COOLING - COMBUSTION & CHAMBER LIFE TECHNOLOGY -

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## TEST PROGRAM

### Cooling, Combustion, and Chamber Life Technology

INTRODUCTION

To be consistent with the nations space transportation systems plan for the future, a fully reusable second stage vehicle is needed in conjunction with the space shuttle. The burden of consecutive developments, i.e. both shuttle and a fully reusable 2nd stage or tug, will be eased by first using the shuttle with an interim tug, created ostensibly from existing hardware, and used expendibly. As the development burden can be assumed, late 70's, a transition to a fully reusable tug will be made to complete the stable of cost effective propulsion systems needed through the 80's. Anticipating this required development, of a fully reusable system, the present technology work is directed at preparing to help accomplish this national goal.

The tug engine has been described as small (20K), high pressure (2000 psia), pump fed, staged combustion cycle, highly versatile and efficient, and capable of operating reliably in a space environment for long periods of time and over many duty cycles. The performance goal is in excess of 470 sec. Isp. Also for application to the Space Tug vehicle, reusability for up to 50 mission's a prime objective. To meet these requirements requires a design based on extensive test experiments. The Cooling, Combustion, and Chamber Life Technology program is directed at the reusability of thrust chambers and injectors.

## OBJECTIVE

Specifically, the objective of the program is to:

Assist in developing a design analysis for long life, high heat flux, reusable thrust chambers. Chambers that have been designed using the best analytical procedures will be cyclic tested to failure. The life data will be compared with analytical predictions and if necessary, modifications to the analytical programs will be made to obtain agreement between experiment and theory. Cyclic fatigue data will be obtained at chamber pressures of 600 to 2000 psi using two different high conductivity copper base chamber liner materials.

## TEST CONFIGURATIONS

Because of size restraints, the engines proposed for the Space Shuttle and Space Tug are high pressure (Pc = 2000 psia). Their design is further complicated by the requirement of reusability up to 50 missions. Traditional thrust chamber designs such as the RL-10 with steel alloy tubes in brazed construction will not be able to withstand the increased heat fluxes of the high chamber pressure. The problem is not the overall heat load, since with $LH_2$ coolant the total heat capacity is more than adequate. The problem is with the higher temperature gradient across the metal of the combustion chamber wall. It turns out that the temperature gradient across the steel alloy tube wall is great enough to cause localized surface melting on the flame side surface while the coolant side surface is quite near the coolant temperature ($LH_2$). Clearly the high chamber pressure engines must be designed with a more conductive material for the combustion chamber wall.

Careful analysis and extensive design work have yielded a general design for the Space Shuttle and Space Tug engines. The proposed construction scheme will be a copper, or copper alloy liner with coolant passages in the form of grooves on the O.D. (The ribs between the grooves then act as "cooling-fins"). The grooves are to be "capped" or "closed out" by Electro-plating (or Electroforming) a jacket of Nickel around the O.D. The construction is then completed by adding the necessary manifolds. Even though the copper has a significantly lower melting temperature, and a lower yield strength than steel alloys, its higher conductivity more than compensates.

Proceeding to the second problem, namely the reusability, we find that the cycling of the temperature gradients (from multiple firings) causes localized thermo-stress beyond the yield strength and we get plastic deformation. The number of times that the combustion chamber can endure these plastic deformations is finite. Our task is to find how to predict how long the usefull life is and how we might increase it.

We have now described the rationale for our fatigue chamber configurations, we will now describe the configurations.

The Space Tug engine will be approximately 20,000 lbs thrust @ 2000 psia with approximately a $2\frac{1}{2}$" diameter throat. Until we get our facility uprated for higher pressure (The C of F is in Washington) our work will be at 600 psia, the facility limit. The Combustor will be of the Space Tug size though, 2.6" dia. throat, about 4,500 lbs thrust. We will investigate three combustion chamber contours. They are called "Prime", "Skinny", and "Fat". They are shown in Figure 1. We will use three materials for the inner liner. Pure OFHC copper, annealed Zirconium Copper alloy, and 1/2 Hard Zirconium copper alloy. Two coolant passage design Philosophies will be used. The first is one that is expecced to yield rather short fatigue life in order to give us experience in fatigue failures; and the second is expected to give somewhat longer fatigue life to demonstrate that we have learned what affects fatigue life and by how much. The arrangement of these three variables (contour, material, and life) for the first 8 configurations is illustrated in Table I.

The left column is to be tested at the lower chamber pressure (600 psia) and the right column is to be tested at the full chamber pressure (2000 psia). The top configuration in each column is a "long-life" design, whereas the other configurations are the "short-life". The numbers in the configuration blocks refer to actual combustion chamber numbers. 19 chambers are accounted for.

Table II shows a tabular listing of the first 17 fatigue combustion chambers. These are all to be tested at 600 psia chamber pressure. The "Design type" designation of the last column is explained in Table III which shows a list of drawing numbers and a description of each design type.

So much for the fatigue test configurations. Let us now look at test configurations for supporting research.

In order to better define the combustion chamber environment for thermo-stress analysis and life predictions we need to have specific measurements of the flame side heat transfer. For that purpose we have a series of uncooled, heavy walled, copper heat sink chambers with the three contours described previously (Figure 1). Two versions of calorimeters will be built into these chambers; one is the traditional "infinite copper rod" with thermocouples along its length, the other is the "isolation segment" made by trepanning into the copper wall of the heat-sink chamber. In addition to being used for heat transfer research the heat-sink chambers, because of their mass, are suitable for injector testing and ignitor developing. Injector and ignitor development will also be done with Ablative and Graphite engines.

One advantage of chambers of this type for injector work is that even minor injector variations are prominantly shown in the char and erosion patterns of the chambers.

The injector that is needed to run a fatigue program must be:

1. High performance
2. Uniform heat distribution.
3. Stable combustion.
4. Long lived.
5. Constant with age.
6. Reliable

In order to find an injector for the job, two configurations will be investigated; "The concentric tube", and the "all flow thru". The "concentric tube" will be first tested with a copper heat sink face, and later with a porous regimesh face. The injectors will be first tried with flourine ignition and later with electrical ignition. A final variable will be an attempt to ignite the chamber by "back-lighting". This will be to remove the ignitor from the injector and place it at the chamber exit.

IV. TEST FACILITY

The above program is primarily a rocket thrustor testing program, and as such properly belongs in the RETF (So-40). The present facility limit for chamber pressure is 600 psia, so the preliminary work will be at a thrust level of 4500 lbs thrust. When the higher pressure capability becomes available (some time in 1977), we will go to approximately 2000 psia chamber pressure and 20,000 lbs thrust. The discussion in this test program will limit itself to the 600 psia chamber pressure work.

The propellants are Ambient temperature gaseous hydrogen for the fuel, and liquid oxygen for the oxidizer. The fuel is stored in the facility high pressure gas storage bottles at the present pressure limit, while the LOX is stored in the LOX Tank submerged in a bath of Liquid Nitrogen, located in the facility OX pit. The coolant is Liquid Hydrogen stored in the $LH_2$ dewar outside of the RETF thrust stand.

The propellant and coolant flows are controlled by several closed-loop servo controllers. One loop modulates the ox fire valve to maintain and/or vary chamber pressure according to the program timer. Another loop modulates the fuel fire valve to maintain constant O/F ratio. A third loop modulates the coolant supply valve to maintain constant coolant flow, while a fourth loop modulates the coolant back pressure valve to maintain constant coolant pressure at the exit of the engine regardless of whether its firing or not.

Figure 2 is an instrumentation schematic showing the flow scheme for the propellants and coolant along with the location of all the facility instrumentation. Shown here also is the electrical ignitor with its supply of gaseous Hydrogen and Oxygen along with the instrumentation to measure its performance.

The Data from the Instrumentation is digitized by a high speed digitizer and recorded on magnetic tape at a rate of 5000 words per second. The magnetic tape is then played into the 360 computer and our data calculated and typed back at 1/10 sec. intervals with 5 point smoothing.

The Total Data Recorded is shown in Appendix A which is the instrument sheet for our 100 word program No. 621. Listed here are also the parameters on the actual Engine that are recorded. The engine instrumentation is described more fully in Appendix B which is the instrumentation on Engine 40. This is a typical set-up. Appendix B also shows the location of this instrumentation on a plot of circumferential location versus axial length. The Engine instrumentation planned for future engines is shown in Appendix C which is a List of the next 13 engines.

NDE-Non Destructive Evaluation

The reasons for performing the nondestructive tests to be described later are as follows:

1) To document, during fabrication, particular characteristics which might later contribute to an understanding of the chamber operation. This information may be both qualitative and quantitative.

2) To document, at mid-life points, particular characteristics that may lead to an ability to recognize an incipient failure.

The NDE steps that are planned during the fabrication of the chambers are listed in Table IV. The table includes other information as well to serve as a chronological reference. Item 4 from the table, metalographic analysis, includes (1) polishing, (2) etching, (3) determining ASTM grain size, (4) hardness and (5) microphotography of a slab of material cut from each forged billet from which a liner is to be hogged out. A similar analysis is done on the excess stock produced with each spun liner (item 12). Also, after spinning but before grooving an ultrasonic map of each liner is made (item 10). A complete liner inspection is made including (1) location of throat (2) wall thickness at several axial stations, (3) diameters, and (4) rib and channel widths and heights. The combination of Ultrasonic C-scan and Veeco (items 30 and 32) are used to determine the degree of bonding between the ribs and the nickel closeout.

By performing an acoustic emission test simultaneously with the first high pressurization of the coolant passages, a base line acoustic output is determined (item 49). A repeat of this test after the instruments are installed (item 60) establishes the baseline for any instrument movement. Similarly, baseline data are taken for the Profile Map (item 51) and the Eddy Current (item 53).

After hot firing of the engine begins, the NDE steps may be divided into two types. One type includes those processes which can be done with the engine remaining on the test stand. The other type requires engine removal. These are listed in Table V where some typical planned NDE intervals are also shown. This table will be discussed further in the following section. By carefully taking data from the variety of instruments mentioned above it is hoped that a process may be developed to recognize incepient fatigue cracking

<u>Test Procedure</u>

The procedure for testing is described below for the three different types of tests included in the overall program, i.e. (1) establishing an injector design that had good life characteristics and high combustion performan (2) determining the chamber contour effect on heat transfer, and (3) fatigue testing the matrix of cooled chambers.

<u>Injector performance</u> tests were made using heat sink chamber with either graphite or ablative liners. All the tests were made at 600 psia chamber pressure using liquid $O_2$ and ambient temperature $GH_2$. Using flourine IGNITION, the combustion performance, physical integrity, and circumferential uniformity were determined for both the concentric tube design and the all flow thru design. The performance was determined by making short (1-5 sec) tests using graphite and by using thrust and flow rate data to determine performance.

The circumferential uniformity was determined by making longer (15-20 sec) tests using ablative chambers. Also $H_2O$ flow tests of the LOX tubes were made to support the ablative tests. After selection of the all flow-thru design, electrical ignition through the injector face was added. Two different types of starting sequences were then established. First, very fast ramps (approx. 45 msec) to 600 psia chamber pressure for use with the heat transfer tests and second, a quick ramp to approximately 300 psia then a 1.5 second slow ramp to 600 psia for the fatigue testing (see Fig. 3)

Heat transfer testing used the first starts technique and was done at 600 psia chamber pressure. Because of the nature of the infinite copper rods to deteriorate with succeeding tests, no attempt was made to obtain da with a range of operating variables. Only one O/F and Pc was tested. With each of the 3 different contour configurations approximately 6 tests were made to obtain statistical data. By making some tests with $F_2$ ignition as well, an attempt was made to determine the possible effect of fluoride coatings on the heat transfer characteristics.

Heat transfer tests using the isolation segment chamber will be made at Pc's of 200, 300, and 600 psia and at an O/F of 6.0. The start transient will be very fast as in the other H.T. tests. Two different chamber contours will be tested - fat and prime.

Fatigue chamber testing is done using the all flow thru injector with electrical ignition. The injector flows are $LO_2$ and $GH_2$. The chambers are separately cooled using $LH_2$. This coolant hydrogen is then burned off through a torch on the scrubber stack.

The test cycle is shown in Figure 3. The initial test with each chamber is to be a single cycle. After confirming the integrity of the chamber, multiple cycle tests will be conducted. NDE proceedures will be stratigically interspersed with the testing. Table (V) shows a plan for test and NDE intervals for 3 like chambers. The first chamber (#40) will require approximately 5 weeks to fully fatigue at the expected life of 50 cycles. Shown on the table are NDE proceedures (type A) that can be done with the engine on the stand on alternate days with testing. Also shown are NDE proceedures (type B) which require approximately 3 days with the engine removed from the stand. Type B NDE are only planned once for each chamber. By comparing the plans for chamber #50 and #51 with #40 it can be seen that the type B NDE is planned for different points during the anticipated life of the chambers. It can also be seen that the length of time required to fully fatigue a chamber is reduced with successive copies of like chambers by running longer cycle trains (up to the facility limit of 20).

The chamber life comparisons that will be possible with the planned test matrix are shown in Table VI. The effect of chamber contour on life will be made between the "fat" countour and the "prime" contour. The chamber numbers required to make this comparison are shown in the table. Duplicate engines provide statistical support for the conclusion. Two different material effects comparisons are planned - OFHC cu with annealed Zr-Cu and annealed Zr-Cu with 1/2 hard Zr-Cu. The appropriate chamber numbers are shown in the table. The effect of wall temperature on life for large changes in temperature are planned by comparing the "short life" and "long life" chambers. At the expense of statistical data for the above comparisons smaller changes in wall temperature could be made by adjusting the coolant flow rate to an off design value and then fully fatiguing the chambers. As mentioned in the table the effect of grain size on life can also be noted but with only one large grain chamber planned.

The order in which the chambers are planned to be fabricated is shown in Table VII. Comparison on the table with Table VI shows that at least one chamber from each part of the test matrix will have been tested after the first 7 chambers are fully fatigued.

VIEWPOINT

This Program constitues a major portion of our Nations future technology base for Rocket Engine Reusability. Along with complementing research performed by other project groups, namely the Quentmeyer-Kazaroff program of fatigue of plug configurations and analytical work by Price-Kasper and CONTRACT work ~~on contract to~~ by Mar-Test Corporation and the Boeing Corporation, our work as described in the preceeding sections will contribute significantly to an understanding of the fatigue problem.

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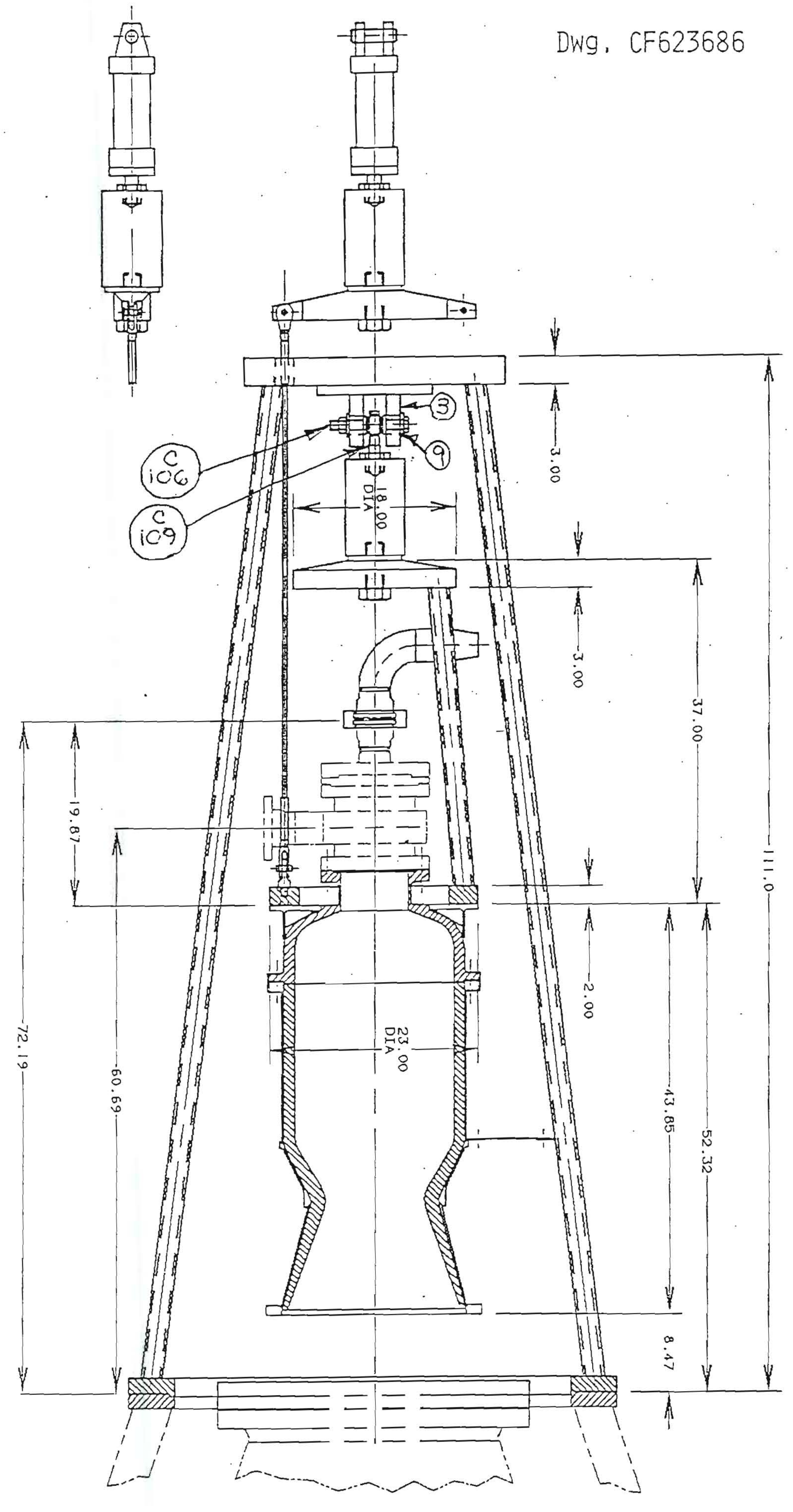

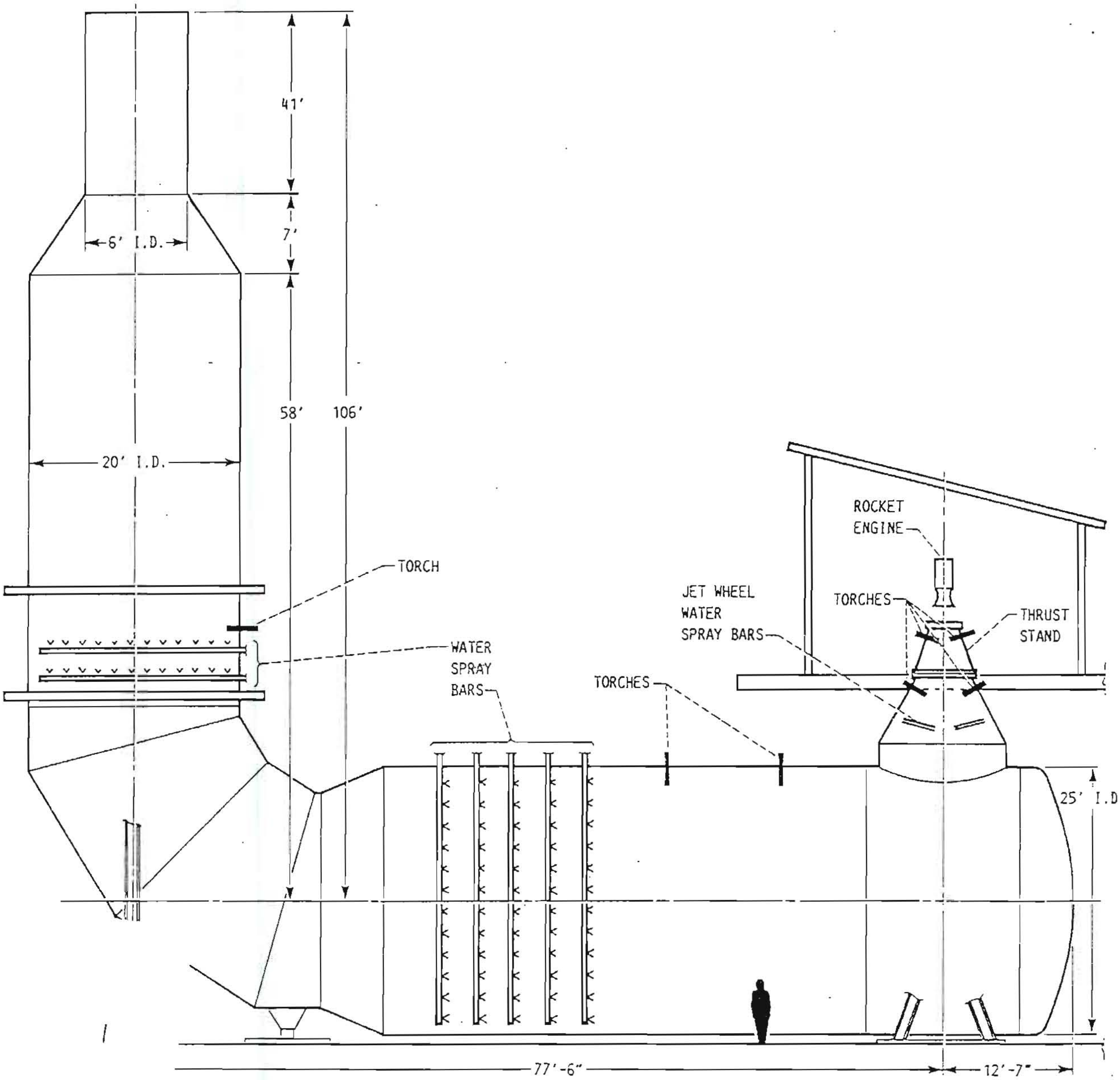


FIGURE 1.- ROCKET ENGINE TEST FACILITY SCRUBBER EXHAUST DUCT.