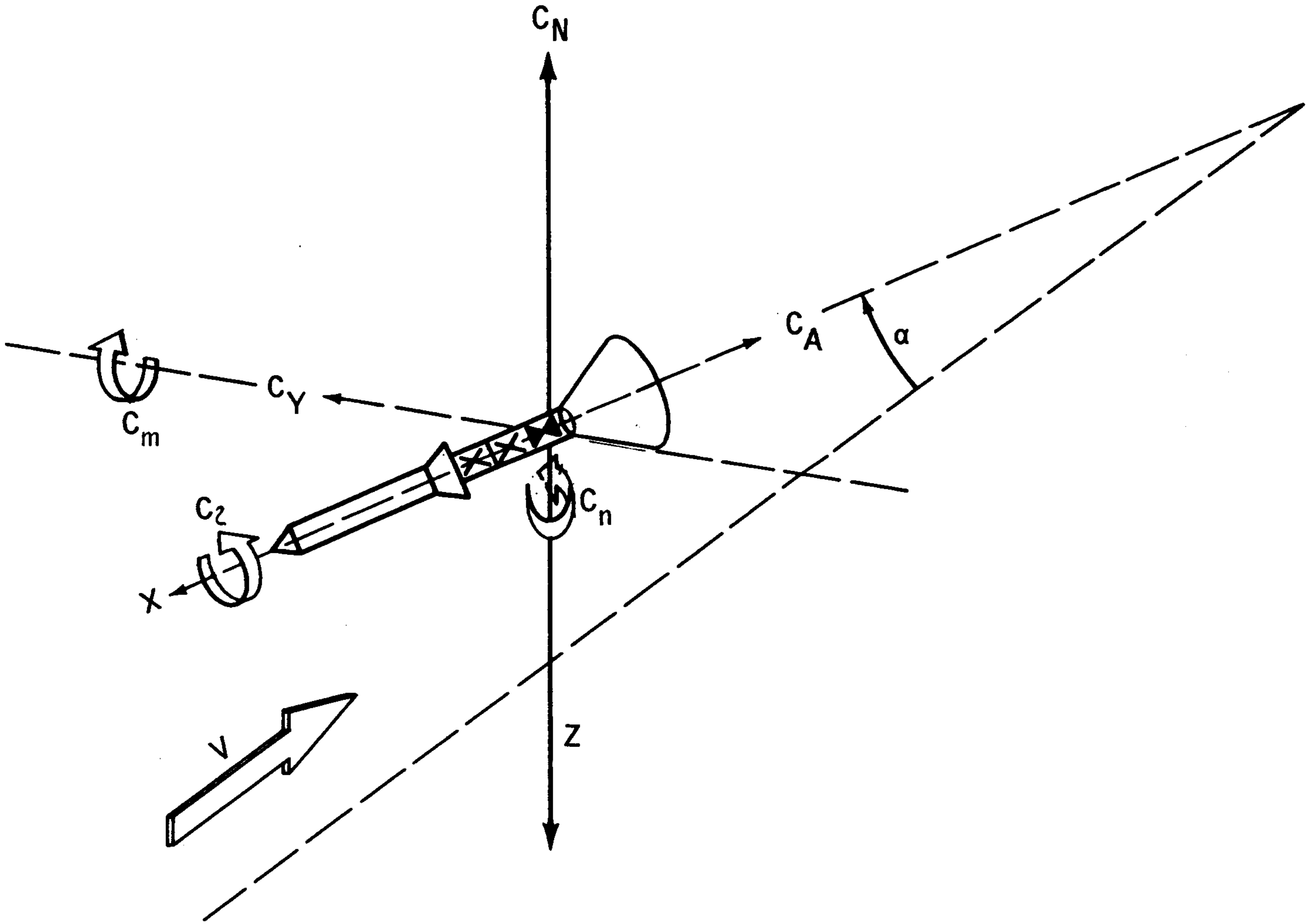

Figure 1. - Sketch of the Apollo LEV showing the body system of axes.

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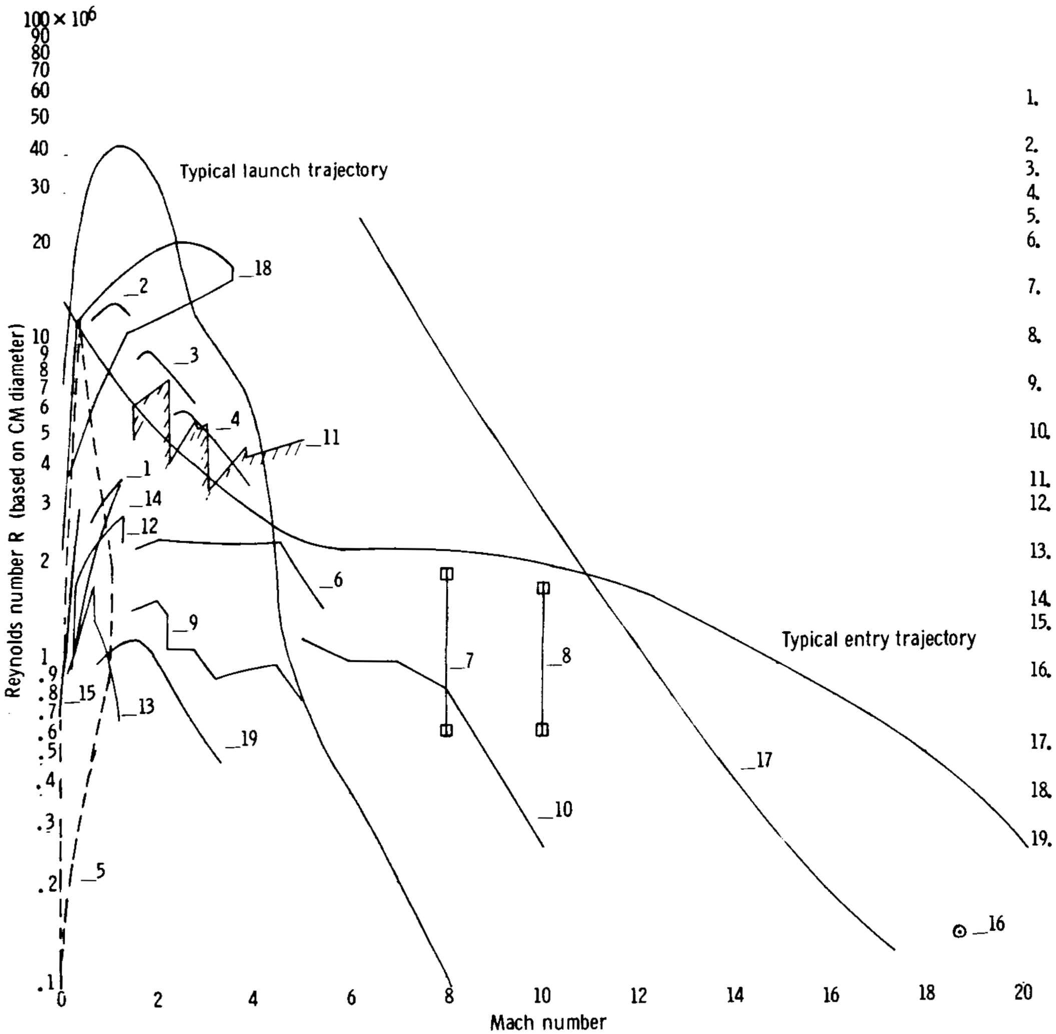
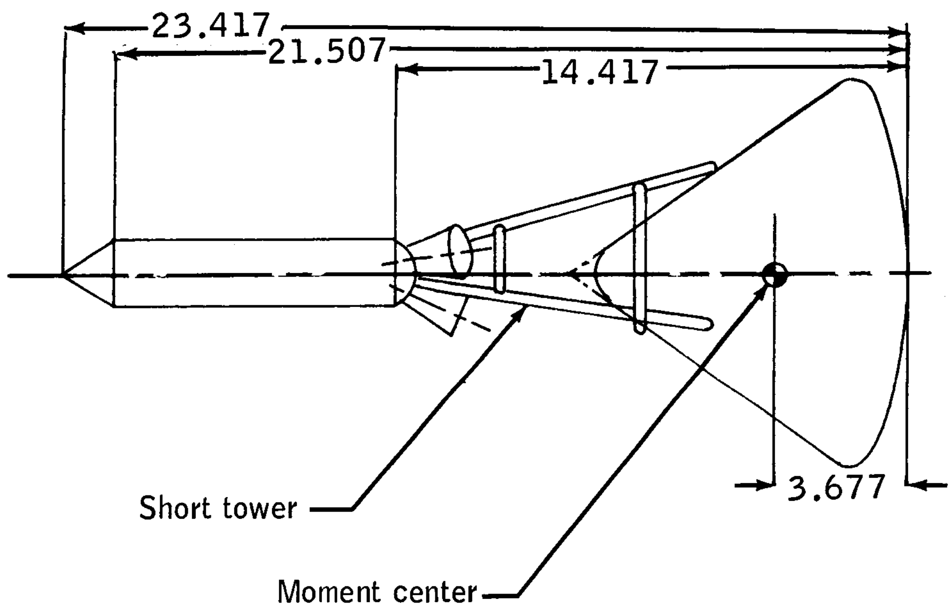


Figure 2. - Test facilities capabilities.

1. Ames 14-Ft Transonic Wind Tunnel (0.055 scale)
   Ames Unitary Plan Wind Tunnels (0.105 scale)
2. 11 by 11 ft
3. 9 by 7 ft
4. 8 by 7 ft
5. Ames 12-Ft Pressure Tunnel (0.10 scale)
6. Arnold Engineering Development Center, von Karman Facility 40-In. Tunnel A (0.045 scale)
7. Arnold Engineering Development Center, von Karman Facility 50-In. Tunnel B (0.045 scale)
8. Arnold Engineering Development Center, von Karman Facility 50-In. Tunnel C (0.045 scale)
9. Jet Propulsion Laboratory 20-In. Supersonic Wind Tunnel (0.045 scale)
10. Jet Propulsion Laboratory 21-In. Hypersonic Wind Tunnel (0.02 scale)
11. Langley Un..ary Plan Wind Tunnel (0.055 scale)
12. Langley 8-Ft Transonic Pressure Tunnel (0.055 scale)
13. Langley 16-Ft Transonic Dynamics Tunnel (0.08 scale)
14. Langley 16-Ft Transonic Wind Tunnel (0.085 scale)
15. North American Aerodynamics Laboratory 7- by 10-Ft Low-Speed Wind Tunnel (0.105 scale)
16. Arnold Engineering Development Center, von Karman Facility 50-In. Hot-Shot II, Tunnel H (0.04 scale)
17. Cornell Aeronautical Laboratory 48-In. Shock Tunnel (0.05 scale)
18. North American Aviation 7- by 7-Ft Trisonic Wind Tunnel (0.105 scale)
19. North American Aviation Supersonic Aerophysics Laboratory (0.02 scale)

NASA-S-66-4901 MAY 20



Drawing not to scale

Dimensions in feet

Figure 3. - Basic design of the Apollo configuration.

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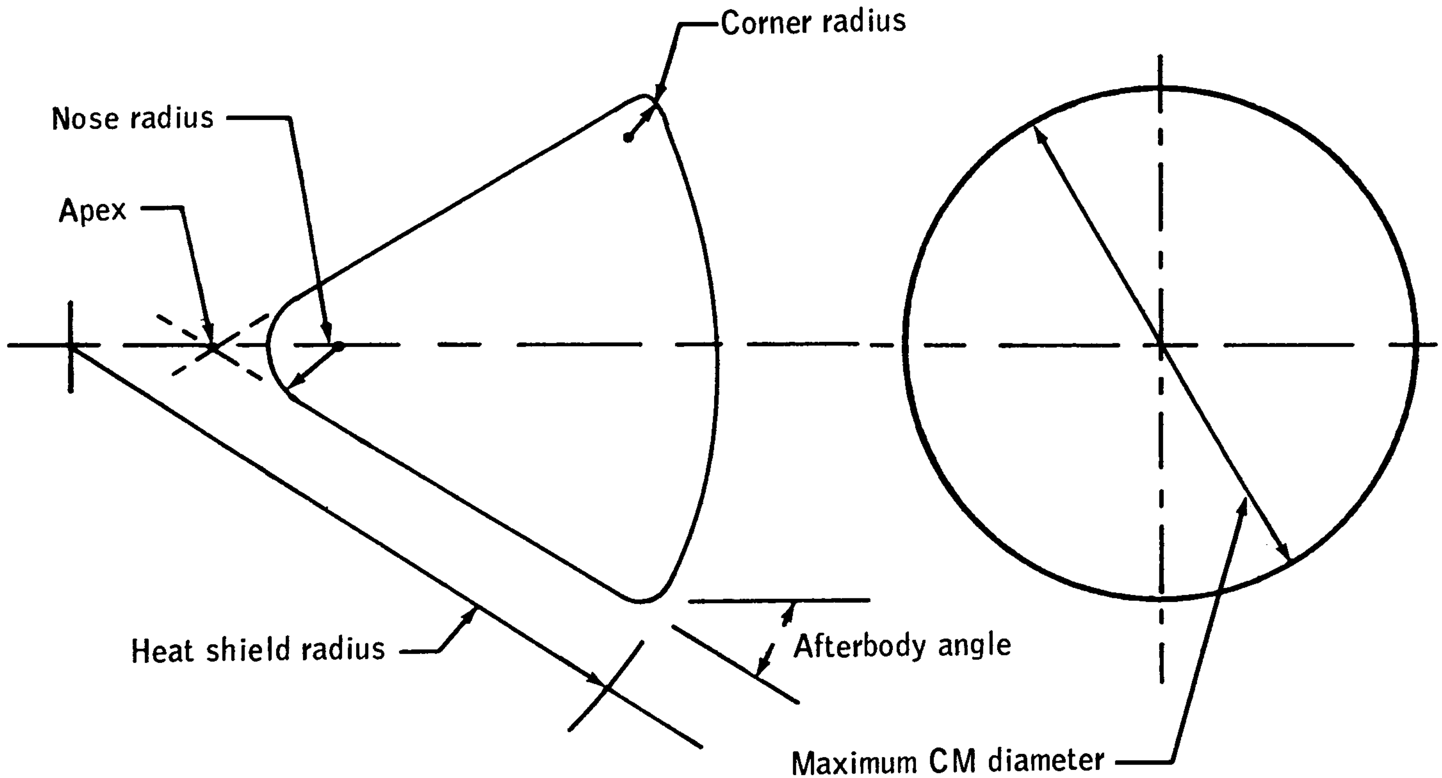
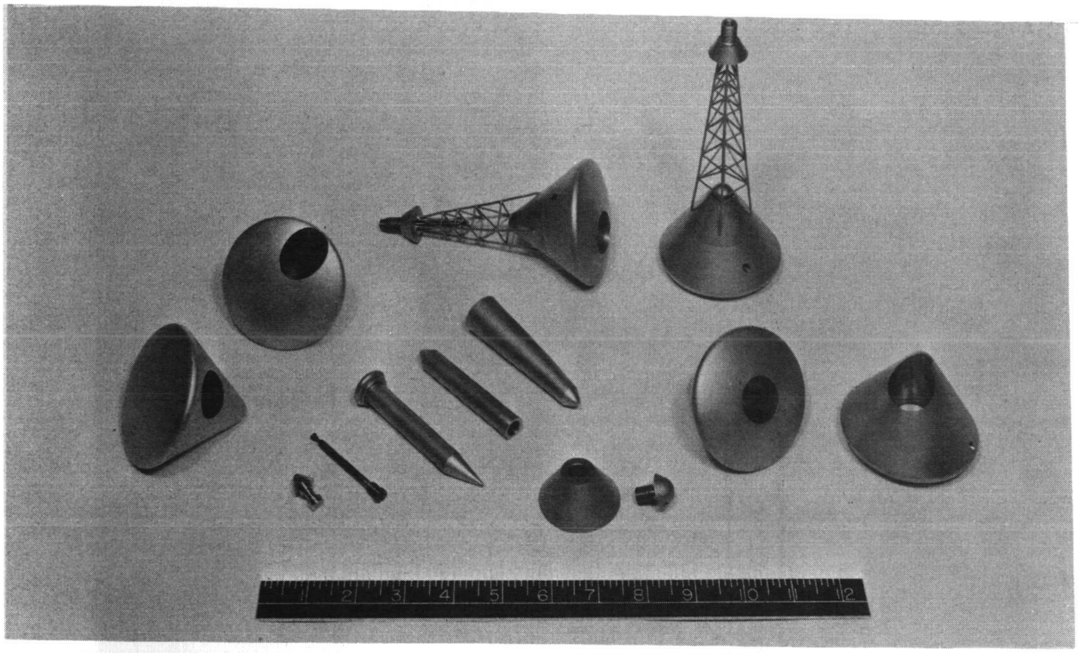


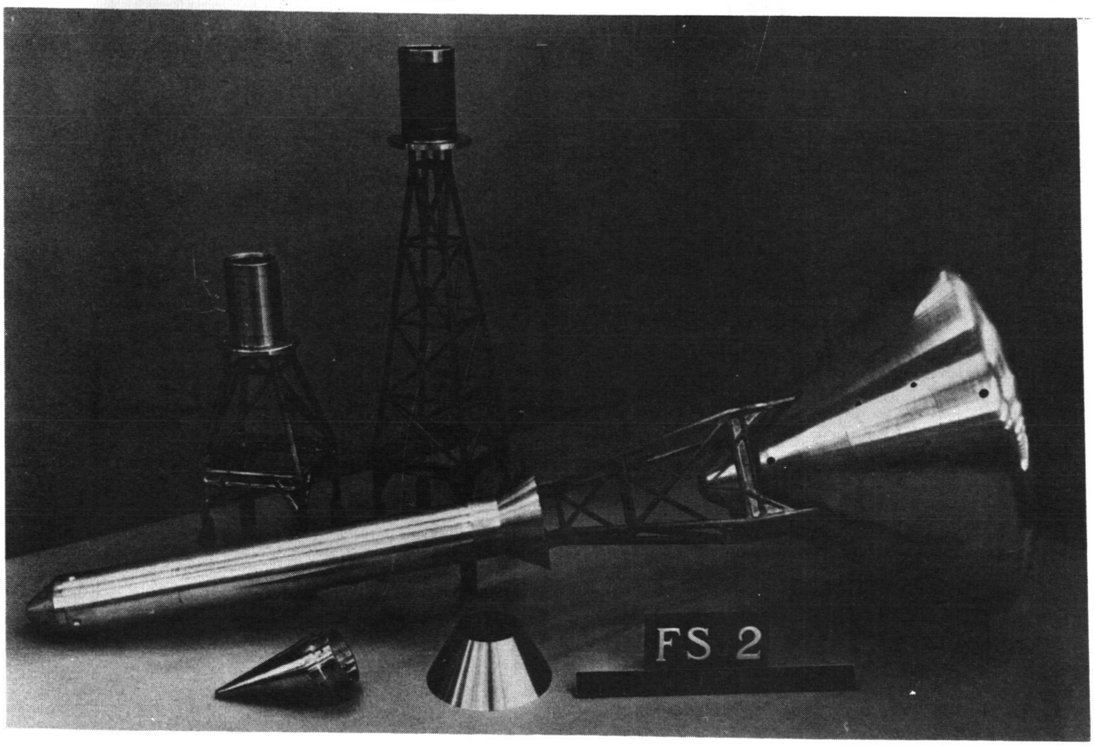
Drawing not to scale

Figure 4. - Sketch of the basic Apollo CM with controlling dimensions.

NASA-S-66-9523 OCT 6



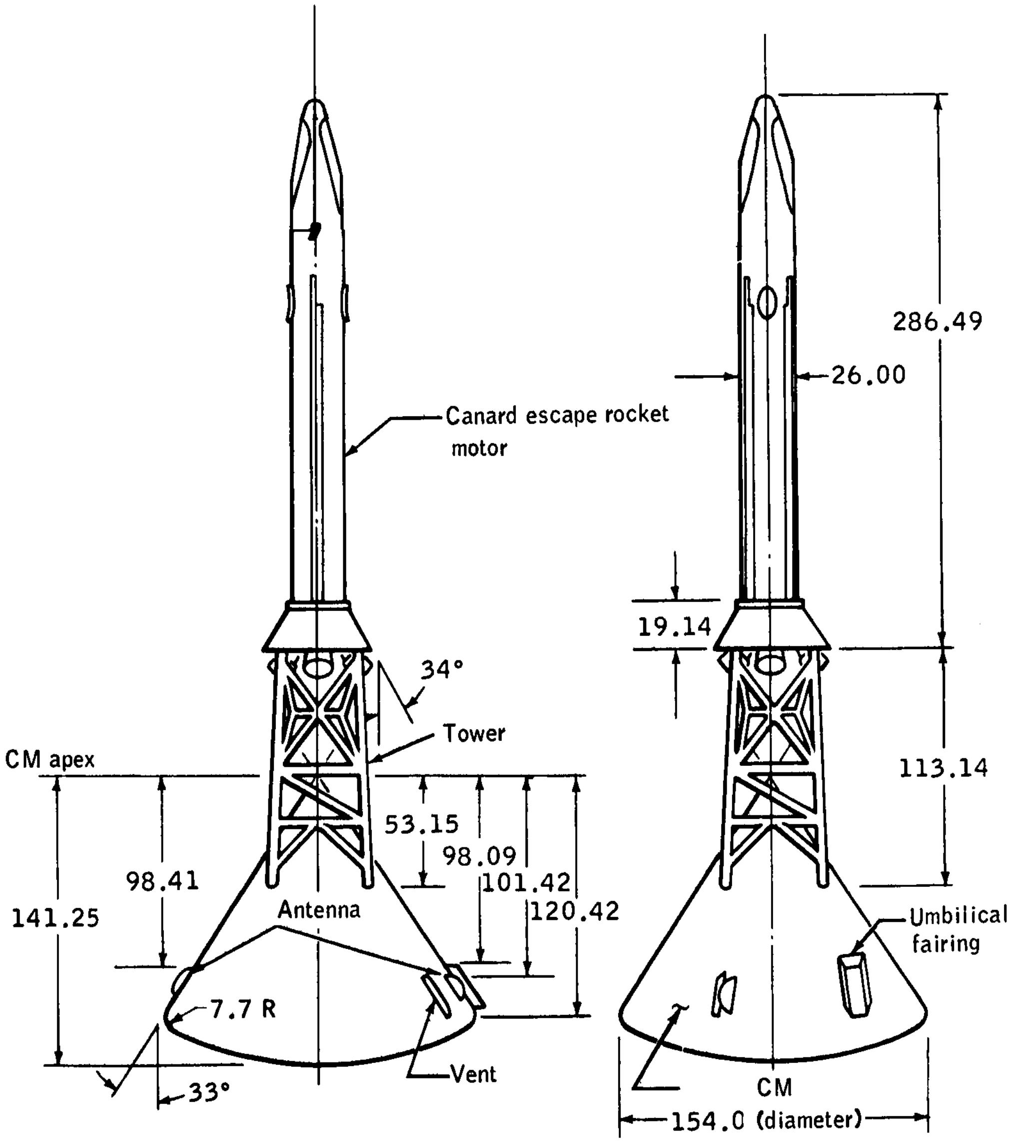
(a) FS-1 model demonstrating some typical modifications tested.



(b) FS-2 model demonstrating some typical modifications tested.

Figure 5. - Photographs of some Apollo wind tunnel models.

NASA-S-66-4906 MAY 20



Full-scale dimensions in inches

Drawing not to scale

Figure 6. - Sketch of an Apollo LEV.

NASA-S-66-4907 MAY 20

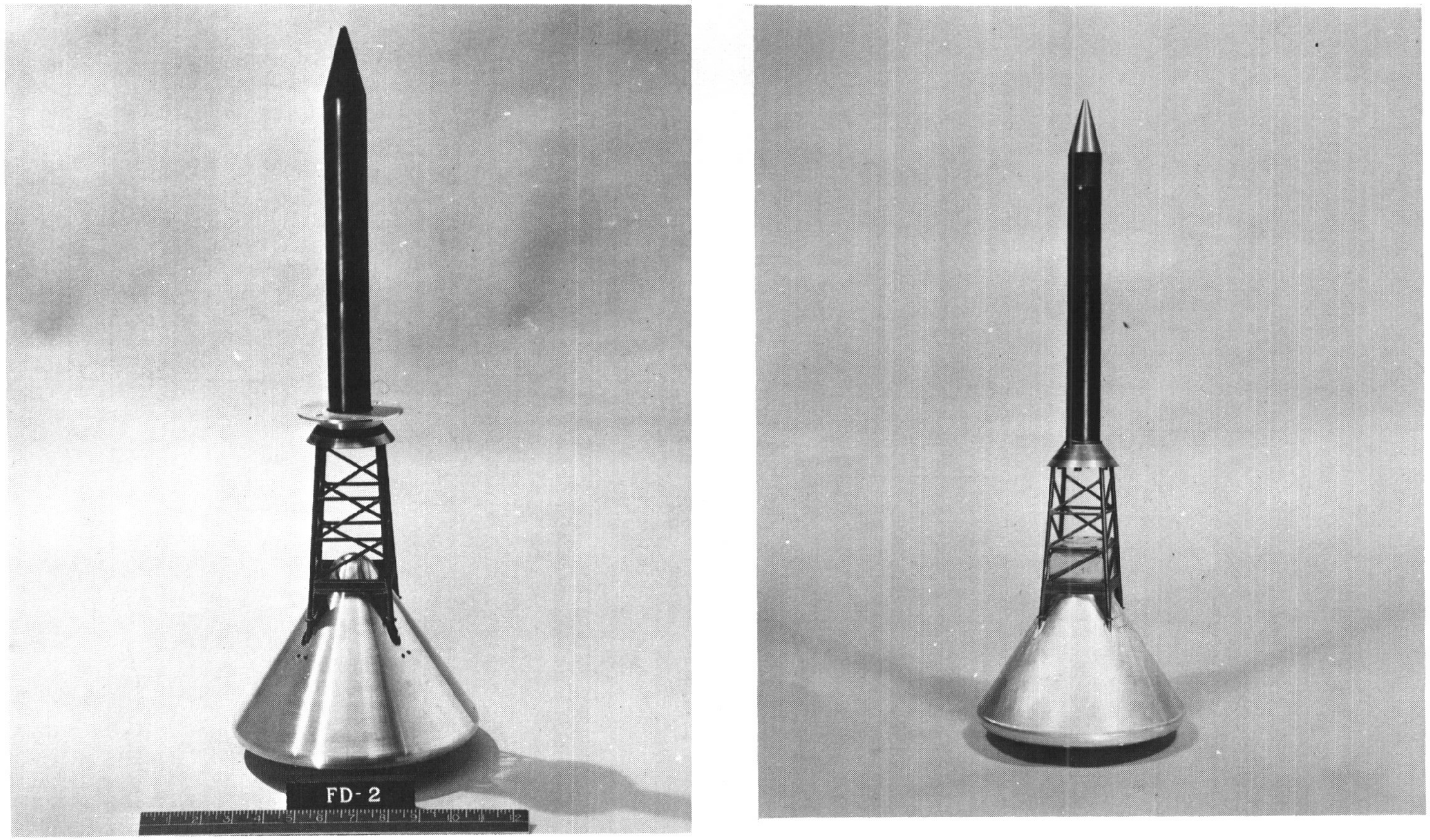


Figure 7. - Photographs of the LEV (FD-2 model) with and without flow separator.

NASA-S-66-4908 MAY 20

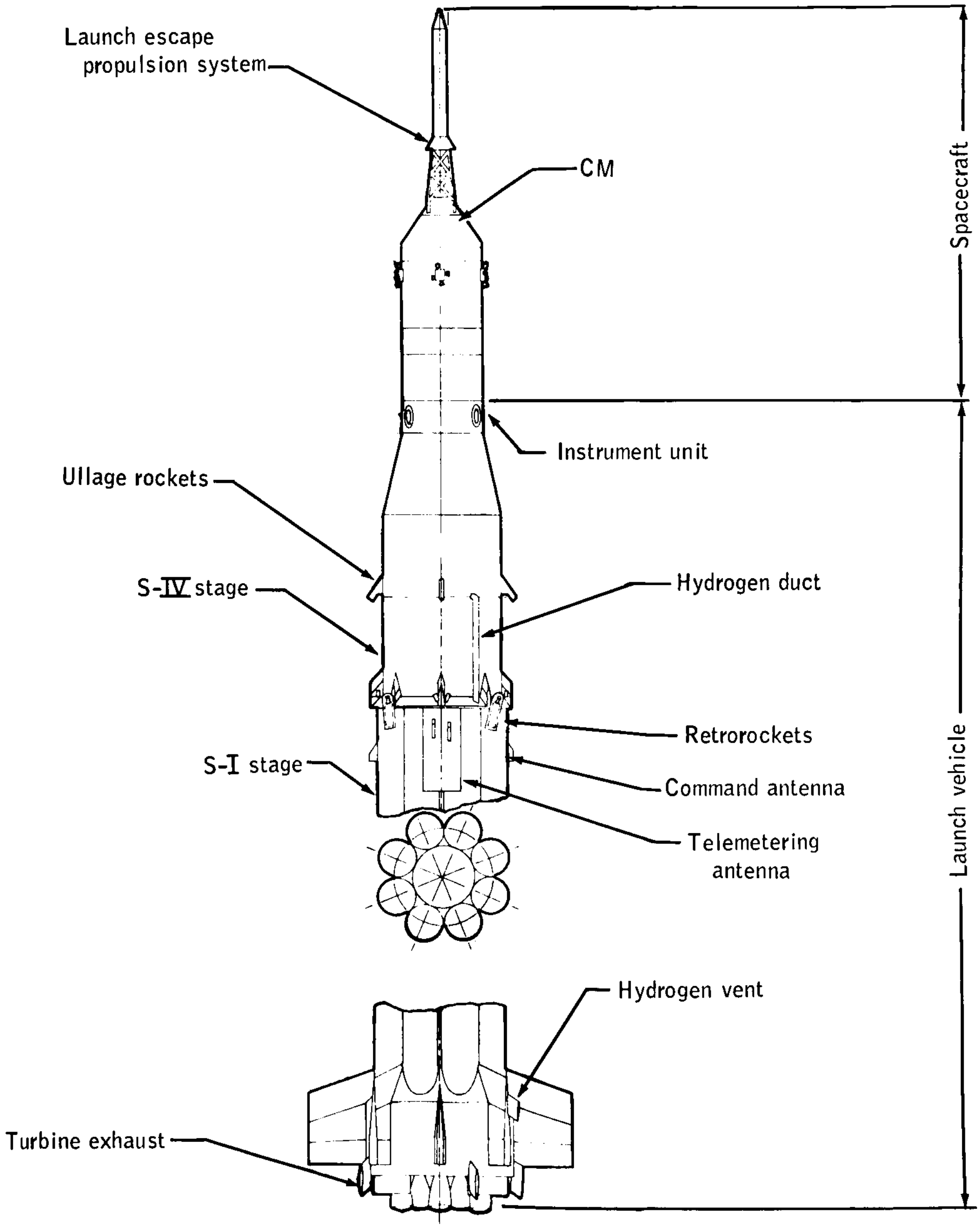
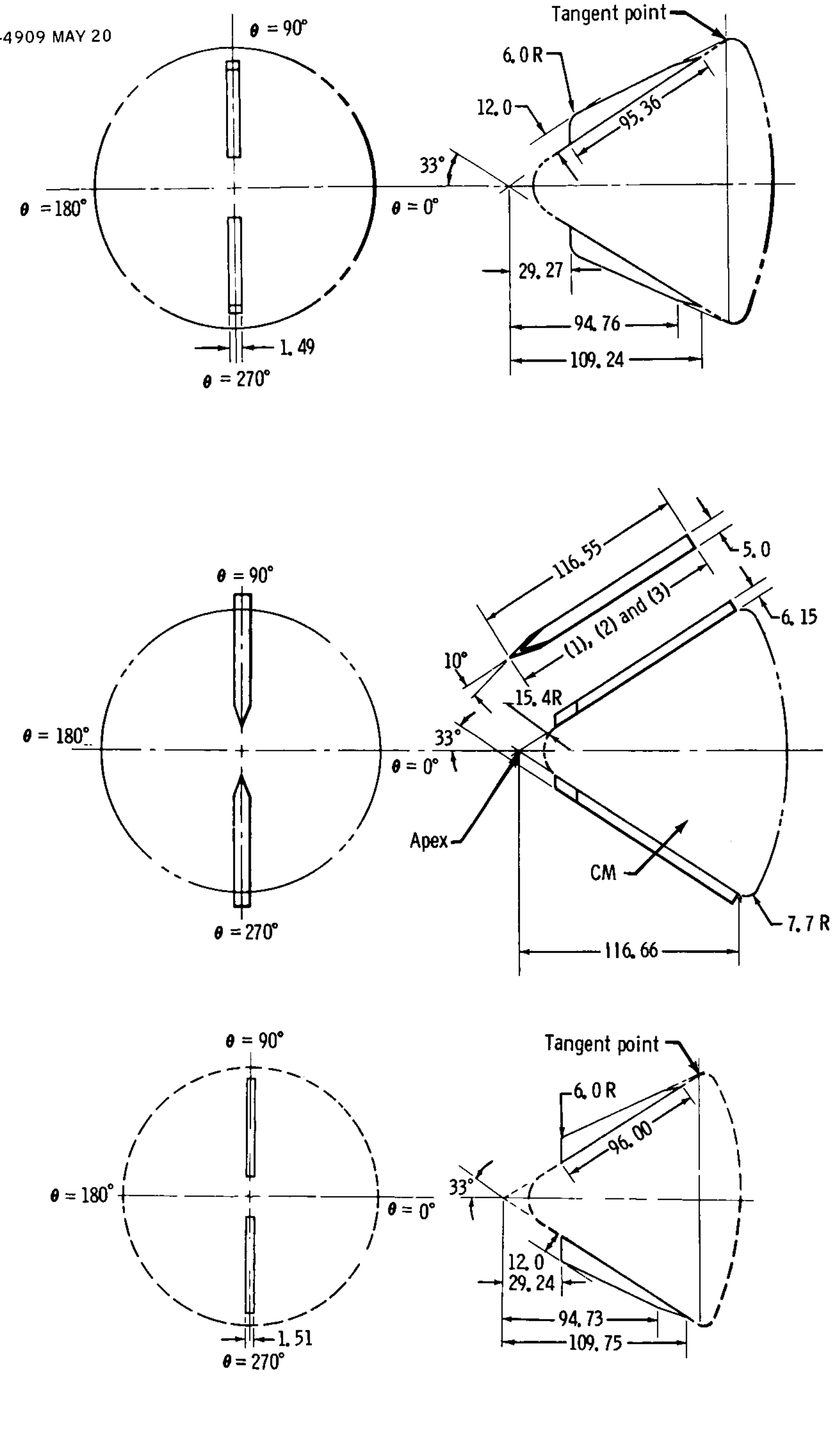


Figure 8. - Sketch of the basic configuration mounted on Saturn I booster.

NASA-S-66-4909 MAY 20



Full-scale dimensions in inches

Drawing not to scale

Figure 9. - Three spoiler or strake configurations tested in the AWTTP.

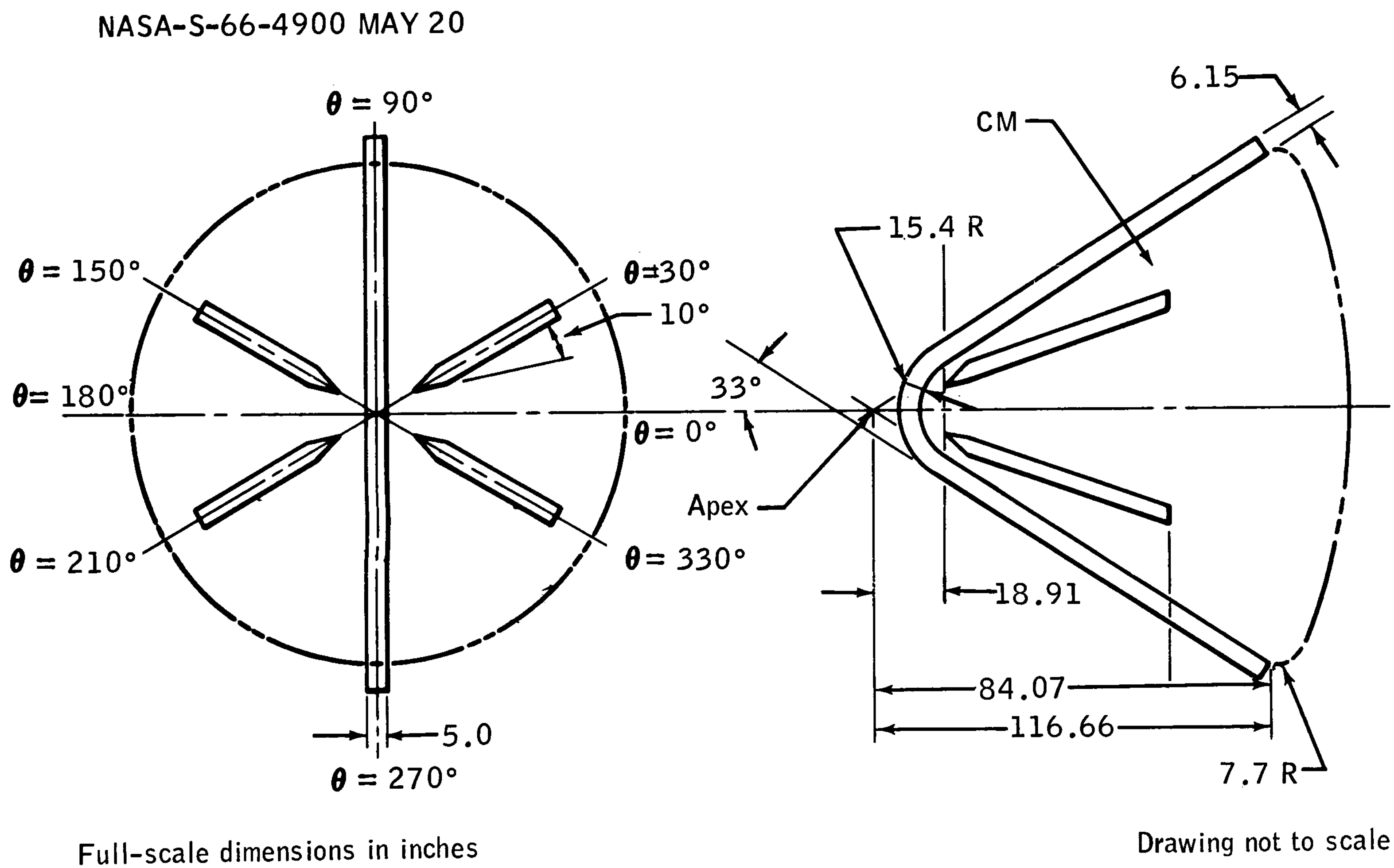


Figure 10 . - Typical keel configuration tested in the AWTTP.

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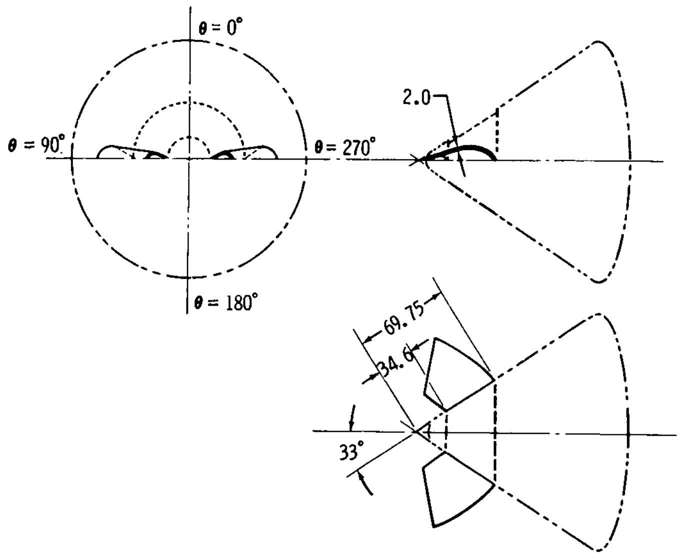
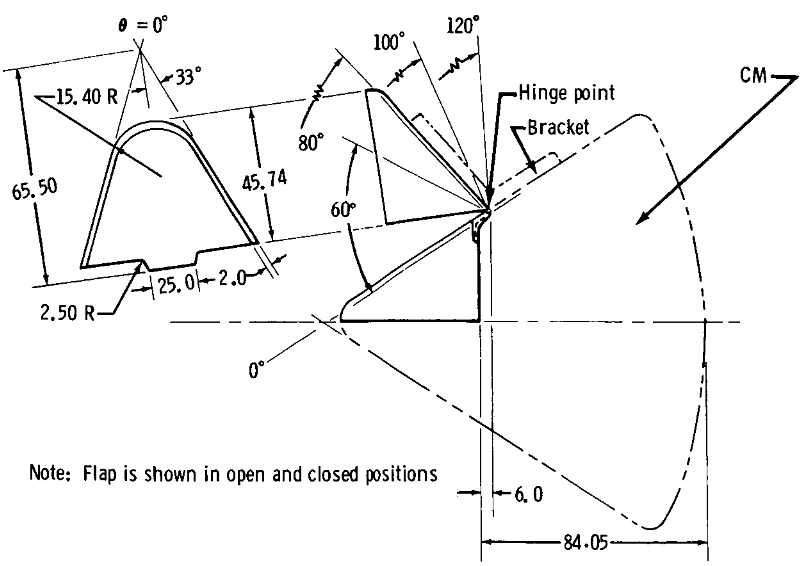


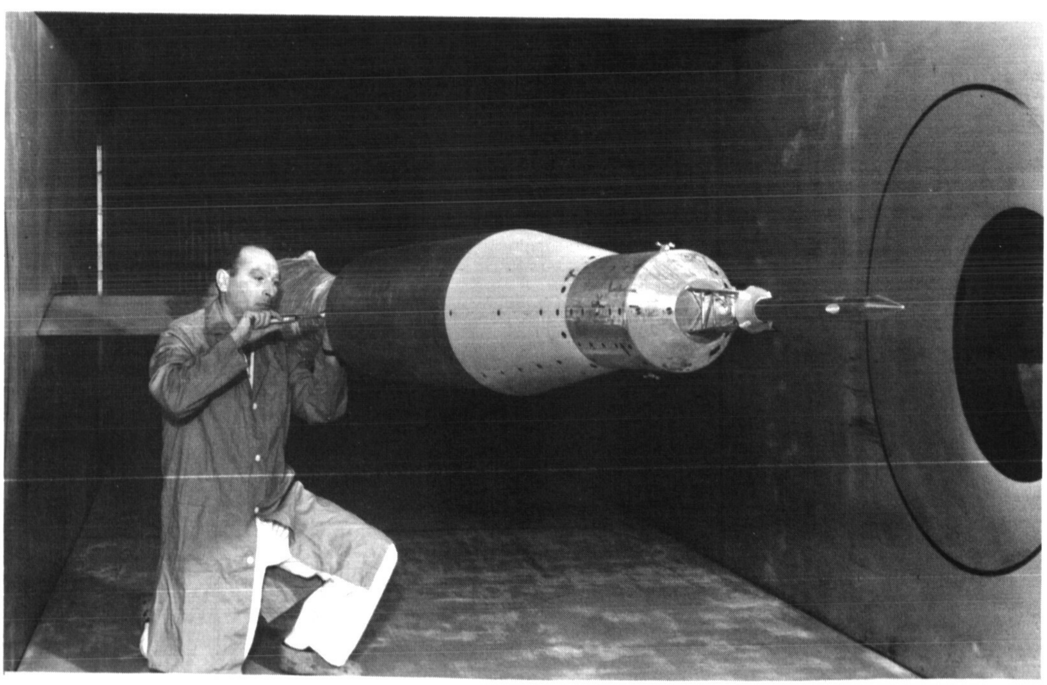


Full-scale dimensions in inches

Drawing not to scale

Figure 11. - Sketch showing two CM flap modifications tested.

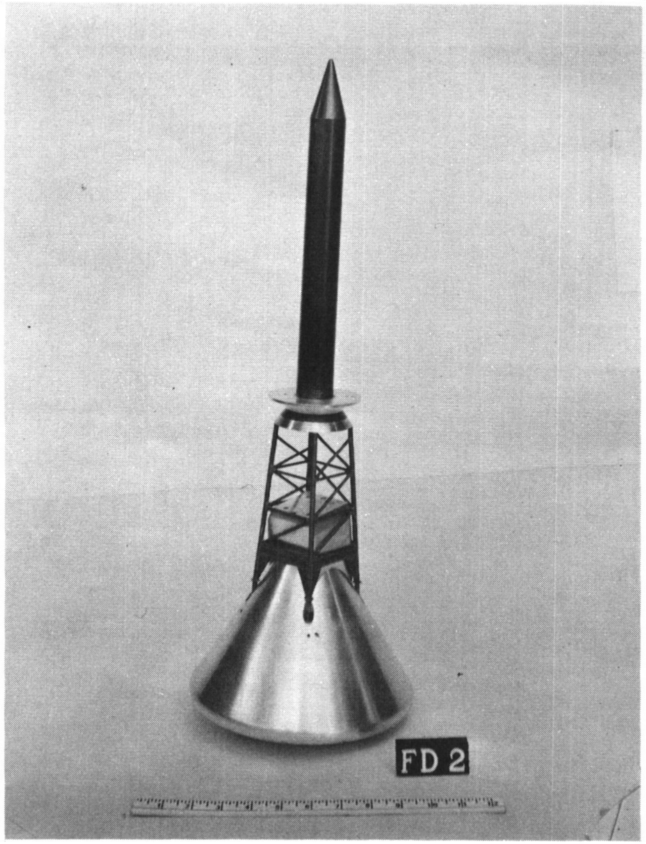
(a) PSTL-1 model mounted in the North American Aviation Trisonic Wind Tunnel.



(b) PSTL-2 model being installed in the Ames 9- by 7-Ft Unitary Plan Wind Tunnel.

Figure 12. - Photographs of two models used in transient pressure investigation.

NASA-S-66-4912 MAY 20



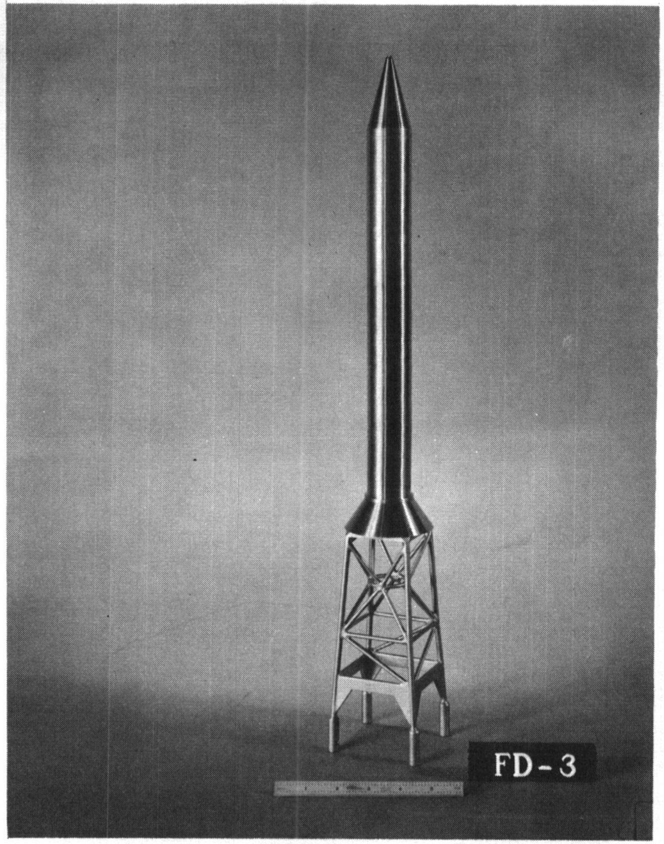


Figure 13. - Photographs showing typical tower web modifications tested on the FD-2 and the FD-3 models.

NASA-S-66-4913 MAY 20

Figure 14. - Photograph showing some of the canted heat shields and heat-shield corner modifications tested on the FS-1 model to improve L/D characteristics.

NASA-S-66-4914 MAY 20

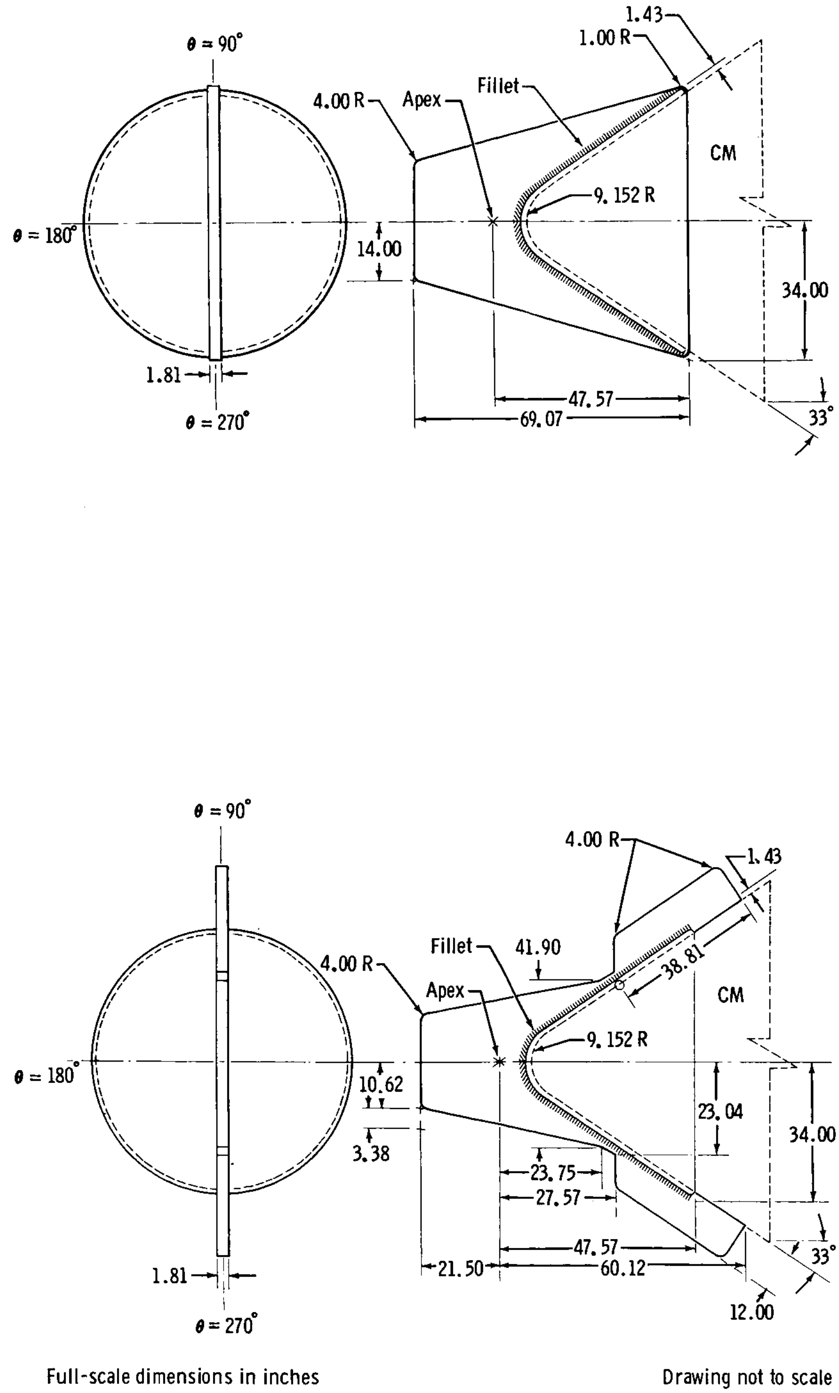
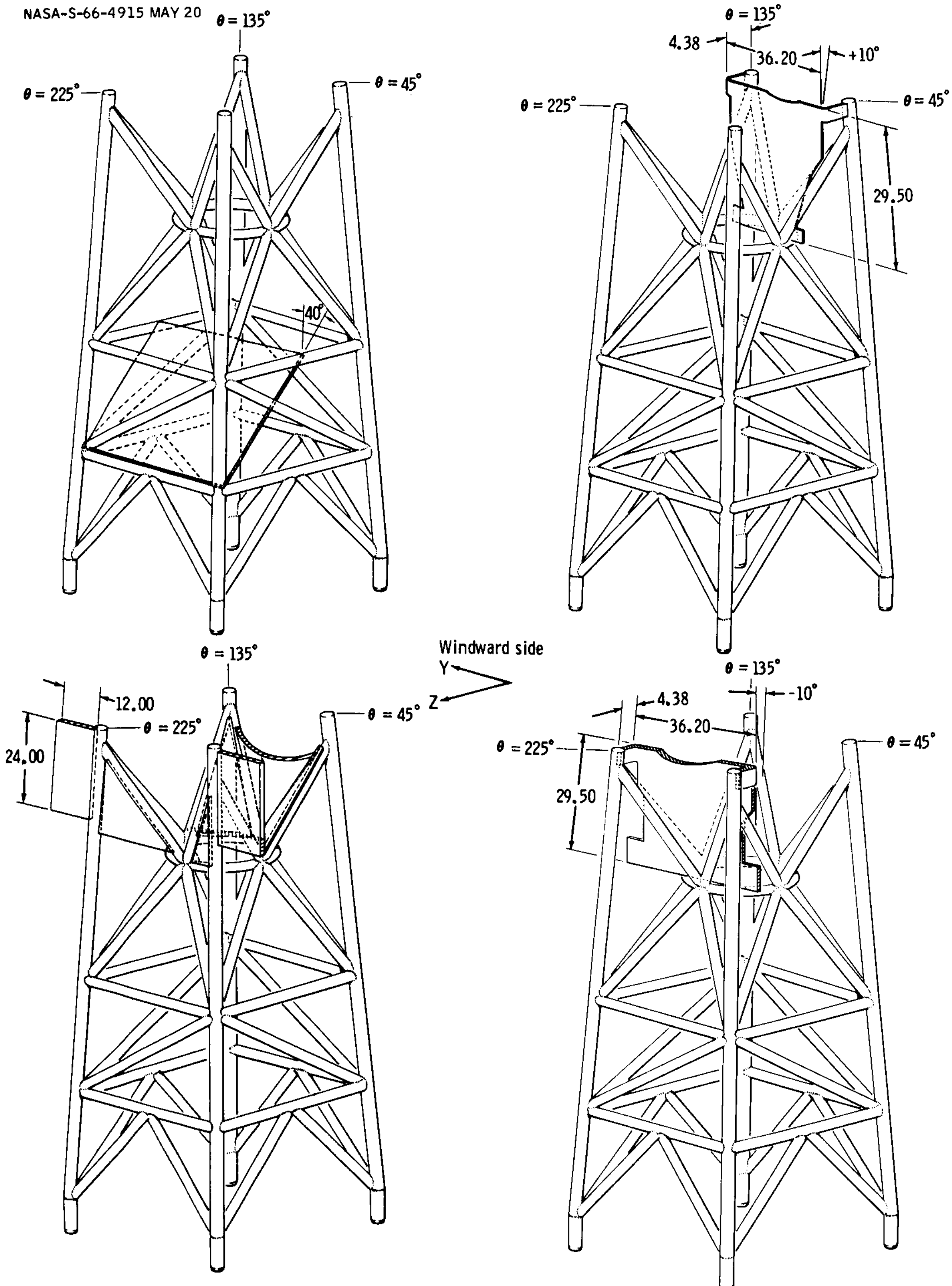


Figure 15. - Illustrations showing two beanie cap strake configurations that were studied.



Full-scale dimensions in inches

Drawings not to scale

Figure 16. - Sketch showing four escape tower configurations that were tested.

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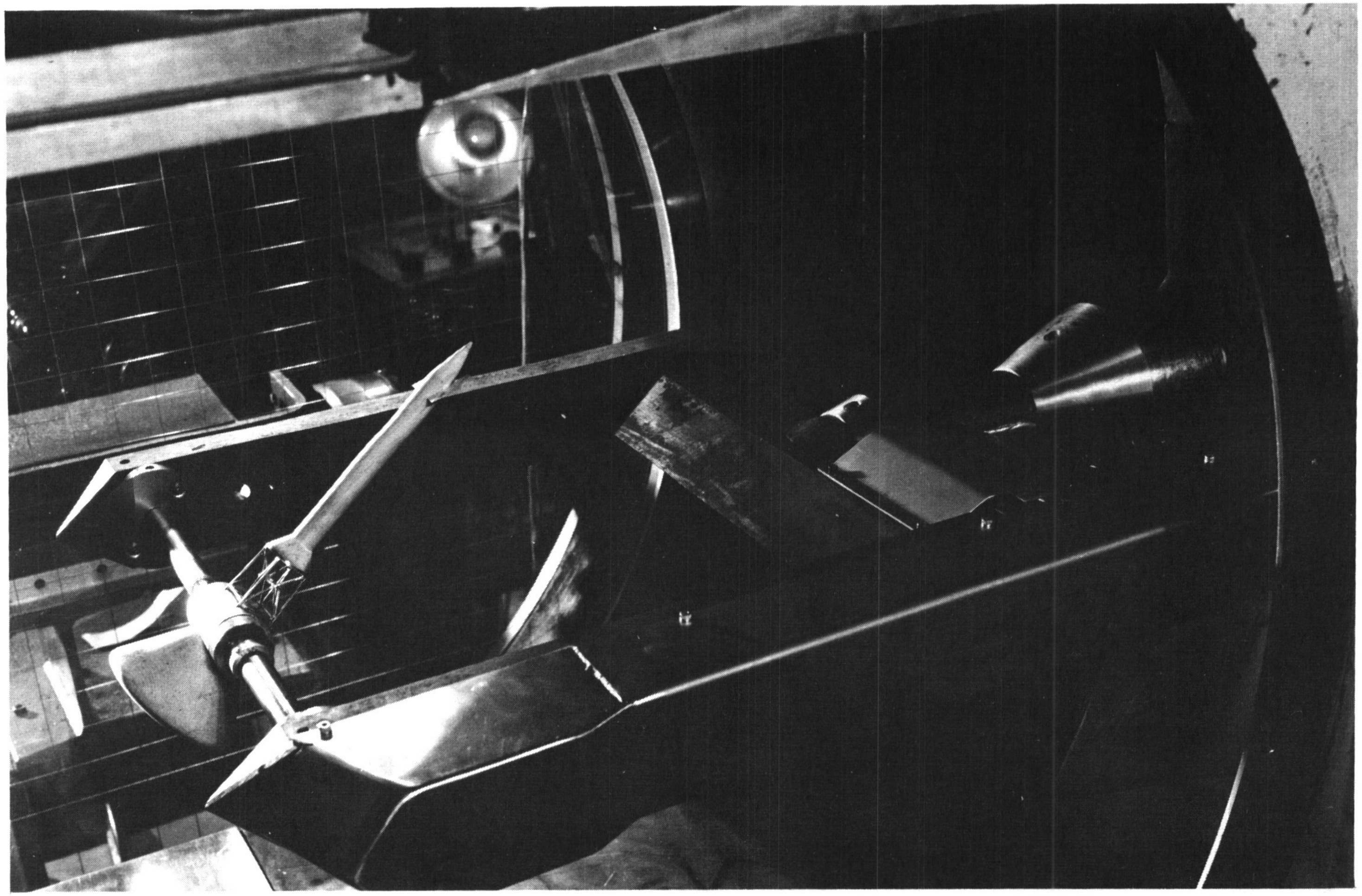

Figure 17. - Free-to-tumble model with the canard configuration mounted in the Jet Propulsion Laboratory 20-In. Wind Tunnel.

NASA-S-66-4917 MAY 20

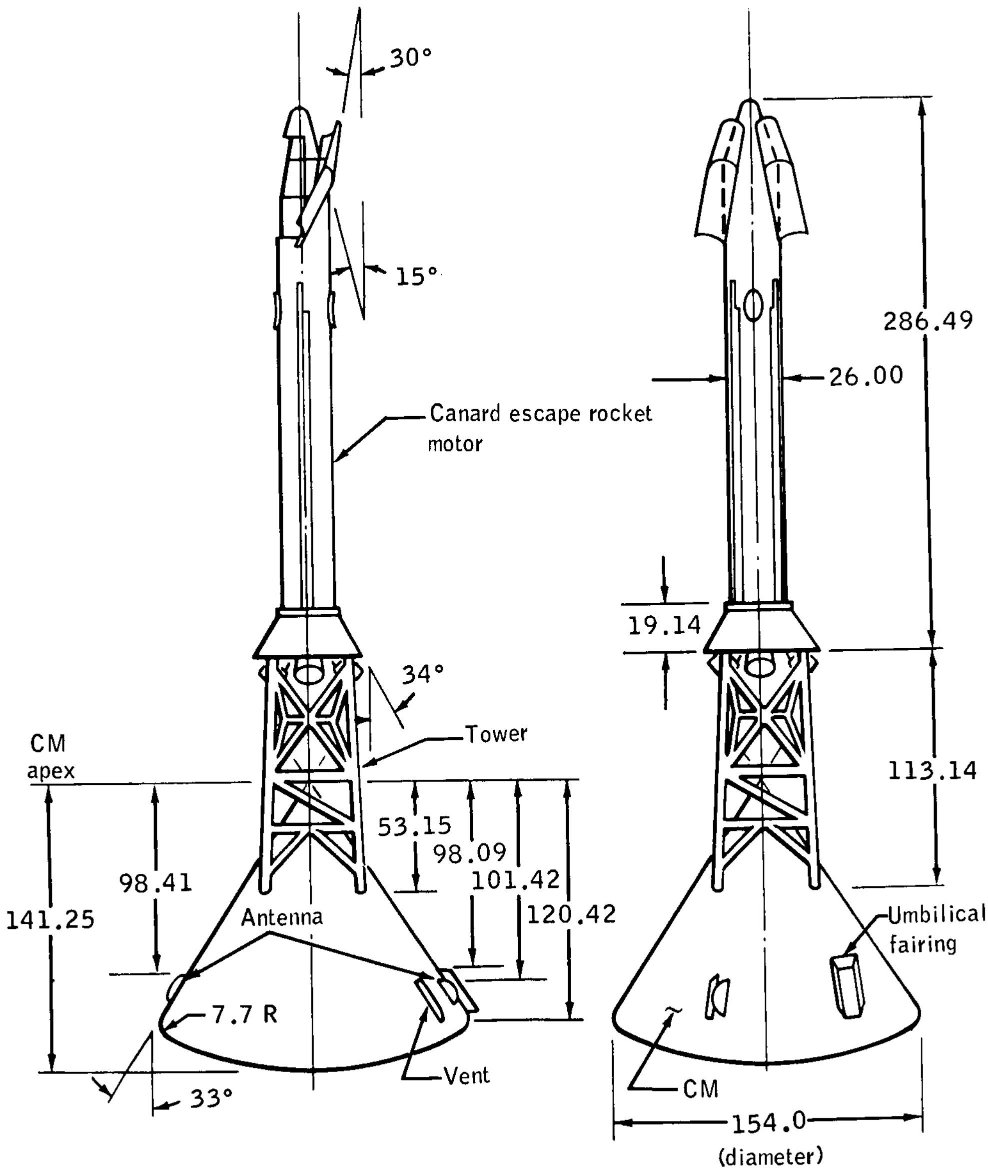

Figure 18. - Sketch of the canard LEV configuration.

NASA-Langley, 1966 S-114

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—NATIONAL AERONAUTICS AND SPACE ACT OF 1958

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