module (LM), the AAP dry workshop, the space shuttle, and the space station and its attendant experiments. The space shuttle will feature short turn-around capability, advanced maintenance concepts, and airline handling techniques. The launch site will resemble an airport of the future (Fig. 3). The space shuttle will support many programs (Table 1), each having characteristics and requirements which must be satisfied by the launch facility's capability. Reduced operational costs will be vital to the success of such future programs; continued high costs would reduce the scope of future programs by limiting the funds available for flight hardware, experiments, and research.

Summary

The fulfillment of NASA's space role will depend on the establishment of an efficient operational concept for processsing vehicles such as the space shuttle. The facilities, floor space, equipment, and manpower commitment can be minimized through implementation of the institutional approach, which would provide a high degree of standardization, adaptability, and flexibility in facilities and equipment.

A Tube Wind-Tunnel Technique for Rocket Propulsion Testing

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A TUBE wind tunnel for providing supersonic flows has been developed at the Cornell Aeronautical Laboratory (CAL) for rocket propulsion testing and has been in use for a number of years, primarily under the NASA/MSFC base-heating program.[1,2] The original motivation for the development of this short-duration wind tunnel was to provide an inexpensive means of producing an appropriate external flow environment (including a wide variation in freestream Reynolds number) for base-heating tests with models of Saturn-type boosters which avoided the complexity, cost, and frequent scheduling problems associated with testing rocket models in existing continuous flow or blowdown wind tunnels. The operation of the CAL tunnel and its use in base-flow investigations are discussed in this Note.

Short-Duration Tube Wind Tunnel

The main components of the tunnel are shown in Fig. 1. It is a modification of the type of blowdown tube wind tunnel conceived by Ludwig,[3,4] and it operates according to the nonsteady wave principles discussed in Ref. 5. A Ludwieg type of tube tunnel is presently undergoing check-out at NASA/MSFC for application in aerodynamic testing at high Reynolds number conditions.[7] In the tube tunnel at CAL the test gas is initially contained in a 42-in. i.d., high-pressure supply tube by a plastic diaphragm located upstream of the nozzle. Upon the mechanical cutting of this diaphragm, a centered expansion wave propagates upstream in the supply tube and accelerates the test gas from rest to a steady velocity. The gas then expands through the nozzle and into an evacuated 8-ft-diam, 30-ft-long dump tank. The expansion wave in the supply tube propagates upstream at acoustic velocity, and until the wave reflects from the far end of the supply tube and returns to the nozzle inlet, the nozzle supply conditions remain constant and the flow is steady. For the 30-ft-long supply tube currently in use, a steady nozzle supply is maintained for nearly 40 msec when using room temperature air as the test gas. This time is more than adequate to establish the nozzle and model flowfields and make the desired measurements. The nozzle starting time is on the order of 10 msec for this configuration with the diaphragm upstream of the nozzle. Various gases can readily be used as the test medium in this tunnel since the operation of the facility simply involves loading the desired gases into the closed supply tube prior to the test.

Since the volume of the supply tube is relatively small, a reasonably sized evacuated dump tank can be used. The latter allows higher altitudes to be simulated without concern for flow separation within the nozzle, and enables the tunnel to be operated in the laboratory environment by completing a closed system within which any test gas and rocket exhaust products are confined upon completion of a test, and overpressure and acoustic effects are contained. The evacuated dump tank also acts as a convenient "pumping" source for the choked-wall perforated nozzle developed for use with this tunnel.[6] The reduced pressure in the test section prior to the opening of the diaphragm also has the beneficial effect of reducing the starting load on the models to a level below the steady-state operating load. High starting loads on the models can be a significant problem in the Ludwieg form of the tube wind tunnel.[5]

The majority of tests performed in this tunnel have been concerned with base flow effects on scale-model rocket vehicles and basic investigations of rocket plumes. To avoid possible tunnel blockage and the effects of shock reflection and strut wake in base-flow investigations, the models are installed in a nacelle which is cantilevered from struts located in the supply tube and extends through the length of the nozzle to the test region (Fig. 2). The techniques developed by CAL for duplicating the liquid and solid-propellant rocket exhausts in such short-duration rocket testing are described in detail in Ref. 8 and will only be discussed briefly here.

Because test periods are short, uncooled combustor hardware of straightforward design and flexible operating capabilities can be used to provide the desired rocket exhausts relatively simply and economically. Liquid rocket-propellant combustion is simulated by burning gaseous fuels and oxidizers. Thermochemical duplication of most liquid propellant combinations can be achieved by the use of appropriate multicomponent combustible gaseous mixtures[9]; in some cases a less exact, but adequate and more convenient simulation achieved by the use of a single component gas. The use of gaseous propellants greatly simplifies the injector, combustor, and propellant supply system. The gaseous propellant supply system developed for these tests capitalizes on the same nonsteady gas-dynamic wave processes as used in the tube wind tunnel, and is simplified to a pair of pressurized gas storage tubes containing the oxidizer and fuel gases. To operate the rocket engine, supply tube valves are opened, allowing the gaseous propellants to flow out of the tubes through sonic metering venturis, through the injector, and into the combustion chamber where they mix and burn. As in the tunnel itself, coincidently with the start of flow from the tubes, nonsteady expansion waves propagate upstream in the supply tubes at acoustic velocity. Until these waves reflect from the far end of the tubes and return to the metering station, propellant supply conditions remain constant and steady combustor flow is achieved.

Actual full-scale propellant is burned in model rocket combustion chambers to provide solid-propellant rocket exhaust plumes. A thin web thickness of the propellant is

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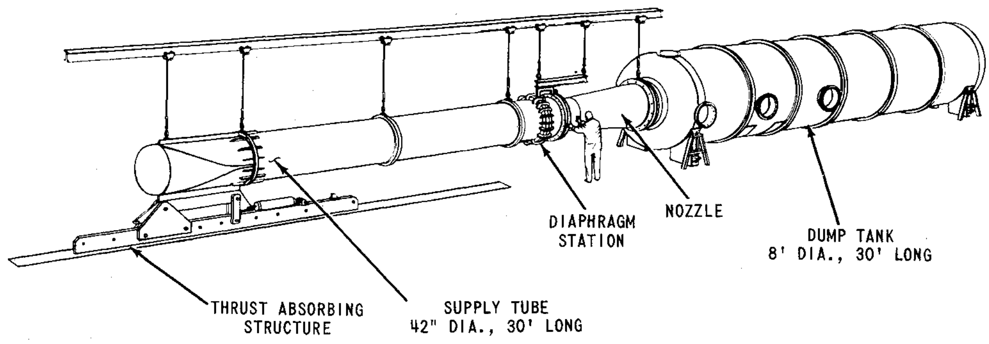


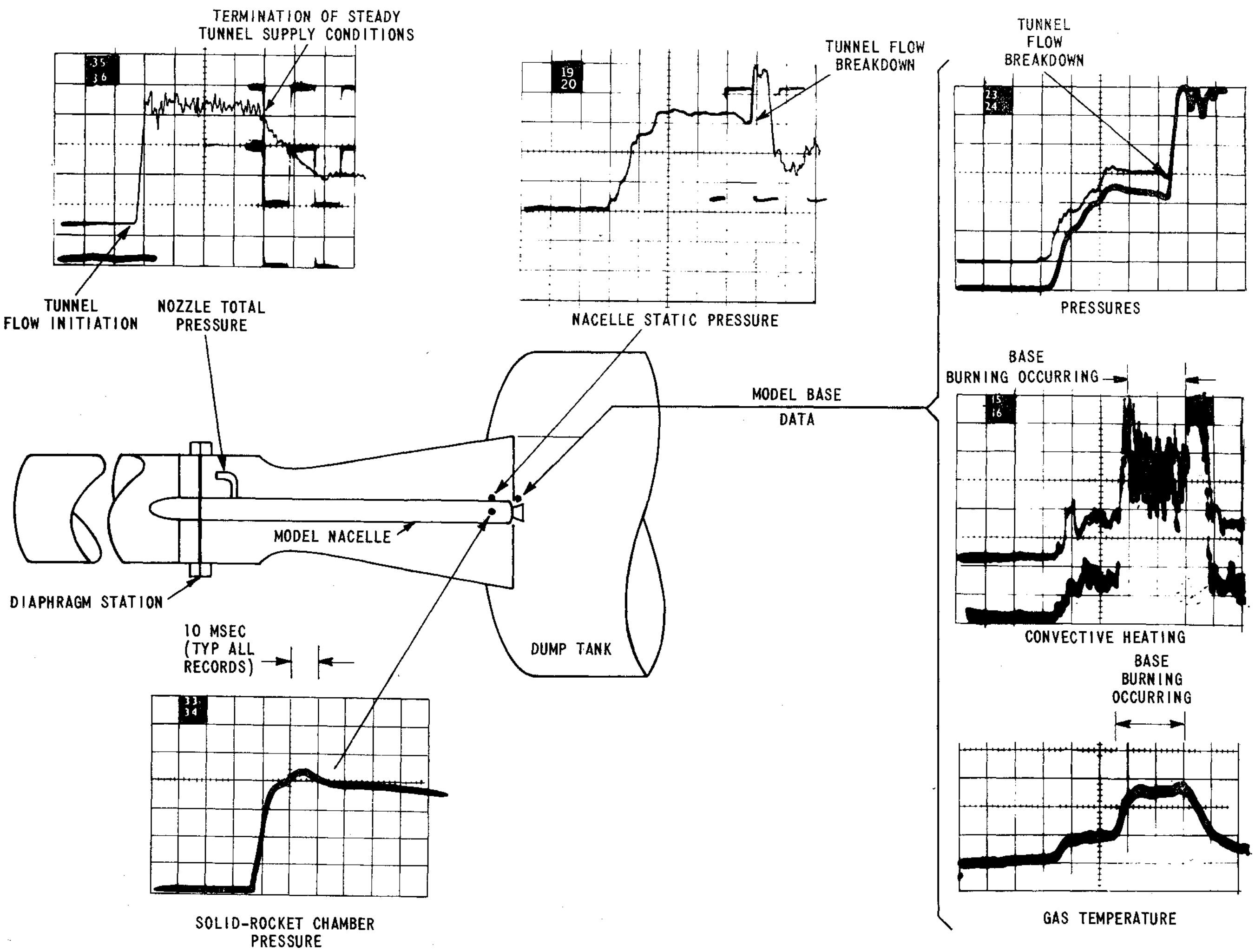
Fig. 1 Short-duration tube wind tunnel.

used to produce a total burn time on the order of 100 msec. The propellant load for the model motor is not poured or formed in the conventional manner but is simply made up from thin strips (order of 0.050 in. thick) of propellant cut from larger blocks and cemented to the motor and/or propellant holder walls.

Solid-Propellant Booster Base Flow Studies

The base flow and "base burning" encountered on a single-nozzle solid-propellant booster were investigated using a $\frac{1}{8}$-scale model of the booster aft end. Instrumentation for measuring pressure, heating rate, and gas temperature was provided at numerous locations throughout the region of interest. A graphite-phenolic liner was used in the rocket nozzle in order to more closely simulate the full-scale temperature of the boundary-layer gas which is recirculated into the base region. This nozzle design was such that during the test period the liner surface temperature reached a level close to that encountered in the full-scale nozzle. For these tests, the tunnel supply temperature and pressure were such as to provide complete duplication of the Mach 2 flight condition.

Representative oscilloscope records illustrating the various test events and their time-wise relationship are shown in Fig. 2. It is seen that the rocket and external flows were properly synchronized during the 40-msec test event and that model pressures, heating rates, and recirculated gas temperatures equilibrated well within the available test period. Base heating rate $\dot{q}_b$ and gas temperature T records illustrate the occurrence of base burning. It is noted that an initial data level was established for some 10–15 msec, followed by rapid rises in $\dot{q}_b$ and T at $\sim$20 msec into the test event. These



step increases are attributed to the initiation of afterburning of a mixture of recirculated fuel-rich combustion gases and freestream air. Further evidence of this base burning phenomenon was provided by remote optical radiation sensors viewing the base region which exhibited the same step behavior, as well as high-speed motion pictures, which clearly showed the appearance of flame in the base region at the time the other instrumentation adjusted to higher levels. When nitrogen was used as the external stream working fluid, only single lower level data were obtained, indicating that no base burning was occurring, as would be expected with such a nonoxidizing external stream.

Saturn S-IC Booster Base-Flow Investigations

These tests have involved the use of a $\frac{1}{45}$-scale model of the aft end of the S-IC booster installed in the centerbody nacelle. The five geometrically scaled F-1 nozzles operate at a chamber pressure of approximately 1150 psia and are supplied from a common combustion chamber burning gaseous oxygen and ethylene to provide a reasonably close thermodynamic simulation of the LOX/RP-1 propellants employed in the S-IC stage. Provisions are included in the nozzles for injecting turbine exhaust gases (either cold combustible fuels or an actual hot gas mixture generated by mixing combustion chamber gases with excess ethylene).

At the simulated Mach 2 trajectory conditions available in the tube tunnel, various scoop and flow deflector configurations have been studied to evaluate their effectiveness in forcing freestream air into the base region for the scavenging of recirculated rocket exhaust gases. An inoperative engine and various engine-gimbal combinations have also been investigated to determine their effect on the base thermal environment.

Some comparisons between flight data (from Saturn V AS-501 and AS-502 flight tests) and model test data have been made by Mullen and Bender.[10] They conclude that because of large uncertainties in scaling effects associated with the use of a small scale model and the significant differences in exhaust plume carbon content associated with the use of ethylene to simulate RP-1, the model data cannot be used directly to indicate absolute magnitude of convective or radiative base heating. However, as they indicate, the model data correctly predict the trends in the base heating environment associated with such configuration changes as removing flow deflectors or gimballing engines, as well as predicting the behavior of the exhaust plume with altitude.

References

[1] Hendershot, K. C., "The Application of Short-Duration Techniques to the Experimental Study of Base Heating," Rept. HM-1510-Y-18, April 1965, Cornell Aeronautical Lab., Buffalo, N.Y.

[2] Wilson, H. B., Jr., "Results from Short-Duration Altitude-Chamber Techniques for Simulating Rocket Base Heating Problems," *Journal of Spacecraft and Rockets*, Vol. 4, No. 5, May 1967, pp. 693–695.

[3] Ludwieg, H., "The Tube Wind Tunnel—A Special Type of Blowdown Tunnel," Rept. 143, July 1957, AGARD.

[4] Cable, A. J. and Cox, R. N., "The Ludwieg Pressure-Tube Supersonic Wind Tunnel," *The Aeronautical Quarterly*, Vol. 15, 1963, pp. 143–157.

[5] Falk, T. J. and Hertzberg, A., "A Tube Wind Tunnel for High Reynolds Number Supersonic Testing," Rept. AD-2297-A-1, Jan. 1967, Cornell Aeronautical Lab., Buffalo, N.Y.; also Rept. 68-0031, Feb. 1968, Aeronautical Research Lab.

[6] Davis, J. W., "A High Reynolds Number Wind Tunnel and Its Operating Concept," *Journal of Spacecraft and Rockets*, Vol. 5, No. 10, Oct. 1968, pp. 1225–1227.

[7] Sheeran, W. J., Hendershot, K. C., and Martin, J. F., "A Perforated-Wall Nozzle for Variable Mach Number Testing," *Journal of Spacecraft and Rockets*, Vol. 5, No. 9, Sept. 1968, pp. 1101–1103.

[8] Sheeran, W. J., Hendershot, K. C., and Llinas, J., "A Short-Duration Experimental Technique for Investigating High-Altitude Rocket Plume Effects," Rept. TOR-0200(S4960-10)-1, *Proceedings of the Rocket Plume Phenomena Specialists Meeting*, Vol. II, Oct. 1968, Aerospace Corp., San Bernardino, Calif., pp. 4-134 to 4-175.

[9] Sheeran, W. J. and Hendershot, K. C., "Simulation of Earth-Storable Liquid Bipropellants with Gaseous Reactants," *Journal of Applied Mechanics*, Vol. 36, Ser. E, No. 2, June 1969, pp. 347–348.

[10] Mullen, C. R. and Bender, R. L., "Saturn V/S-IC Stage Model and Flight Test Base Thermal Environment," AIAA Paper 69-318, Houston, Texas, 1959.

A Simplified Method for Determining Stagnation-Point Heat Transfer to an Elliptical Model

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Nomenclature

a,b	= minor and major semiaxes, of elliptical nose
C_{H_o}	= "no-blowing" Stanton number
k	= empirical correlation constant in heat-transfer coefficient equation
M	= freestream Mach number
P_o	= stagnation point pressure
P_∞	= freestream pressure
r^*	= radial distance from center line to sonic point
R_{eff}	= effective body nose radius
R_n	= radius of curvature of body at the nose
u_e	= velocity at edge of boundary layer
x^*	= axial distance from nose to sonic point
Φ^*	= sonic point inclination angle
Φ_{sph}^*	= sonic point inclination angle on a sphere
ρ_e	= density at edge of boundary layer

Introduction

An initially spherical ablating model tends in time to become more blunt and more elliptically shaped if the boundary layer is laminar.[1,2] To obtain accurate analytical analyses of experimental ablation results, it is necessary to consider the effect of this blunting on the stagnation point heat-transfer coefficient (cf Ref. 3). A method is outlined in this Note for obtaining a "no-blowing" heat-transfer coefficient from the stagnation pressure, the model radius of

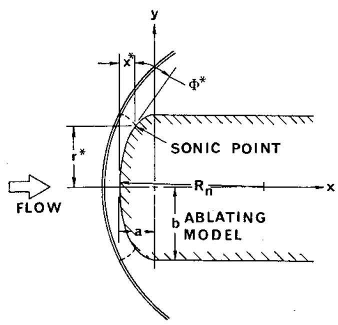

Fig. 1 Elliptical model geometry and nomenclature.

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