

VERY LOW THRUST GASEOUS OXYGEN-HYDROGEN
ROCKET ENGINE IGNITION TECHNOLOGY

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ABSTRACT

An experimental program was performed to determine the minimum energy per spark for reliable and repeatable ignition of gaseous oxygen (GO_2) and gaseous hydrogen (GH_2) in very low thrust 0.44 to 2.22-N (0.10 to 0.50-lb_f) rocket engines for spacecraft and satellite attitude control systems (ACS) application. Initially, the testing was conducted at ambient conditions, with the results subsequently verified under vacuum conditions. An experimental breadboard electrical exciter that delivered 0.2 to 0.3 mJ per spark was developed and demonstrated by repeated ignitions of a 2.22-N (0.50-lb_f) thruster in a vacuum chamber with test durations up to 30 min.

INTRODUCTION

The non-hypergolic propellant combination of gaseous oxygen (GO_2) and gaseous hydrogen (GH_2) requires an ignition source for combustion in a rocket engine. Very low thrust engines in the range of 0.44 to 4.45 N (0.10 to 1.0 lb_f) used in the attitude control systems (ACS) of spacecraft and satellites may be required to fire as many as a million cycles. The igniters of this class of engines must be highly reliable and repeatable as well as small size, lightweight, and use a minimum power input. Many different methods of ignition have been considered in the open literature. In Reference 1, Bell Aerospace Textron has made a qualitative comparison of six categories of ignition techniques suitable for a 56.7-kN (1500-lb_f) thrust GO_2/GH_2 engine for possible use in the Space Shuttle Auxiliary Propulsion Systems (SS/APS). The methods of ignition included the following:

- (1) Hypergolic or third chemical
- (2) Passive thermal activation
- (3) Dynamic thermal activation
- (4) Ionic
- (5) Catalytic
- (6) Electric spark.

Bell selected electrical spark ignition systems as being the most adaptable to their unique reverse flow engine design. However, the rating criteria cited are generally applicable to most engine applications. For small spacecraft and satellite ACS, a specially developed spark ignition system would appear to be the first choice, with a spontaneous catalytic ignition system a second candidate. Both of these systems, however, would require additional study and development to scale down the existing large engine ignition technology to make them compatible with very low thrust GO_2/GH_2 rocket engines.

It has been reported by the Marquardt Co. that high energy spark discharge had caused localized deterioration of protective coatings on molybdenum and columbium thrust chambers used in their "Water Electrolysis Propulsion System", Reference 2. In this instance, the electrical exciter delivered 10 mJ per spark and was acknowledged to be more than required for the 0.44-N (0.10 lb_f) thruster.

The objective of this study was to determine the minimum spark energy required for reliable and repeatable ignition of GO_2/GH_2 in small thrusters ranging from 0.44 N (0.10 lb_f) to 2.22 N (0.50 lb_f). After the spark energy level had been determined, the second objective was to design and build a breadboard electrical exciter and evaluate its performance with a small thruster operating at vacuum test conditions.

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EXPERIMENTAL METHODS

FACILITY

The experiments reported herein were performed in JPL test facilities at both ambient environment and vacuum environment test conditions. The same propellant supply system shown schematically in Figure 1 was used in each case. GO_2 , GH_2 and GN_2 were all obtained from standard K-size cylinders. GN_2 was used for leak checking, flow calibrating and purging both the oxidizer and fuel sides of the system. Spring loaded regulators were used to set the operating pressures ranging from 0 - 2758 kN/m^2 (0 - 400 psig). Gaseous flow control and flow rate measurements were made using sonic orifices. During ambient testing, three-way Marrotta solenoid valves were used for injector control. In these instances, the normally-open side was used to preset the proper supply pressure for desired propellant flow rate. The use of two separate injector valves also allowed the selection of either an oxidizer or fuel lead and lag during test firings. A single two-way Hydraulic Research bi-propellant valve was used for injector control on subsequent vacuum facility testing. A photograph of the vacuum chamber test facility is shown in Figure 2.

INSTRUMENTATION

All test data, with the exception of the electric spark parameters, were recorded on an oscillograph. The electric spark voltage and current were recorded and photographed on an oscilloscope using the block-circuit diagram shown in Figure 3. In each case, the data was manually reduced and calculated with a probable accuracy of ± 5 percent. Pressures were measured using Statham flush-diaphragm type transducers and temperatures were measured using type-K (chromel-alumel) thermocouples. Thrust was not measured during ambient testing, but was included in the recorded data during vacuum testing. A photograph of the thrust stand assembly is shown in Figure 4.

SPARK EXCITERS

Variable Energy Exciter. The exciter used to produce a variable energy spark was a Velonex Model 360 high-power pulse generator with a Model 1730 plug-in output unit to raise the output voltage up to a maximum of 10 kV. Using a 115-V, 60-Hz single phase power input, this instrument can be programmed to generate an electrical pulse of selected amplitude (voltage), width and repetition rate. The pulse has a well-defined wave shape, accurate repeatability and minimum ringing. This exciter was used to determine minimum spark energy required for reliable ignition during ambient testing and was later used to confirm this level during vacuum testing.

Experimental Breadboard Exciter. Once the minimum spark energy for reliable ignition had been determined, an experimental breadboard spark exciter was designed and fabricated to the following specifications:

- (a) Nominal input voltage +28 volts dc
- (b) Nominal output voltage 4.0 to 6.0 kV
- (c) Pulse width 2 to 3 μ s
- (d) Pulse repetition rate 200 to 300 Hz
- (e) Nominal spark current 200 mA
- (f) Minimum size and weight.

An electrical circuit diagram for this inductive type exciter is shown in Figure 5. The circuit may be considered as providing three electrical functions described as follows:

1. Oscillator/Switch Circuit - The oscillator (Q_1) is a standard programmable unijunction transistor (PUT) circuit oscillating at about 260 Hz. It provides gate pulses of around 200 mA peak to the silicon control rectifier (SCR) switch Q_2 . The SCR was selected for its small size and relatively fast turn-on capability. Though the switch is fast, there is a 5-V voltage drop across the switch at the peak discharge current of 30 A.
2. Charging Circuit - To compensate for the voltage drop across Q_2 switch and to still use a commercially available output transformer (turns ratio = 185 to 1), the energy storage capacitor (C_1) had to be charged to more than the 28 Vdc input. This was done by using a damped resonant charging circuit (L_1 , R_1 and D_1). The component values were selected to charge C_1 to 35 V when the input voltage is only 28 V.

3. High Voltage Circuit - After an initial check-out at ambient environmental conditions, the high voltage circuit was separated into a single module and potted in epoxy in order to eliminate the electrical corona during pulse generation in vacuum environment. This circuit consists of the output transformer, clamping diodes to prevent voltage reversal at the spark plugs, and damping resistors to limit the peak output voltage when the spark igniter does not fire. To verify the exciter operation during test firings, a current sensing transducer was added to one of the high voltage leads. A photograph of the breadboard exciter before separation and potting of the high voltage circuit is shown in Figure 6.

SPARK IGNITION CONFIGURATIONS

Standard Spark Plug. A standard Champion 10-mm Model Y-82 miniature spark plug, which was the smallest commercial plug readily available, was used for initial spark ignition test configuration. This spark plug is manufactured with a single ground electrode. It was base-mounted into the propellant injector head that had a single unlike doublet injection orifice. The combustion chamber was 0.64 cm ID x 6.35 cm LG (0.25 in. ID x 2.50 in. LG) with mounting bosses for a flame sensing thermocouple and a chamber pressure transducer. Three interchangeable screw-on type exit nozzles with throat diameters of 0.076 cm (0.030 in.), 0.152 cm (0.060 in.) and 0.208 cm (0.082 in.) were used. Each nozzle had an expansion ratio of 1 to 1. All these component parts were made of stainless steel with the exception of the annealed copper gasket seals. An exploded view photograph of this test configuration is shown in Figure 7.

Modified Spark Plug. A second Champion Model Y-82 spark plug was modified by removing the ground electrode and electron beam (EB) welding an extension onto the center electrode. This plug was base-mounted into an injector head containing a conical cavity, the exit end of which was coaxial with the extended spark electrode and formed a concentric ground electrode. GO_2 was injected radially into the cavity and exited axially through the annular spark gap. GH_2 was injected coaxially just below the spark gap. The same combustion chamber and exit nozzles described above were used in this test assembly. This configuration is basically a small scale version of the familiar GO_2/GH_2 torch igniter used by many large rocket engine manufacturers, namely Rocketdyne, Pratt and Whitney and Aerojet (Reference 3). For simplicity, this configuration has no secondary passage for GH_2 cooling as was later employed in the vacuum test thruster described below. An exploded view of the modified spark plug ignition test configuration is shown in Figure 8.

Spark Ignition Thruster. The modified spark plug igniter configuration was used in the design of a 2.22-N (0.5-lb_f) thrust rocket engine used in the vacuum test facility to evaluate the breadboard electrical spark exciter. The injector head was modified to improve combustion efficiency and provide barrier type cooling for the combustion chamber wall. To obtain better radial distribution of the GO_2 in the spark plug cavity, an injection manifold was added that had four radial passages 90° apart. For thrust chamber cooling, a coaxial sleeve containing two flow splitting metering plates was inserted into the GH_2 manifold. The first plate metered a small portion of the GH_2 into the annular spark gap for ignition and primary combustion. The second plate metered the major portion of GH_2 along eight axially grooved passages in the sleeve and discharged the GH_2 parallel to the combustion chamber wall. This provided both fuel for secondary combustion and a GH_2 -rich barrier to cool and protect the wall from the burning core mixture. The combustion chamber was also modified to include a conical exit nozzle with a 40 to 1 expansion ratio. For this test configuration, the sleeve was made of nickel, the metering plates from brass, and the combustion chamber used type 347 stainless steel. This same thruster configuration was later used in another program to evaluate a rhenium metal combustion chamber for very high temperature operations. A schematic drawing of the 2.22-N (0.50-lb_f) thrust rocket engine is shown in Figure 9 and a photograph of the assembled engine in Figure 10.

TEST PROCEDURE

AMBIENT TESTS

Ignition Test Conditions. Ignition test conditions were selected based on the 0.44-N (0.1-lb_f) thrust GO_2/GH_2 rocket engine developed by Marquardt for the Water Electrolysis Propulsion System reported in Reference 2. This engine was designed to operate at an O/F ratio of 8 and a chamber pressure of 517 kN/m² (75 psia). The reported specific impulse at this condition was around 325 sec. For ignition testing, the range of thrust was expanded to also include a nominal 1.33 N (0.3 lb_f) and a nominal 2.22 N (0.5 lb_f). Three interchangeable exit nozzles were sized to produce a chamber pressure of around 5.7 kN/m² (75 psia) at these thrust levels and an O/F mixture ratio of 8. Since the combustion chamber diameter and length remained constant, the resulting L* were 58, 109 and 442 cm (23, 43 and 174 in.). Additional O/F ratios of

2, 4, 6 and 10 were also included. The changes in O/F ratios were obtained by holding GH₂ flow rate constant and varying only the GO₂ flow rate. This also resulted in the chamber pressure varying between 107 and 552 kN/m² (30 to 80 psia).

Standard Spark Plug Test. The minimum spark energy for reliable ignition was first determined using the standard spark plug configuration. Starting at the 0.44-N (0.1-lb_f) thrust level and an O/F ratio of 8, the applied pulse voltage was reduced in steps from 8000 V to just below 2400 V, and the pulse width from 10 to 1 μs. Pulse repetition rate was increased from 30 to 300 Hz. When the minimum spark energy level was determined for short 2 to 3 s steady-state firings, it was further verified by a series of 5 to 10 pulsed-mode firings having a duty cycle of approximately 0.5 s "on" and 1.0 s "off". The same steady-state and pulsed-mode firings were then repeated for O/F ratios of 2, 4, 6 and 10. This same procedure was repeated for the nominal 1.33-N (0.3-lb_f) and 2.22-N (0.5-lb_f) thrust levels after changing to the appropriate exit nozzle and re-adjusting the propellant flow rates. All of the above tests were conducted with a 30-ms GH₂ lead on "start-up" and a 30-ms GH₂ lag on "shut-down" to prevent oxidation of hot metal surfaces.

Modified Spark Plug Test. The minimum spark energy for reliable ignition was determined for the modified spark plug configuration by repeating the entire test procedure described above. The only change in procedure was to change to a 30-ms GO₂ lead on "start-up" to prevent over-pressurization (detonation) inside the spark plug cavity. Simultaneous operation of GO₂ and GH₂ for "start-up" and "shut-down" was also evaluated.

VACUUM CHAMBER TESTS

Spark Ignition Thruster Test. The 2.22-N (0.5-lb_f) thrust GO₂/GH₂ rocket engine was installed on the thrust stand in the vacuum test facility. To support this program, the gaseous propellant system, instrumentation, data recorders and variable energy spark exciter were all relocated from the ambient test area. Initial tests were conducted at ambient conditions with the vacuum cell open to confirm the previous test results and to check-out facility operations. The vacuum cell was then closed and evacuated to 13.8 N/m² (0.002 psia). The tests to determine minimum spark energy were conducted at the following nominal conditions:

- Thrust, F = 1.78 to 2.67 N (0.4 to 0.6 lb_f)
- Chamber pressure, P_c = 345 to 483 kN/m² (50 to 70 psia)
- Overall O/F ratio, MR = 3.5 to 4.5
- Core O/F ratio, MR_c = 13 to 40.

Both short steady-state tests of about 5.0 s and 5 to 10 pulse mode tests with a duty cycle of about 0.5 s "on" and 1.0 s "off" were made. The duration of tests were performed so as not to exceed 815°C (1500°F) temperature on the stainless steel nozzle throat. After the minimum spark energy and repetition rate had been documented for reliable ignition at vacuum test conditions using the variable pulse generator, the breadboard electrical spark exciter was designed and fabricated. A final series of tests using these same conditions were conducted to verify the operation of the experimental exciter in both ambient and vacuum environment.

High Temperature Thrust Chamber Test. The breadboard electrical spark exciter was used with the 2.22-N (0.5-lb_f) thrust GO₂/GH₂ rocket engine to evaluate the high temperature rhenium metal thrust chamber. The same range of vacuum test conditions described above were used in this program except that steady-state run durations were conducted for periods up to 30 min. The spark exciter on test firings exceeding 5 s was turned "off" once ignition had been confirmed either visually or from the o-graph data trace.

TEST RESULTS

STANDARD SPARK PLUG MINIMUM IGNITION ENERGY

In the tests performed at ambient conditions using the standard spark plug configuration, the spark gap was set at 0.89-mm (0.035-in.) width. For this gap, the break-down voltage at spark initiation was around 2.5 kV and was accomplished in less than 2 μs. Once the spark gap was ionized, the voltage dropped to 0.1 kV for the remainder of the pulse width. Peak current at the time of initiation of the spark was around 0.25 A, then dropped to about 0.08 A at 10⁶ Hz. For a 10 μs pulse width, the major portion of spark energy was discharged in the first 2 μs it took to initiate the spark. It was believed that this initial energy discharge

was sufficient to cause ignition of the GO_2/GH_2 propellants and that a longer pulse width was not required. It was also believed that the required exciter voltage had to be only slightly higher than the spark break-down voltage to assure ignition. To verify this, the set voltage on the variable pulse generator was reduced in steps from 8.0 kV to 2.4 kV. Any setting below 2.4 kV would not produce a spark to ignite the GO_2/GH_2 . Following this, the pulse width was reduced in steps through 10, 5, 2.5 and $1.0\mu\text{s}$. A pulse width setting below $1.0\mu\text{s}$ would not produce a spark. Typical photographs of oscilloscope traces for the four different pulse widths are shown in Figure 11. Pulse repetition rate was also reduced in steps through 300, 200, 100, 50 and 30 Hz. The more reliable repetition rate for pulse mode type thruster firing was achieved at 100 Hz or higher. As a result of these exploratory tests, the exciter parameter settings for the standard spark plug configuration were as follows:

- (a) Supply voltage of 4.0 kV because in later tests the spark plug breakdown voltage increased from 2.5 to 3.5 kV,
- (b) Pulse width of $2\mu\text{s}$ to obtain only the initial current surge during ionization of the spark gap,
- (c) Pulse repetition rate of 300 Hz for both steady-state and pulse mode duty cycles.

These settings consistently delivered about 0.2 mJ per spark, which proved to be sufficient to ignite GO_2/GH_2 with the standard spark plug configuration at all variations of propellant flow rates, mixture ratios and resulting chamber pressures. Reliable and repeatable ignition were obtained for both the short steady-state and pulse mode type firing operations. Posttest examination of the spark plug electrodes showed little, if any, signs of deterioration.

MODIFIED SPARK PLUG MINIMUM IGNITION ENERGY

The exploratory tests described above were repeated for the modified spark plug ignition configuration. Since the annular gap between center electrode and the inside diameter of the coaxial GO_2 injection orifice was also 0.89 mm (0.035 in.), the resulting spark exciter parameter settings were identical and produced the same 0.2 mJ per spark. Ambient condition test firings at all variations of GO_2/GH_2 flow rates, mixture ratios and chamber pressures resulted in successful ignitions and combustion for both short steady-state and pulse mode operations. Posttest examination of the spark electrode and the GO_2 injector showed no evidence of material deterioration.

VACUUM CHAMBER SPARK IGNITION

The vacuum chamber test firings with the 2.22-in (0.50 lb_f) thrust GO_2/GH_2 rocket engine were successful in achieving ignition and combustion using the same minimum energy settings on the variable pulse spark exciter for the nominal test conditions. The only modification required to eliminate an electrical corona inside the vacuum chamber was to enclose the high voltage lead with Tygon tubing sealed at both ends to maintain one atmosphere pressure around the insulated conductor. Ignition at the lean O/F mixture ratios of 13 to 40 in the core region of the spark plug were readily attained at vacuum start conditions. Although it was not documented, it was assumed that spark was initiated as soon as the thrust chamber pressure exceeded 34.5 kN/m^2 (5.0 psia) due to gaseous propellant injection. The oscillograph trace of thrust chamber pressure showed a smooth increase in pressure with no over-shoot spike. Figure 12 shows a typical o-graph data recording for pulse-mode firings. Oscillations on the thrust trace are caused by mechanical motion within the injector valve during opening and closing.

BREADBOARD ELECTRICAL SPARK EXCITER

The breadboard electrical spark exciter was completed in time to be used in support of the high temperature thrust chamber evaluation. As stated, this program utilized the same spark ignition 2.22-N (0.50 lb_f) thrust GO_2/GH_2 rocket engine developed for minimum spark energy testing. The only change involved a replacement of the stainless steel thrust chamber with one made of rhenium metal. With this chamber it was possible to test the modified spark plug configuration and the experimental spark exciter with long engine firing times, some up to 30 min duration. As the result, many successful ignitions and combustion firings were demonstrated. A typical voltage/current trace for this exciter is shown in Figure 13. Because this unit uses an inductive circuit, as different from the capacitor circuit of the variable energy pulse generator, the trace shows a more oscillatory pulse wave form. The energy delivered is still 0.2 to 0.3 mJ per spark. Posttest inspection of the spark plug electrode and injector head revealed no deterioration of metal surfaces from spark discharge or over-heating.

CONCLUSIONS

The minimum spark energy for reliable and repeatable ignition of very low thrust 0.44 to 2.22-N (0.10 to 0.50-lbf) GO_2/GH_2 rocket engines can be as low as 0.2 mJ per spark. A modified commercial spark plug can be incorporated into a coaxial injector head configuration that will result in low break-down voltage, cooled durable electrodes, fast ignition response, and dependable thruster performance. To provide excitation for this spark ignition configuration, a small, lightweight inductive type pulse generator operating with a nominal 28 Vdc input can be made to deliver the required high voltage, current, pulse width, and pulse repetition rate. The spark plug and electrical exciter can be packaged into a single unit, without interconnecting high voltage cabling, similar to the ignition system developed by Rocketdyne for the Space Shuttle Auxiliary Propulsion System (SS/APS), Reference 4. This compact design will minimize size and weight, and eliminate radiated radio frequency interference (RFI), all of which are highly desirable specifications for any spacecraft or satellite attitude control system (ACS).

ACKNOWLEDGMENTS

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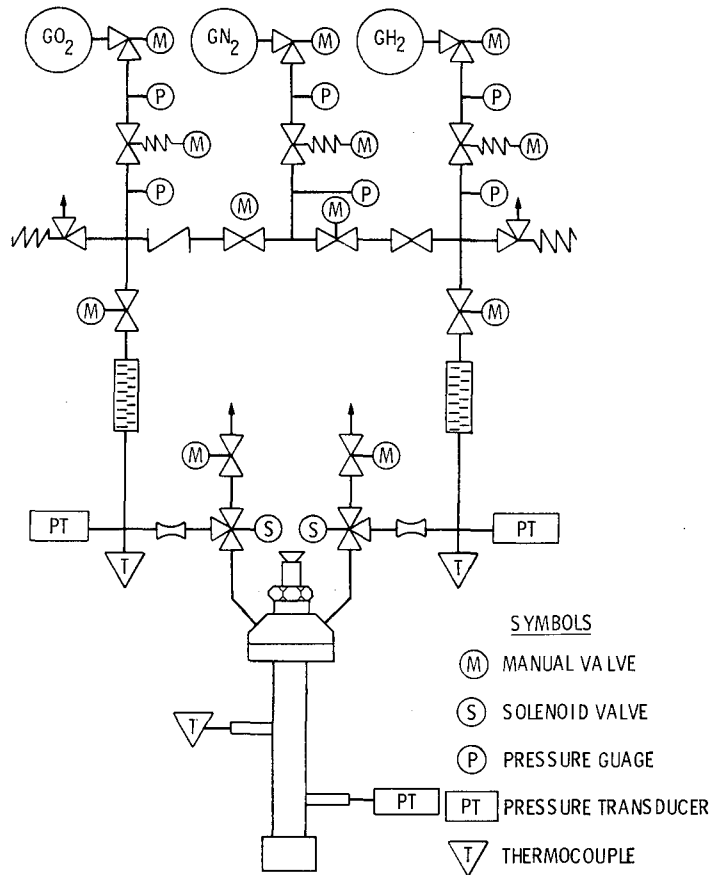


Figure 1. Schematic Drawing of G02/GH2 Propellant Test System

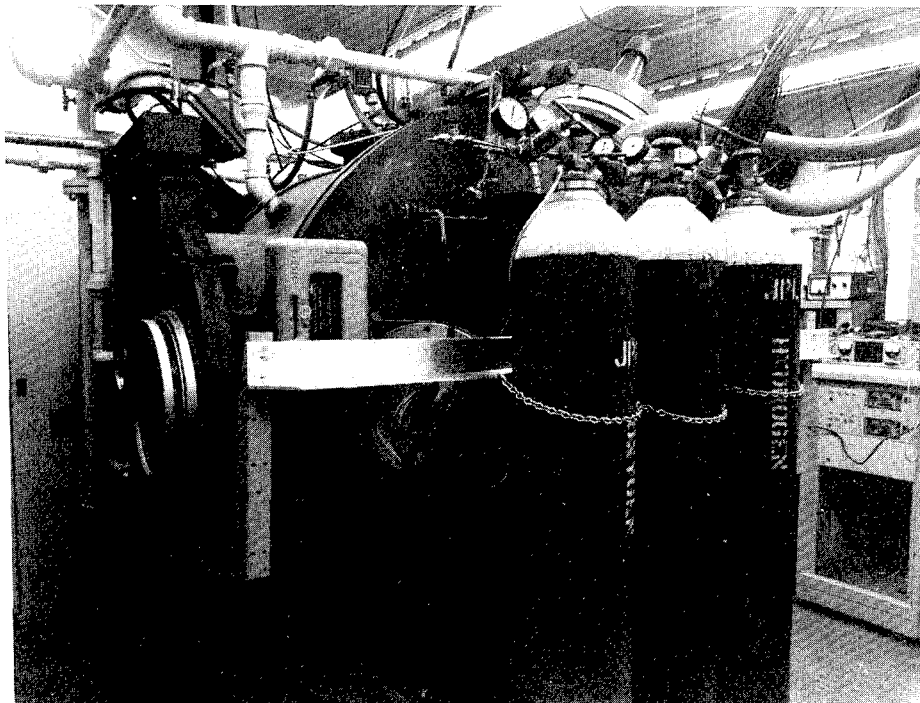


Figure 2. Vacuum Chamber Test Facility for Low Thrust Rocket Engines

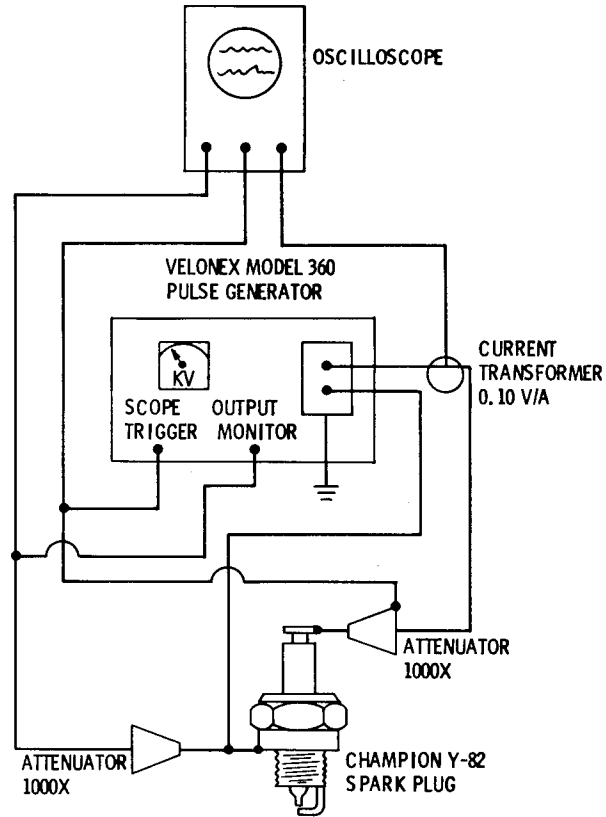


Figure 3. Block Circuit Diagram for Spark Ignition Data Recording

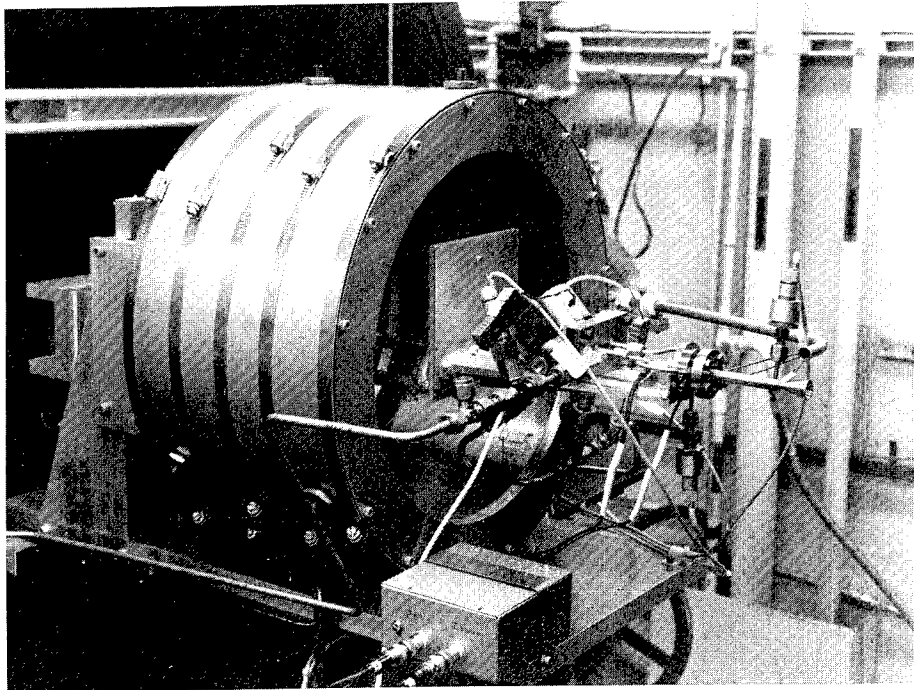


Figure 4. Thrust Stand Assembly for Low Thrust Rocket Engine Testing

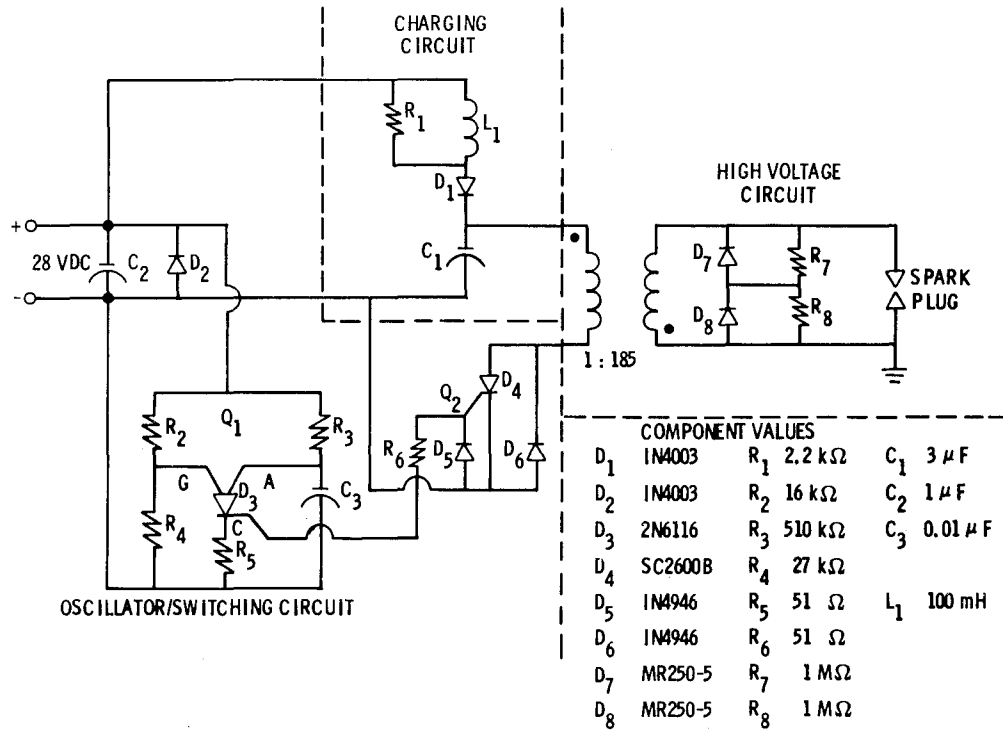


Figure 5. Electrical Circuit Diagram for Experimental Breadboard Spark Exciter

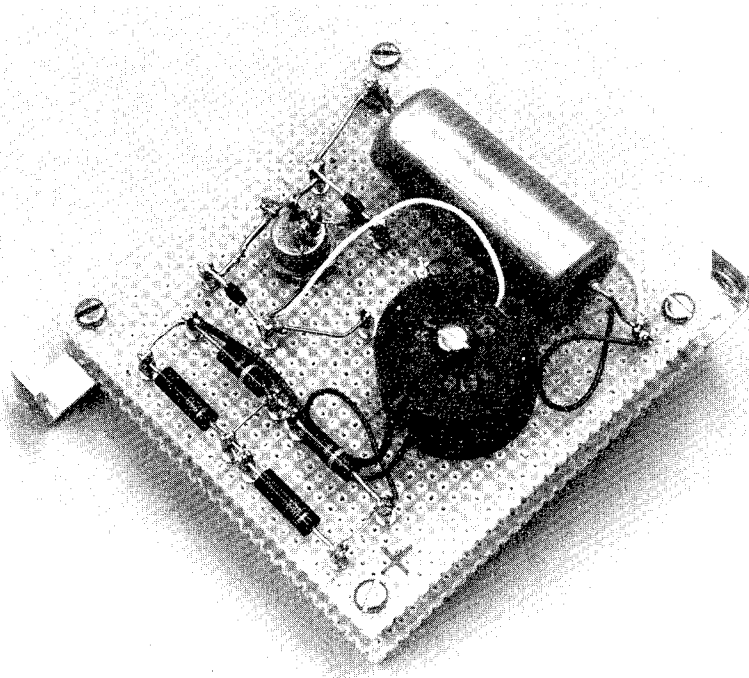


Figure 6. Experimental Breadboard Spark Exciter

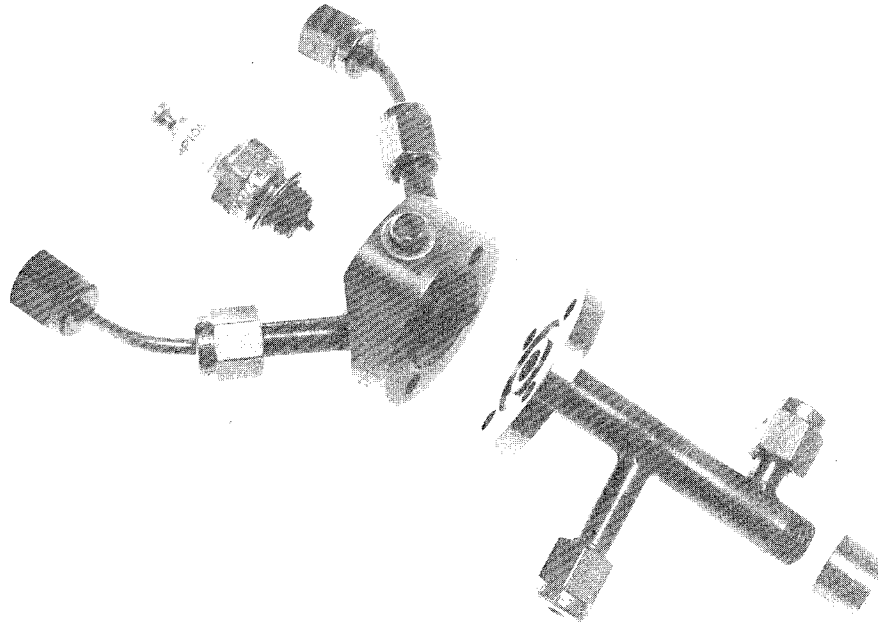


Figure 7. Standard Spark Plug Ignition Test Configuration

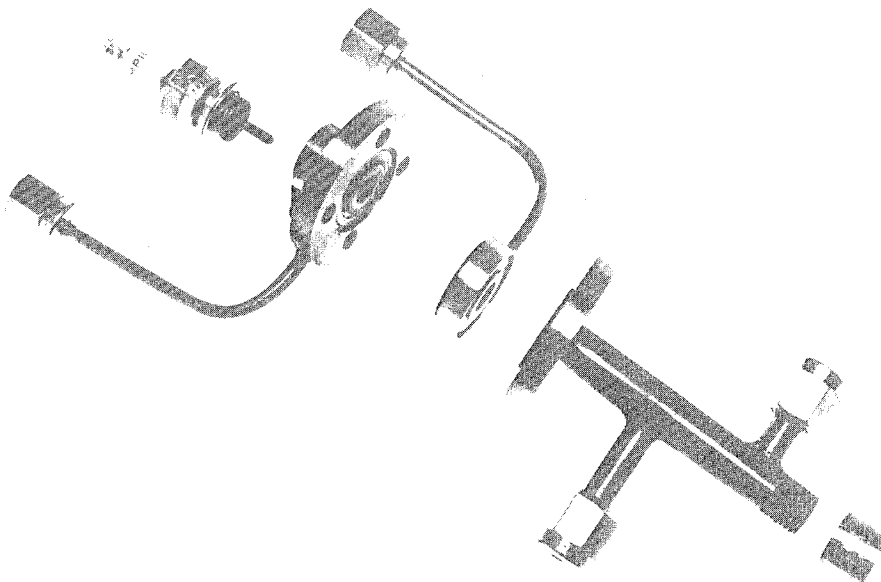


Figure 8. Modified Spark Plug Ignition Test Configuration

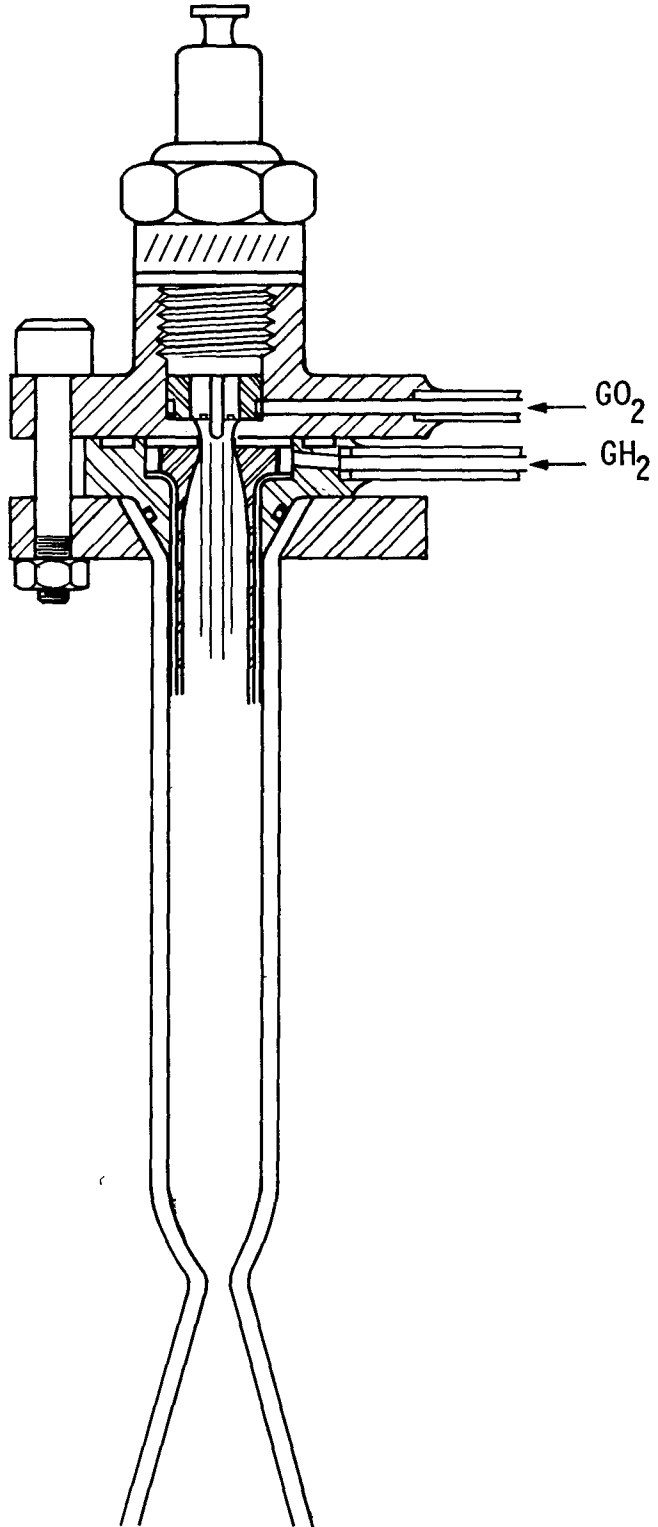


Figure 9. Schematic Drawing of 2.22-N (0.50-lbf) Thrust G0₂/GH₂ Rocket Engine

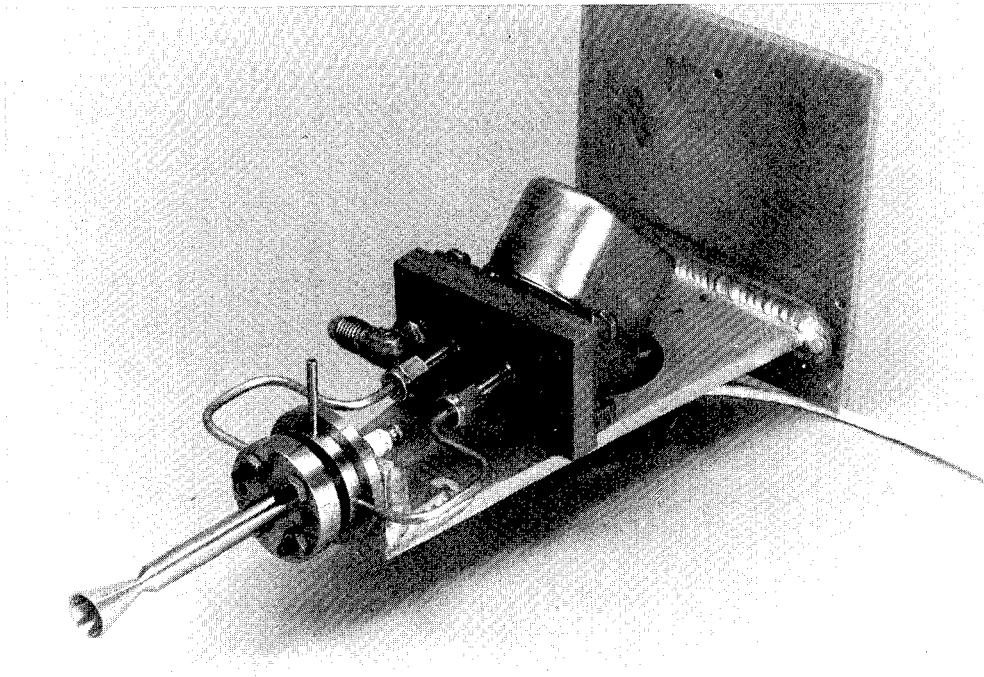


Figure 10. 2.22-N (0.50-lbf) Thrust G02/GH₂ Rocket Engine

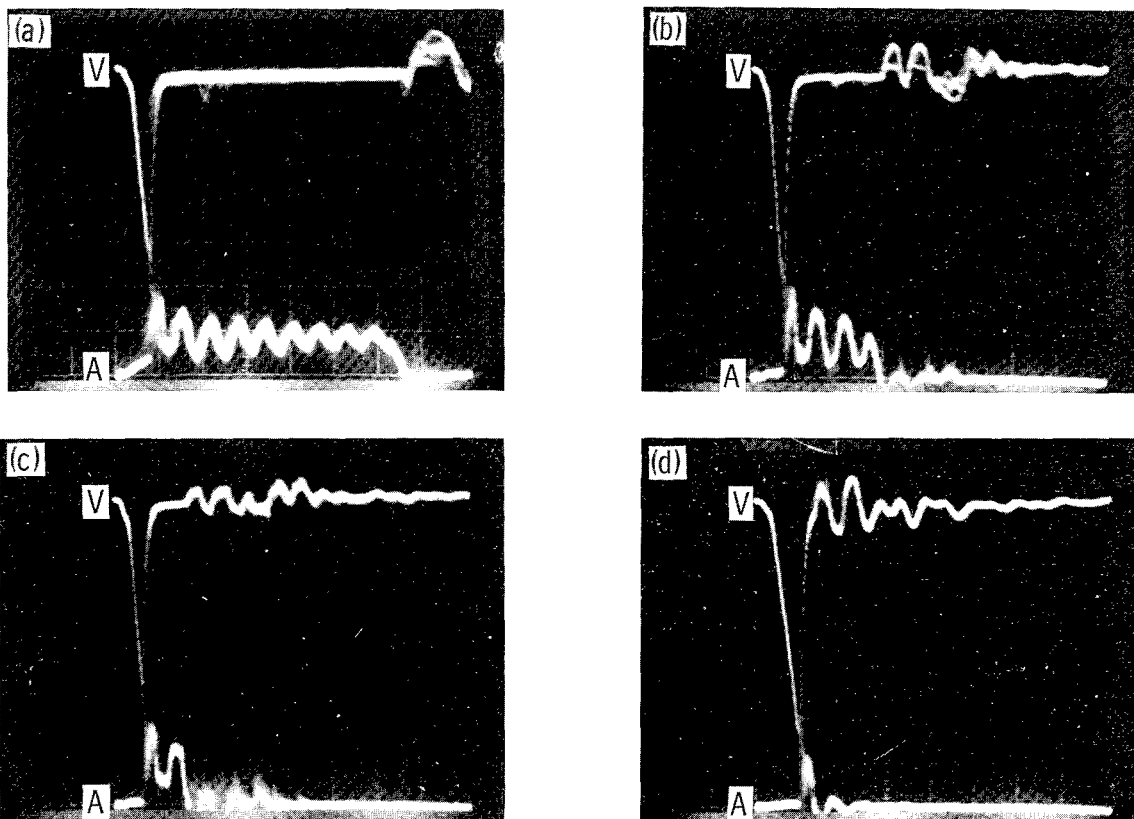


Figure 11. Typical Oscilloscope Spark Traces: (a) 10 μ s Pulse Width, (b) 5 μ s Pulse Width, (c) 2.5 μ s Pulse Width, and (d) 1.0 μ s Pulse Width. Scale Factor: Voltage 0.5 kV/Division, Current 0.1 A/Division, and Time 2.0 μ s/Division

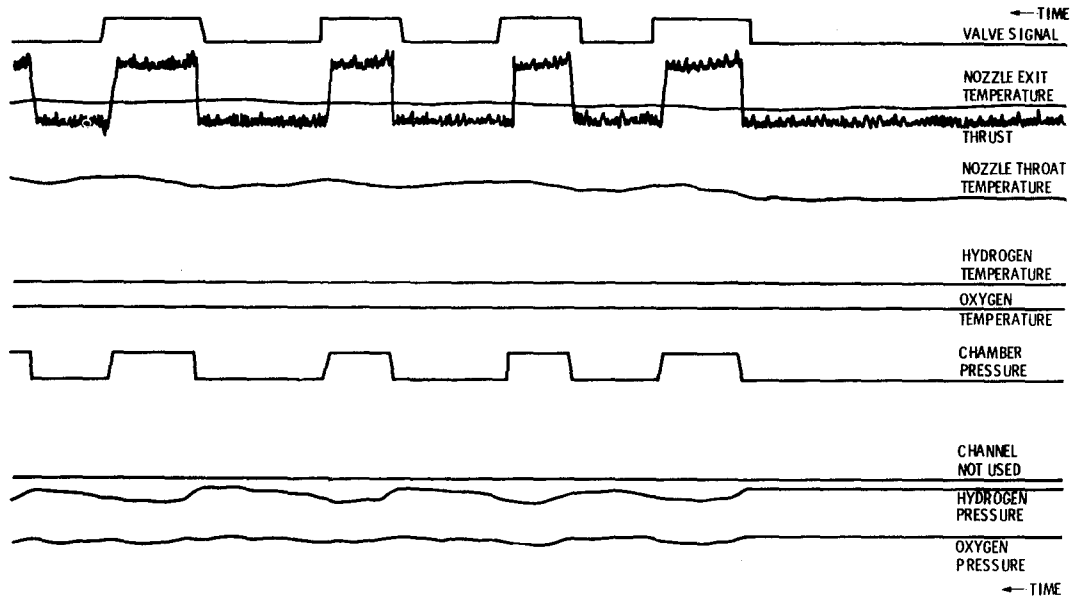


Figure 12. Typical Oscillograph Data Trace for Pulse Mode Thruster Operation in Vacuum

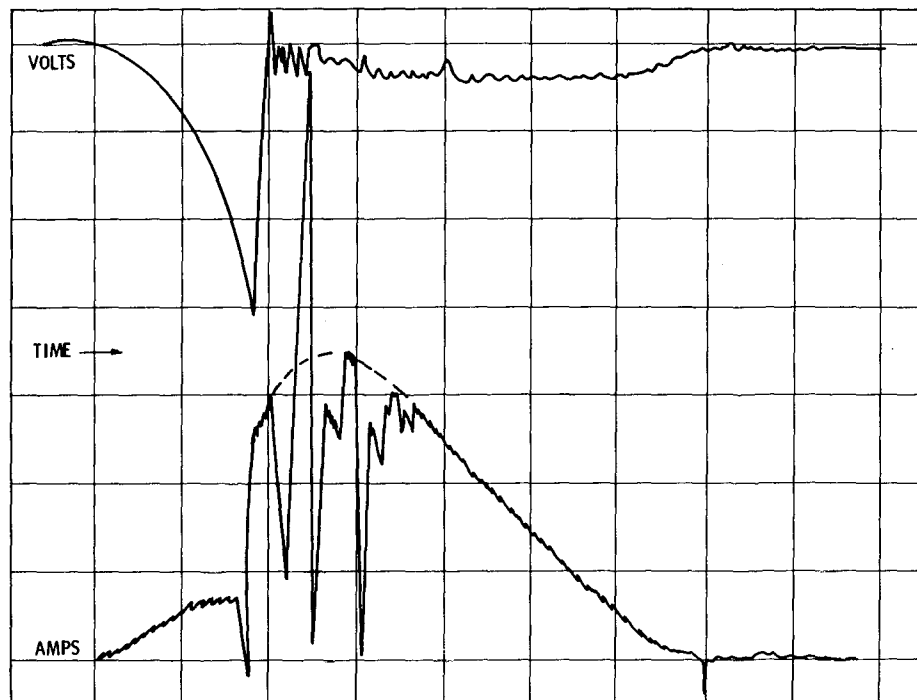


Figure 13. Typical Oscilloscope Traces for the Experimental Breadboard Exciter
 Scale Factors: Voltage 1.0 kV/Division, Current 0.1 A/Division, and Time 1.0 μ s/Division