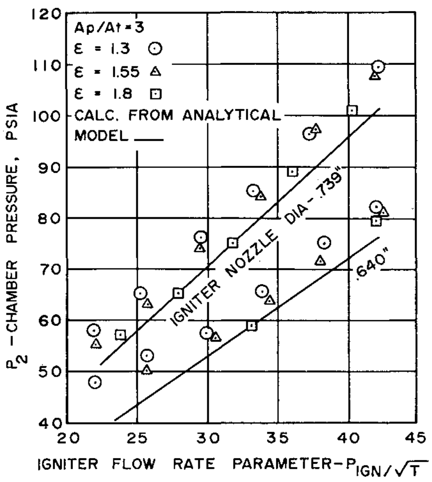

Fig. 3 Aft-end ignition cold-flow test data for motor port to throat of 3.

ducer calibration accuracy and about 3% for reading the oscillograph traces. For $P_2 > 65$ psia, the error in the analytical model compared to test data was +6% to −14% which corresponds to motor length penetration errors of approximately +2% to −4%, respectively. The analytically predicted pressures for $A_p/A_t = 3$ correlated better with the data than did those for $A_p/A_t = 1.3$, probably because the assumption of isentropic flow without viscous mixing is poorer for the latter case. This assumption also is amplified because of the physical size of the test hardware, where the mixing layer is larger relative to the jet size than it would be in the full size igniter and motor.

Conclusions

The analytical model is useful for sizing aft-end ignition systems with regard to motor pressurization and igniter-gas penetration. Figures 3 and 4 show the correlation of the analytical model to the test data. It is probable that the greatest cause of error in the analytical model is the assumption of no mixing. Jet mixing reduces the effective flow area in the motor throat which tends to make the pressures

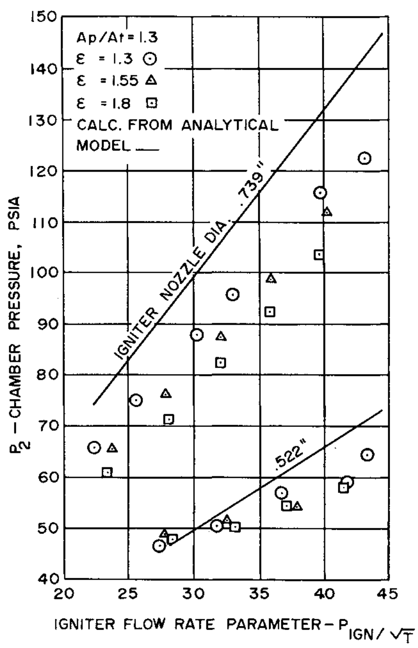

Fig. 4 Aft-end ignition cold-flow test data for motor port to throat ratio of 1.3.

higher; whereas, shredding off of the incoming jet stream would tend to reduce the incoming momentum and consequently lower the chamber pressure. Since these two effects are in opposition, the relative effects depend on the geometry and value of igniter-motor parameters. No attempts are made here to apply empirical corrections to the analytical model, since such corrections may not be scalable and may introduce errors greater than the assumptions. The error in the analytical model is less than the variation in chamber pressurization caused by igniter flow rate changes resulting from temperature variations of 60° to 100°F in ordinary solid propellants.

The axial position of the igniter (which controls ϵ^*) has relatively little effect on the motor pressurization for $1.3 < \epsilon^* < 1.8$; the only effect was a slight lowering of pressure at greater stand-off distances, probably due to the assumption that mixing did not occur between the igniter jet and the main gas stream. Pressure varied very little along the motor length; this lends credence to the conclusion that most of the igniter jet turning takes place in a relatively short distance inside the motor throat.

Saturn S-II Operations Development Progress

W. F. PARKER*
North American Aviation, Inc., Downey, Calif.

THE S-II is the second stage of the Saturn V launch vehicle. After separation of the S-IC first-stage engines, the S-II will, during its 395-sec burning time, increase the altitude of the S-IV-B and Apollo spacecraft from 200,000 to 600,000 ft and the velocity to 21,000 fps. The power to achieve this is provided by five Rocketdyne J-2 engines, each of which will have a thrust of 200,000 lb in vacuum, using LH_2 and LOX. The S-II (Fig. 1) is 33 ft in diameter and $81\frac{1}{2}$ ft long, weighing approximately 90,200 lb empty and 1,071,000 lb loaded. The total propellant capacity is 970,000 lb. The stage structure follows the conventional semimonocoque design using 2014-T6 aluminum alloy for the greater part. The major portions of the S-II are occupied by the LH_2 tank forward and the LOX tank aft, whose volumes are 38,400 ft³ and 12,930 ft³, respectively. The two propellant tanks are separated by a "common" bulkhead that comprises a forward-facing sheet as the lower bulkhead of the LH_2 tank and an aft-facing sheet as the forward bulkhead of the LOX tank. These sheets are separated by and bonded to an insulating core composed of a phenolic honeycomb (which varies in thickness from 4.75 in. at the center to 0 at its outside edges) that prevents the LOX from being frozen by the LH_2.

The LH_2 section comprises six cylindrical segments welded together at their peripheries to form a tank 56 ft long. Each cylindrical segment is composed of four curved sections. Prior to forming, these sections are machine milled to form a waffle pattern on the inside. The protrusions of this pattern, approximately $\frac{1}{4}$ in. thick with a depth of 1 in., provide for attachment of flanged tank frames. The LOX tank is a 33-ft-diam × 22-ft-high spheroid fabricated from ellipsoidal shaped upper and lower halves. The upper half is the "common" bulkhead already referred to. A J-section is welded to

Presented as Preprint 65-201 at the AIAA/NASA Flight Testing Conference, Huntsville, Ala., February 15–17, 1965; revision received April 9, 1965.

* Vice President and Program Manager, Saturn S-II, Space and Information Systems Division. Member AIAA.

the common bulkhead forward-facing sheet and to the lower LH_2 tank cylinder. The forward bulkhead of the LH_2 tank is welded to the no. 6 cylinder of the LH_2 tank wall. The S-II is attached to the S-IV-B stage through a forward skirt and to the S-IC stage through an aft interstage skirt. Separation of the S-IC from the S-II is accomplished by means of a linear shaped charge that physically severs the stages at station O on the S-II interstage. This function is electrically controlled from a signal received from the Saturn V Instrument unit.

Eight ullage rocket motors are spaced equidistantly around the aft interstage. These are fired at the time of separation from the first stage to provide sufficient S-II acceleration insuring a positive propellant pressure for the J-2 engine start. In order to rid the S-II of 8900 lb of unneeded structure shortly after the commencement of its powered flight, the interstage is jettisoned by explosively cutting 216 tension straps at station 196.

North American Aviation/Space & Information Division (NAA/S&ID) has contracted to deliver ten flight S-II stages to NASA's Marshall Space Flight Center by late 1968, in addition to a static-test /dynamic-test vehicle, an all-systems-test vehicle, and a facilities-check-out vehicle. LH_2 tank sec-

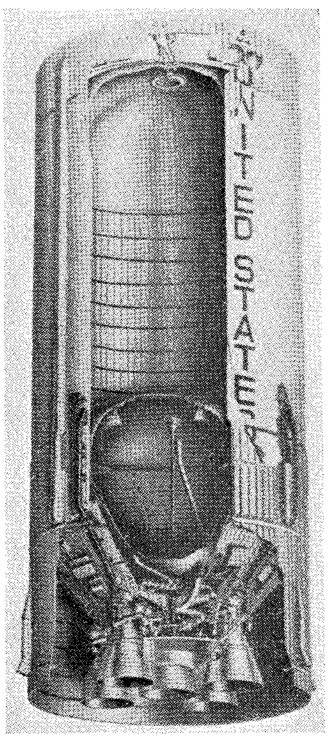

Fig. 1 Saturn S-II stage length = 81.5 ft, diameter = 33 ft, engines = 5 Rocketdyne J-2 engines, total thrust = 1,000,000 lb (in vacuum), and propellants = LH_2 815,400 lb and LOX 154,600 lb.

tions are machined and formed in North American's Los Angeles Division. Los Angeles Division also machines and preforms the gore sections that are welded together to form the tank bulkheads. After machining and preforming, these segments are formed through a high energy process at the NAA facility on the El Toro Marine Air Station in California. Nonpressurized portions of the stage, i.e., forward skirt, aft interstage, and thrust structure are manufactured at the NAA/S&ID facility at Tulsa, Okla. Electrical and hydraulic components are supplied by many subcontractors.

The S-II stage is assembled, hydrostatically tested and checked out at the NAA/S&ID Seal Beach, Calif., facility that is composed mainly of a bulkhead fabrication building, vertical assembly and hydro-test building, pneumatic test building, a structural static test tower (Fig. 2), a water conditioning unit, a warehouse, and a service building. Within the fabrication building, incoming gore segments of the tank bulkheads are trimmed, jig fitted, and weld-assembled by a skate welder. Each bulkhead is then radiographically in-

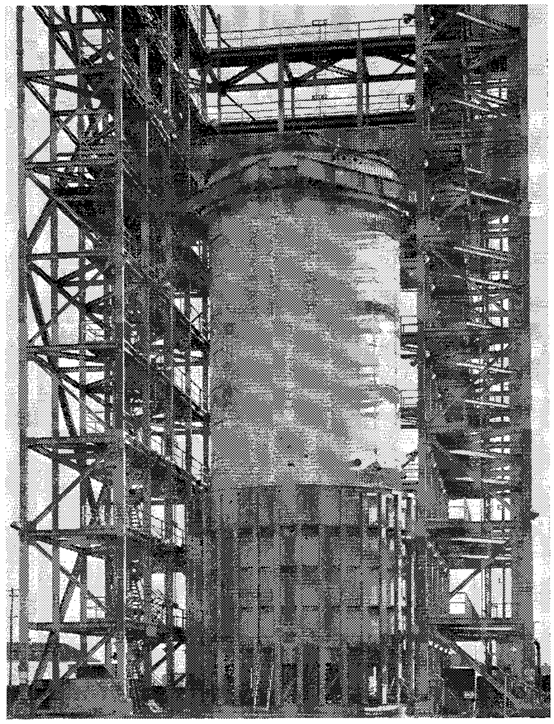

Fig. 2 Saturn S-II stage in static test tower.

spected, hydrostatically tested, and helium leak checked. The bulkhead is then thoroughly cleaned and etched. In the common bulkhead, the aft-facing sheet is next prepared for the application of the insulated core. This insulation is placed over a skin-tight vinyl adhesive sheet, after which it is vacuum-bagged, pressure-cured, and baked in the autoclave. The forward-facing sheet is placed in a vacuum bell, and its interior contour is read out while it is under vacuum. This information is fed to a contour miller that mills the insulation accordingly. An impression check is made to assure a 100% bond. Final assembly of the common bulkhead begins with the installation of a vinyl adhesive sheet to the interior of the forward-facing sheet, which is then placed over the honeycomb core in the autoclave, wherein the bonding operation is completed.

The stage shell is assembled in the vertical assembly building by placing the components in a vertical stack and welding them together. After radiographic inspection, the shell is then moved by a transfer dolly to the hydrostatic test area, where it is tested, leak checked, and cleaned in the vertical but inverted position. It is then returned to the vertical installation area, where the insulation closeouts are applied and the various systems installed. Individual and integrated checkout tests are performed with computer-controlled automatic-checkout equipment.

The static test tower will be used to certify the structural integrity of the complete airframe under simulated critical load and pressure differentials. The stage used in this testing will be the S-II-S, which will be a structurally complete S-II stage including tanks, bulkheads, thrust structure, and aft interstage. Engines will not be required but engine local support structure will be installed.

Adjacent to the S-II Downey facility, an electrical mechanical mock-up (EMM) has been constructed. This EMM consists of a full-size replica of the S-II forward skirt and LH_2 tank forward bulkhead in one section and a full-size replica of the LOX tank, aft skirt, thrust structure, and interstage. It represents a total S-II system including fully integrated and checked out sets of ground support equipment (GSE) and automatic checkout equipment. The EMM provides an excellent capability to train and indoctrinate test personnel, provide nonhazardous testing, investigate field problems, and incorporate design changes. A totally integrated systems tape will be verified on the EMM and used at Seal Beach. The test and computer room uses actual field test equipment.

In the Coca area of Rocketdyne's Santa Susana Field Lab., the battleship test stand, for engine systems testing, and the common bulkhead test stand have been constructed. The first, constructed of heavy-walled stainless steel patterned after the flight-weight configuration and utilizing a double-walled, vacuum-jacketed outer shell on the LH_2 tank, permits 1) full-scale evaluation of the J-2 engine cluster performance, 2) evaluation of engine start and shutdown characteristics, 3) verification of stage system component functions, and 4) duration firing evaluation and product improvement testing.

System redundancy on all of the cyrogenic systems has been installed for complete safety of operation. A control center and service center are provided which simulate those outlying activities and utilize many similiar pieces of checkout equipment. All of the checkout operations will be manually controlled.

The common-bulkhead-test tank is a full-diameter specimen consisting of a forward skirt, LH_2 bulkhead, common bulkhead, and foreshortened LH_2 tank located between the LH_2 and common bulkheads. Primary objective of its test is to verify structural integrity of the bulkheads and external tank insulation through application of flight bending moments, critical internal pressures, leak checks, heat losses, and a full proof test to destruction. Use of cryogenic liquids will determine actual heat transfer during ground environmental tests along with repeated pressure conditions.

The all-systems-test vehicle (S-II-T) is a complete flight-weight stage that, in a static-firing test stand, permits combined static-firing and cryogenic testing of the stage systems. All systems testing represents the first use of complete automatic GSE during servicing, checkout, and countdown procedures. The planned testing will be accomplished at Mississippi Test Facilities (MTF). Specific test objectives include 1) proving system performance over simulated entire flight sequence within environmental limitations, 2) proving stage operating and handling procedures, 3) determining optimum propellant chilldown and topping rates, and base cooling and cold soak rates, 4) developing detanking and shutdown procedures, and 5) determining compliance with flight requirement specifications.

The dynamic test stage (S-II-D) will be the first stage delivered to the NASA Marshall Space Flight Center. This simulation article will be used to measure bending mode frequencies and mechanical interface with the S-IC and S-IV-B stages. It will incorporate dummy engine masses capable of gimballed movement to simulate inertial characteristics. Equipment or components not required for the tests will be replaced by dummy masses.

The facility-checkout stage (S-II-F) is a flight-weight article similar in appearance to the S-II except for rigidly mounted engine masses with simulated dummy systems and partially operable systems. Its primary mission is to verify compatibility between the S-II flight stage, GSE, and facilities. The S-II-F will be delivered directly to Merritt Island Launch Area (MILA) at Kennedy Space Center.

To provide final reliability and qualification checking of the S-II, a series of tests will be run at Seal Beach, MTF, and MILA facilities utilizing computer-controlled automatic checkout equipment that will provide a fully integrated checkout of stage systems, record checkout operations and test results and stores, and compare and verify stage-system operational capability. In the event of a malfunction, it will be possible to switch from automatic mode of checkout to a local control mode. Handling equipment, which includes large rings, fixtures, and transporters, has been compounded by the State of California regulations that forced us to design two separate complete transporters, one for use on California roadways and the other for direct delivery to NASA bases.

Interaction of a Weak Shock with a Combustion Region

G. M. Lehmann* and S. N. B. Murthy†
Purdue University, Lafayette, Ind.

Nomenclature

a	=	acoustic velocity $(\gamma RT)^{1/2}$, fps
c_v	=	specific heat at constant volume, ft-lb/lbm-°R
H	=	dimensionless heat release function, Eq. (23)
K	≡	$\{[2 + (\gamma - 1)M^2][2\gamma M^2 + 1 - \gamma]\}^{1/2}$
M	=	Mach number of gas entering shock (relative to shock)
p	=	static pressure, lb/ft^2
T	=	static temperature, °R
t	=	time, sec
U_c	=	heat of combustion, ft-lb/lbm
u	=	gas velocity, fps
$\dot{w}$	=	mass flow rate per unit area, lbm/sec-ft^2
x	=	distance, ft
Y	=	concentration of reactant (ρ_r/ρ)
α	=	combination frequency and steric factor, sec^{-1}
γ	=	specific-heat ratio (c_p/c_v)
ϵ	=	activation energy, °R
ρ	=	density, lbm/ft^3

Subscripts

r	=	reactant flow property
s	=	steady-state flow property in undisturbed stream

Operators

$$\delta_+(\ )/\delta t = \partial(\ )/\partial t + (u + a)\,\partial(\ )/\partial x$$
$$\delta_-(\ )/\delta t = \partial(\ )/\partial t + (u - a)\,\partial(\ )/\partial x$$
$$\delta(\ )/\delta t = \partial(\ )/\partial t + u\,\partial(\ )/\partial x$$

Introduction

A PROBLEM of interest in the study of the unstable operation of a chemical propellant rocket motor is the formation and subsequent propagation of a weak shocklike pressure pulse in the combustion chamber. The mechanism involved in the formation of such a pulse in a flowing, chemically reactive gas is not adequately understood at present. If, however, the interaction of the rear of the pulse with its shocklike front can be neglected, the behavior of such a pulse may be examined by considering its propagation to be that of a weak normal shock.[1] Several investigators have employed either analytical or numerical methods for studying the propagation of a weak shock in a stationary gas containing a density gradient and in a stationary gas contained in a duct of variable area.[2–4] The analytical methods demonstrate that an approximate relation can be established between the change in the strength of the shock M and the change of the flow properties T, p, Y, etc. of the medium. The present note develops such a relation for the propagation of a normal shock through a chemically reactive flowing gas and compares the results obtained with those obtained by numerical integration of the conservation equations with appropriate boundary conditions.

Analysis

The combustion region of a rocket motor is three-dimensional, contains gradients in the flow properties, and is composed of a large number of chemical species. In the analysis,

Received April 1, 1965; revision received May 10, 1965. The investigation has been supported by the Office of Naval Research under Contract No. Nonr 1100 (21) and by the Air Force Office of Scientific Research under Contract No. AF AFOSR 753-65. The assistance of M. J. Zucrow, Director of the Jet Propulsion Center, Purdue University, is gratefully acknowledged.

* Research Assistant, Jet Propulsion Center.
† Visiting Professor of Mechanical Engineering, Jet Propulsion Center.