# NASA TECHNICAL MEMORANDUM

NASA TM X-52224

N67-16041

(ACCESSION NUMBER)

55

(PAGES)

TMX-52224

(NASA CR OR TMX OR AD NUMBER)

FACILITY FORM 602

(THRU)

1

(CODE)

11

(CATEGORY)

NASA TM X-52224

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GPO PRICE $ ____

CFSTI PRICE(S) $ ____

Hard copy (HC) 3.00

Microfiche (MF) .65

ff 653 July 65

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LECTURE presented at the University of Tennessee Space Institute
Tullahoma, Tennessee, January 31 - February 10, 1967

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION · WASHINGTON, D.C. · 1967

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# PROPULSION SYSTEM DEVELOPMENT

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E-3791

# GROUND TEST FACILITIES FOR AIRCRAFT AIRBREATHING PROPULSION SYSTEM DEVELOPMENT

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## INTRODUCTION

This report constitutes the lecture on the types of propulsion system facilities required for the development and evaluation of advanced airbreathing engines. The lecture was presented by the author at the University of Tennessee Space Institute, January 31, 1967. This is one of the thirty-two lectures comprising a two-week course in Aerospace Test Facilities offered by the University of Tennessee in cooperation with the Arnold Engineering Development Center for selected Air Force and civilian personnel.

The steady progress of powered flight has necessarily followed closely the development of suitable aircraft powerplants. Without a lightweight but adequately powerful engine, controlled flight of sufficient distance to serve a useful purpose would not be possible. In Germany Dr. Otto created the four stroke internal combustion engine in 1876, but it was not until twenty years later that Daimler was able to perfect the eight horsepower engine which enabled the Wolfert "Deutschland" to make the first gasoline powered dirigible flight. Wilbur and Orville Wright had to develop their own engine before they could achieve successful flight at Kitty Hawk in 1903. Later, Glenn Curtiss met with outstanding success due largely to the engines which he was instrumental

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in developing. The pattern of aviation history has thus evolved and logically larger and more sophisticated engines have lead to increased flight speeds and altitudes, and the development of highly refined aircraft.

As powerplants become more complex along with the aircraft in which they are used, ground test facilities assume a greater primary support role for the development of these propulsion systems. With proper facilities many difficulties of altitude and high speed flight operation may be solved that would otherwise require extremely costly, hazardous, and time consuming flight tests.

In addition to these obvious development aspects relative to current engines, investigations in test facilities provide a wealth of research data and design information for future engines. Various individual component improvements can be integrated into a complete propulsion system, and the system and the interaction effects of its different components can be evaluated under the actual engine conditions corresponding to flight. During the course of the evaluations, many unanticipated research problems are revealed, and immediate solutions or information leading to their definition and eventual solution may be obtained.

The test that follows leads into the description of two major facility types, altitude test chambers and supersonic tunnels, and several component test facilities. The discussion will consider the following four general areas: (1) types of propulsion systems, (2) typical problem areas for experimental programs, (3) performance variables and Reynolds number index, and (4) facility details, supporting systems and their use.

The assistance provided by members of the staff of the NASA Lewis Research Center in the preparation of this paper is gratefully acknowledged.

# PROPULSION SYSTEMS OF CURRENT INTEREST

## Reciprocating Engine

One of the earliest of the airbreathing engine family, the reciprocating engine, probably reached its peak of development activity during the 1940's as dictated by the needs of World War II. It seems safe to say it has reached its ultimate in size and horsepower and certainly will long be with us as the work-horse for low and medium altitudes and air speeds. Product improvement and general design refinements will certainly continue throughout its life span.

For the purpose of this discussion it will not be considered in describing test facility requirements.

## Turbojet

From near the end of the war until now the gas turbine engine has had a quarter century of development in this country into a complex propulsion system with continuously broadening applications. These applications run the gamut of flight speed from 0 to Mach 3 and altitudes from sea level to near 100,000 feet.

The basic turbine engine, shown in figure 1, consists of an air compressor, a combustion chamber in which heat is added to the compressed air, a turbine through which the hot compressed gas is expanded partially to drive the compressor, and an exhaust nozzle to complete the expansion of the gas and accelerate it rearward at high velocities. Thrust is produced when the velocity of the gas leaving the nozzle exceeds the velocity of the air entering the inlet.

Several variations of the basic turbine engine are shown in the following figures. Figure 2 shows an afterburning turbojet. Inasmuch as about 75 percent of the air leaving the turbine is still available for combustion the addition of fuel in the engine tailpipe can provide a total thrust increase of approximately 50 percent or more. Until the advent of the present development activity for the commercial supersonic transport this technique has been limited solely to military use because of weight and noise penalties.

In figure 3 there is shown still another variation of the turbine engine known as the turbofan. For subsonic propulsion the interest in the turbofan engine stems from its improvement in propulsive efficiency due to lower discharge velocities with subsequent improvement in specific fuel consumption. There are a number of possible configurations for a turbofan each with certain advantages and disadvantages. The fan, for instance, may be mounted at the front end of the compressor as shown or at the aft end of the engine, as a part of the turbine. In the case of the forward-fan engine, the exhaust from the fan may be ducted overboard or may be ducted along the outside of the basic engine to mix or not mix with the turbine exhaust before the gases pass through the jet nozzle. The addition of fuel to the fan exhaust in a manner similar to the tailpipe afterburner can provide additional thrust as required by the flight application. One such engine under development for the SST competition is of the duct burning turbofan type. In a like manner a fan engine may utilize the tailpipe afterburner for additional augmentation.

The turbofan engine is presently used in most all commercial transports and many military aircraft such as the F-111.

## Turboprop

Figure 4 shows a schematic of a turboprop engine. Turbojet engines do not fully expand the exhaust gases in the turbine. By adding turbine stages more energy can be extracted from the gases and used to drive a propeller. The necessary propeller reduction and drive gearing is connected directly to the compressor in one version and an alternate arrangement makes use of a free turbine to drive the propeller.

### Ram Jet

The ram jet engine shown in figure 5 is a member of a group of jet propulsion powerplants known as the ATHODYD group - Aero THermODYnamic Duct - and has seen primary use in guided missiles. This engine takes advantage of the aerodynamic ram compression of the air ahead of the cowl inlet followed by further compression in the diverging diffuser. Heat energy added from the combustion process increases the velocity and temperature of the gas entering the exhaust nozzle. The velocity of the gases leaving the nozzle is several times as high as that of the air at the inlet resulting in propulsive thrust.

The ram jet has seen extensive development in the past and is described here because of its relation to a particular testing technique known as free jet, to be described later.

## PROPULSION SYSTEM PROBLEM AREAS

Why has time been spent in a simplified discussion of these engines? Such a primer of information is intended only as a reminder of the variations and complexities of airbreathing propulsion systems and further emphasize the need for ground test facilities and associated unique testing techniques.

In order to better understand some of these system complexities it is felt that a look at some of the typical research and development problem areas is desirable. Accordingly the discussion to follow is outlined in figure 6. With this problem area insight and the subsequent treatment on performance parameters it is hoped that the facility descriptions can be more meaningful.

## System Performance Evaluation and Calibration

Tests performed for this purpose are required to validate the design guaranteed operating points and as such require proper test simulation of pressure ratio, temperature ratio, and Mach number for the selected conditions.

## Altitude Operating Characteristics

The tests conducted on an engine or propulsion system within this category are aimed at providing the basic pumping characteristics of the engine, altitude starting characteristics, and safe operating limits. These limits are determined by such phenomena as compressor stall or combustion blowout.

Starting an engine at altitude requires that (1) ignition in the combustors containing sparkplugs or other ignition devices be accomplished, (2) the flame successfully propagates to the other combustors, and (3) the engine accelerate from the starting speed to maximum speed without encountering combustion blowout or compressor stall and without exceeding allowable temperature limits.

There are three factors that should be examined when considering ignition characteristics of a turbojet. These factors are: fuel vaporization, sparkplug location, and spark energy. Some data obtained in altitude ignition investigations are presented in reference 1.

It is known that as fuel volatility, fuel temperature, or air temperature are reduced the ability of the fuel to quickly and effectively vaporize and mix with the air also diminishes. Accordingly it would be expected that as the fuel temperature or the fuel volatility are decreased ignition would become increasingly more difficult. These expected trends are verified by extensive experimental data. The magnitude of these effects may vary greatly depending on the combustor design. The placement of the sparkplug and provision of sufficient energy can alleviate the problem to some degree.

Proper fuel injection patterns along with the sparkplug located so as not to be in the liquid stream or in too rich or too lean a region are likewise important criteria. The sensitivity of one engine to spark gap immersion is shown in figure 7. These data show that as the spark gap was moved toward the center of the combustor, which for this engine was the region where the fuel was most easily ignitible, the altitude ignition limits vastly improved.

Without going into a detailed explanation, suffice it to say that for reliable ignition at altitude the factor of greatest importance is the energy dissipated per spark rather than the total energy per second.

As altitude is increased the acceleration of an engine from the windmilling speed becomes more of a problem because the airflow through the turbine becomes correspondingly lower and the power available to accelerate the turbine is reduced. Since the inertia of the rotating parts is constant the reduction in gas flow through the turbine results in reduced excess energy available for acceleration. Thus in many cases especially at low flight speeds, a greater length of time is required to accelerate an engine over a given speed increment at altitude than is required for the same speed increment at sea level. Of course variable geometry such as variable exhaust nozzle, inlet guide vanes, compressor stators, or interstage bleed help to alleviate the situation.

To illustrate the altitude starting picture, figure 8 shows the altitude acceleration limits compared with ignition limits using a high energy ignition system. It is noted that although ignition is obtainable at high altitudes for all flight Mach numbers, the maximum altitude at which a successful start can be obtained is greatly reduced at low Mach numbers due to the inability to accelerate the engine.

## Afterburner Development

Afterburner problems involve both performance and operational aspects such as internal aerodynamics, combustor performance, starting, cooling, etc. A sketch of an afterburner is shown in figure 9 with some of the principle problem areas indicated.

The diffuser aerodynamics have an important effect on afterburner performance. Steep velocity gradients at the diffuser exit result in high local velocities with attendant low combustion efficiency, high pressure losses, and reduced operating limits. Furthermore, diffuser flow separation accompanied by burning in the separated region along the diffuser wall can produce low frequency pressure oscillations.

Ignition problems are those of achieving reliable ignition for restarting up to the altitude operating limits of the afterburner. The combustion problem is a major one and probably accounts for the most total effort in afterburner research. Some of the principal factors investigated are flameholder geometry, fuel injector design, burner length, and burner inlet velocity. The objective of investigation of these factors is to obtain combinations giving high combustion efficiency over a range of altitudes while avoiding combustion screech or high frequency oscillations.

Because of the high temperatures present in afterburners, cooling and durability are important problems. Some cooling can be provided by passing gas at the turbine-outlet temperature along the wall of the afterburner or through an internal liner. Some external air flow may also be needed particularly for the nozzle. When ejectors are used for pumping this air, attention must be given to minimizing thrust losses.

Some representative experimental results from an afterburner investigation are shown in figure 10. Combustion efficiency is plotted as a function of afterburner fuel-air ratio for a range of burner inlet total pressures. The corresponding altitudes for the particular engine used at a flight Mach number of 0.9 are also listed. The results show two typical characteristics: decreasing efficiency with decreasing pressure, and decreasing efficiency with both increase and decrease in fuel-air ratio from a narrow optimum range. Detailed treatment of a variety of afterburner research problems is given in references 2 to 5.

## Compressor Inlet Flow Distortion

Engine inlet airflow distortion is a problem that came into existence with the trend in engine design toward higher pressure ratio compressors and the accompanying advance toward higher flight speeds. As stages have been added to the compressor to raise its pressure ratio, the requirement for stage matching over a wide speed range has moved the operating points of some stages to precariously high loadings (i.e., increased specific work per stage). As a result compressor performance and compressor stall have become correspondingly more sensitive to inlet flow distortion. At the same time the trend toward higher flight speeds has led to such aircraft drag reduction techniques as using small radii on the lips of inlets. With the flow separation that occurs at the lower lip during high angle of attack operation when climbing or maneuvering, inlet flows provided to the engine have become more distorted.

The load distribution within the compressor shifts with engine speed. At low speed the front stages of the compressor are more highly loaded and thus closer to stall, whereas the rear stages are more highly loaded at high engine speeds. When any one stage of a compressor stalls, its portion of the overall compressor pressure rise tends to be shifted to the other stages, thus upsetting the load distribution and generally resulting in breakdown of compressor flow. In some case of compressor stall a severe cyclic pressure fluctuation occurs within the compressor as it intermittently stalls and recovers from the stalled condition.

It may be generally concluded that a compressor that has any of its stages operating close to stall will be sensitive to inlet flow distortions. Thus a high pressure ratio compressor, which is seldom able to maintain a uniform stage loading distribution throughout over a wide range of engine speed, will be more sensitive to inlet flow distortions than a low pressure ratio compressor. The fewer stages of the low pressure ratio machine generally allow a more uniform loading among stages over a wider speed range and a corresponding decreased sensitivity to inlet flow distortion. Variable compressor geometry is beneficial in making the engine less sensitive to flow distortion, although with some penalties in specific fuel consumption.

Detailed information on inlet flow distortion effects is contained in references 6 through 8.

## Engine Dynamics and Controls

Full scale altitude test facilities have been a particularly powerful research tool in the field of engine dynamics and controls. Such facility testing has permitted the use of the extensive and specialized instrumentation required for adequate controls investigations.

Among the many dynamics and controls problems are those related to (a) engine starting and acceleration, (b) the avoidance of operating limits such as combustor blowout and compressor stall, (c) protection of the engine against damage by overtemperature and excessive stress, (d) maintenance of optimum conditions of operation, and (e) achieving stable operation and rapid response to changing load demands.

Basic to the problem of system stability and response is a knowledge of the dynamic characteristics of the engine. These dynamics are important because the control is constantly responding to deviations in a control variable and acting to return the variable to its desired value. A turbojet engine is essentially a linear device and as such its dynamic characteristics may be conveniently described in terms of time constants. One such time constant is that which relates the transient engine speed to a change in fuel flow. This time constant is the time required for the speed to reach 63 percent of its final value with a step change in fuel flow imposed on the system.

An example of a problem in high altitude system instability is presented in figure 11. Transient thrust, engine speed and turbine inlet temperature are shown as a function of time. The transient imposed on the engine is that of a throttle change from full nonaugmented thrust to full afterburning thrust - an increase of some 50 percent. Although the control provided rather satisfactory speed and temperature control, the system broke into severe thrust oscillations after some 7 or 8 seconds. As evident in the top trace, thrust oscillation from the full dry to full afterburning occurred with about a 2-second period despite the fact that this engine was very stable at sea level and low altitudes. The source of the oscillatory behavior was found to be in the changing dynamic characteristics of the engine with increasing altitude and associated interaction effects among the various control system loops. Based upon the knowledge obtained by such tests together with analog computer analyses, the control system was redesigned and stable operation at high altitude achieved.

An extensive amount of dynamics and control investigation have been carried out as typified by the references 9 through 15.

### Inlet Pressure and Temperature Transients

Some years ago, several jet fighter aircraft experienced engine difficulties during armament firings. These difficulties were eventually identified as compressor stall phenomena associated with temperature and pressure transients resulting from ingestion of high temperature or high pressure gases in the engine inlet. Extensive investigation in altitude test facilities of this problem and closely related effects of diffuser buzz and rapid maneuvers have provided a fairly good understanding of the details of the compressor flow dynamics and overall engine behavior.

Discussion of inlet pressure and temperature transients is contained in reference 16.

After this brief review of various propulsion systems and some typical problem areas encountered in their development let us turn briefly to performance standardization and the use of Reynolds number index.

## GENERALIZED PERFORMANCE VARIABLES

An aircraft gas turbine engine is called an "airbreathing" engine because most of the work in the engine is done by air. Consequently any change in the state of the air entering the engine, or at the exhaust, results in a proportional change in the work done internally and in the engine's power or thrust output. The number of molecules per cubic foot of air (which is determined by both pressure and temperature) is a direct measure of the working substance being given the engine. In addition, the air temperature has separate effects on compressor and turbine performance which are related to the Mach number at which these units are operating. Because engine performance specifications are normally based on standard atmospheric conditions, corrections must be made for any deviation from these conditions when engine performance is checked or compared with a standard.

Standard correction factors are shown in figure 12.

$$\delta = \frac{\text{Observed absolute pressure}}{\text{29. 92 Hg (or 2116 PSF)}}$$

$$\theta = \frac{\text{Observed temperature } ({}^{\circ}\text{R})}{519^{\circ}\text{ R}}$$

The figure indicates the manner in which these corrections are applied to basic measurements customarily taken during engine performance testing.

$$\text{Airflow} - \frac{W_a\sqrt{\theta}}{\delta}$$

$$\text{Velocity} - \frac{V}{\sqrt{\theta}}$$

$$\text{RPM} - \frac{N}{\sqrt{\theta}}$$

$$\text{Thrust} - \frac{F}{\delta}$$

$$\text{Fuel flow} - \frac{W_f}{\delta\sqrt{\theta}}$$

$$\text{Pressure} - \frac{P}{\delta}$$

$$\text{Temperature} - \frac{T}{\theta}$$

## PERFORMANCE GENERALIZATION USING REYNOLDS NUMBER INDEX

Altitude test facilities have been used extensively to obtain engine performance calibrations and operating limits over the ranges of flight speeds and altitudes to be encountered in service. Such investigations could involve an excessive amount of testing if data were obtained at all combinations of flight altitudes and Mach numbers of interest. Fortunately the amount of required testing can be considerably reduced through use of generalization procedures involving pressure, temperature, and Reynolds number. In practice it has been found convenient to include Reynolds number effects on engine performance through use of a so-called Reynolds number index defined as the ratio of Reynolds number of the compressor-inlet flow at flight altitude and Mach number to the Reynolds number of the compressor-inlet flow at standard sea-level conditions. This index can be reduced to the expression

$$\frac{\delta}{\varphi\sqrt{\theta}}$$

where $\delta$ is the pressure ratio, $\varphi$ is the viscosity ratio, and $\theta$ is the temperature ratio of the altitude to sea-level standard conditions. The manner in which this Reynolds number index varies with altitude and Mach number is shown in figure 13.

If engine tests are conducted at a reasonable number of Reynolds number indices, the data can be used to construct complete engine performance maps for wide ranges of altitudes and Mach numbers. Those performance variables associated with fuel flow do not generalize with Reynolds number index because of the dependence of combustion efficiency on chemical kinetics. Reasonably good generalization of combustion efficiency has, however, been obtained with the product of airflow and turbine-outlet temperature which can be related to the fundamental velocity-temperature-pressure relationship involved in combustion kinetics equations.

In using Reynolds number index technique to generalize over-all performance, the engine pumping characteristics are plotted as, for example, in figure 14. Here total-pressure and temperature ratios are plotted against corrected engine speed for a number of Reynolds number indices. The corrected engine airflow is similarly plotted in figure 15, and combustion efficiency is plotted against the product of airflow and turbine-outlet temperature in figure 16. With these three figures (figs. 14, 15, and 16) and a suitable combustion chart that relates temperature rise to fuel-air ratio and the relation of Reynolds number index to altitude and Mach number shown in figure 13, conventional engine performance curves can be computed for the desired range of altitudes and flight speed. A typical plot of engine performance thus obtained for one altitude is shown in figure 17.

More detailed discussion and experimental results of engine performance are given in references 17 through 20.

## FACILITY TYPES AND DETAILS

With this background discussion on engine types, propulsion system problem areas, methods of performance data standardization and generalization let us now turn our attention to the specific area of ground test facilities.

This discussion will center primarily around facilities for full scale engine testing, namely altitude test chambers and supersonic wind tunnels. Information on facilities for separate testing of components such as the compressor, combustor, turbine, and exhaust nozzle will be covered briefly.

## Altitude Test Chambers

Altitude test chambers are herein construed as test chambers and associated compressor and exhauster equipment that provides the engine inlet and exit conditions corresponding to the stagnation and ambient conditions, respectively, of high speed, high altitude flight. Such facilities accurately simulate the internal flow conditions encountered by a propulsion system in flight but do not provide external flow simulation. Two engine testing techniques that have been used in such facilities are direct-connect testing, common to jet engine tests, and free jet testing, primarily used for ram jet tests. A discussion of both techniques will follow.

### Direct Connect Testing Techniques

Inlet ducting. - A photograph of a typical turbojet engine installation in an altitude chamber is shown in figure 18. A schematic of this installation is shown in figure 19. These figures show the engine to be mounted on a bedplate supported by flexure rods. A duct extends forward of the engine through a bulkhead with a bellmouth mounted on the end of the duct. The pressure and temperature corresponding to the desired altitude flight condition are established in the forward compartment of the test chamber. Air flows from this compartment through the bellmouth and inlet duct to the engine. The engine exhausts into the aft portion of the chamber where the pressure is maintained at the value corresponding to the test altitude. The terminology "direct connect" stems from this inlet duct to front bulkhead configuration.

Thrust system. - Engine thrust is determined from a complete force balance on the engine and inlet duct system. The pressure forces on the engine and duct, both internal and external, and the engine thrust are transmitted through the bedplate to a strain gage load cell anchored to the test chamber. A labyrinth seal maintains the ram pressure difference across the front bulkhead of the test chamber. These seals are usually installed in a reduced area of the duct to lower the magnitude of the inlet pressure force and to increase the air velocity at the flow measuring station.

Pressure control system. - The test chamber installation described is used primarily for most all of the so-called steady state types of testing such as performance calibrations, altitude starting, and altitude operating limits. However to extend the range of facility usefulness to include transient tests such as engine acceleration and stall, a high response rate valve is located in the forward bulkhead. This is shown schematically in figure 20. The rapid action of the bulkhead valve maintains constant inlet pressure while allowing the slow acting facility air and exhaust supply systems to remain at essentially constant flow conditions during changing engine requirements.

To accomplish the high response rate required of the bulkhead valve it is necessary to have a low inertia valve design as well as a high torque electro-hydraulic servomotor (ref. 21). The valve position responds to a signal from a total pressure probe in the engine inlet ducting. This signal is compared to a reference pressure which is preset to the value of the desired inlet pressure. The error signal proportional to the difference between inlet pressure and reference pressure produces the required change in valve position to maintain nearly constant pressure during engine speed transients.

Inlet flow distortion. - As mentioned previously, inlet flow distortion is one of the typical problem areas often investigated in ground test facilities. This discussion was intended to provide a basis for understanding the sensitivity of modern high pressure ratio compressors to inlet flow disturbances during high angle of attack operation resulting from climb or maneuvering. In order to simulate the distortion pressure profiles the simplest and most conventionally used method is to install screen segments in the inlet duct. An example of screen arrangements for circumferential and radial distortions is shown in figure 21 along with some typical pressure profiles.

Pressure and temperature transients. - Operation of turbojet engines in flight can be effected by many kinds of transient disturbances. For simplicity, these disturbances may be divided into two general groups. One group represents transients which are slow enough (1.0 cps or less) to be compensated for by the engine controls and the other group is transients which are significantly faster than the engine control system. The rapid transients which occur at frequencies up to 100 cycles per second are the result of environmental changes in inlet temperature or pressure which may occur as a result of operation of variable-air inlet diffusers, diffuser buzz, flying through wakes caused by armament firing, flying through shock waves, and others.

In order to simulate pressure transients in altitude test facilities a device such as the one shown in figure 22 has been used. This pulse generator consisted of 40 small wedge-shaped ducts, alternate ones of which were supplied high pressure ram air. Air was supplied to the remainder through a large circular inner duct when the total pressure could be reduced by means of a butterfly valve.

The ducts were alternately opened and closed by means of wedge-shaped spokes on a 20-spoke rotating valve. A nearly sinusoidal pressure oscillation could be generated by rotating the valve. Increasing the speed of rotation of the valve increased the frequency of the pressure oscillation and the amplitude of the pressure oscillation was increased by closing the butterfly valve, thus magnifying the pressure difference between the inner and outer ducts.

To obtain temperature transients, an arrangement of inlet duct hardware has been used like that shown in figure 23. As can be seen in the figure a complex assembly of valves, combustor and ducting is required ahead of the engine to provide the desired temperature transient.

During normal operation the hot gases were bypassed around the engine and cold air from the test chamber flowed into the inlet bellmouth to supply the engine. To initiate a transient the hot gas valves were closed and the alternate four valves simultaneously opened. These valves, which were light in construction and controlled a relatively large flow area, were actuated by a servomotor providing a time from open to closed position of about 0.03 second.

## Free Jet Testing Technique

With the free-jet method of testing used principally in the past for ram jet engines, the inlet is submerged in a supersonic jet of air in which the Mach number, pressure, temperature, and angle of attack conditions of flight are reproduced. In this manner, a good simulation of the internal flow conditions in the inlet diffuser, and hence throughout the engine, may be obtained. No attempt is made to simulate external flow conditions around the outside of the engine.

A schematic diagram of a simple form of a free-jet installation is shown in figure 24. Air at desired stagnation conditions in the inlet section of the test chamber is expanded through the nozzle to produce the supersonic free jet. The altitude pressure and inlet-air temperature may be controlled by varying the stagnation conditions in the inlet section of the test chamber. The flight Mach number is determined by the design of the nozzle, which may be either fixed or adjustable. If fixed, a separate nozzle is required for each flight Mach number. Angle-of-attack conditions are simulated by inclining the axis of the nozzle relative to the axis of the engine inlet as shown in the figure.

The pressure ratio required to operate a free-jet facility is determined by the pressure drop across the supersonic nozzle and the pressure recovery experienced by the secondary or bypass flow that passes around the engine rather than by the pressure recovery of the engine itself. Additional equipment is, therefore, frequently used in order to increase the pressure recovery in the secondary flow and thus reduce the overall pressure ratio required. Two arrangements that have been used successfully are shown schematically in figure 25. The jet diffuser shown in the upper part of the figure consists of a divergent shroud attached to the outlet end of the jet nozzle. The annular passage thus formed around the cowl of the engine serves to position a system of shocks that provide a more complete pressure recovery than is obtained in a free-flow field around the engine alone.

The installation of a second throat is shown in the lower part of the picture. This secondary flow passage is essentially an axially symmetric convergent-divergent diffuser. The secondary air is thus decelerated into the throat, followed by a normal shock process at a relatively low Mach number in the divergent section.

A photograph of a jet diffuser installation in conjunction with a 28-inch ram jet engine is shown in figure 26.

The prime requirements in the design and operation of a free-jet facility is that the stream flow into the engine inlet be free of shocks or expansions. Principal factors influencing these conditions are the position of the engine inlet relative to the nozzle outlet, the size or flow rate of the engine relative to that of the nozzle and the pressure ratio across the facility.

Engine position. - The engine inlet must be placed so that the diffuser cone and cowl are within the cones defined by the upstream and downstream Mach lines originating from the nozzle lip. Thus the stream entering the engine is not influenced by the chamber pressure at the jet boundary so long as this pressure is equal to less than the jet pressure. The size of the jet required to test a given engine may, therefore, be reduced as the engine cowl is moved closer to the nozzle outlet. A limitation on this position is imposed by the need for a free light path for use of schlieren or shadowgraph equipment in the observation of engine-inlet shock systems.

Facility mass flow ratio. - The flow rate through the supersonic nozzle divided by the flow rate through the engine under supercritical conditions is defined as the facility mass flow rate.

The importance of this ratio is of primary concern in the design of the supersonic nozzle and the flow capacity of the basic air and exhaust systems. Tradeoffs are usually considered in which there appears to be some advantage in using a mass flow ratio larger than the minimum required for satisfactory engine inlet flow conditions. This appears to be the case when a second throat is used and results in better pressure recovery. Such tradeoffs are again dependent on air capacity and the desired range of test conditions.

Facility pressure ratio. - The pressure ratio required to operate a simple "open type" installation as shown in figure 24 will be higher than that required for a facility using a diffuser or second throat for pressure recovery.

There appears to be two limits for minimum pressure ratio of open-type facilities. If the oblique shock falls near the engine inlet, its strength must be limited to the value that permits a simple reflection from the engine cowl. If the oblique shock falls further downstream on the engine, a limiting shock pressure ratio is 2.26. Shocks of greater strength have been observed to result in a flow breakdown from the nozzle.

Pressure ratios required to operate the same facility with a jet diffuser were reduced 10 to 25 percent depending on Mach number. Still further reduction in starting and running pressure ratios are afforded by the second throat addition. Whereas the simple diffuser afforded a pressure recovery of up to 30 percent of a normal shock, the pressure recovery in the second throat is at least 50 percent of normal shock; with the adjustable area throat, the pressure recovery may be over 60 percent of normal shock.

Although the author knows of no free jet testing of a turbojet up to the present time it is presently under study by several groups and may well be a desirable technique to be applied.

A comprehensive treatment of free jet testing techniques is given in reference 22.

## Choked Nozzle and Exhaust Diffuser Technique

The description of direct connect engine testing in an altitude chamber indicated that stagnation pressure and temperature conditions were provided at the engine inlet, and altitude static pressure was provided at the exhaust nozzle exit. In the choked nozzle testing technique all internal flow conditions throughout the engine are the same as in the direct connect case but the exhaust nozzle exit static pressure may be higher. This condition is limited to operating the exhaust nozzle at sonic velocity, or a nozzle pressure ratio of approximately 2.0. This mode of operation allows a higher facility exhaust system static pressure which may ease the pumping requirements on the central system machinery. If engine thrust is to be measured while testing in this fashion a correction for the deviation from true altitude static conditions must be applied.

The choked nozzle technique is used principally with convergent exhaust nozzles. If a convergent-divergent nozzle is used and a test condition static pressure is required that cannot be met by the facility exhaust machinery, a technique for providing full performance from this nozzle is necessary. This may be accomplished by the use of a "no-flow" or "bootstrap" diffuser. Such a diffuser may have several forms, the simplest being a straight cylindrical pipe attached to the engine tailpipe. The device utilizes the high kinetic energy of the primary stream, by supersonic diffusion, to provide a reduction in primary nozzle back pressure. Such a technique allows an extension of the operating capability of the air handling machinery or conversely extends the operating envelope of the engine under test in the facility.

## Supporting Facility Systems

### Typical Central Air System Complex

In figure 27 is shown a schematic of a complete air and exhaust system required to support the needs of full scale jet engine tests in altitude test chambers. This particular installation is the Propulsion Systems Laboratory at the NASA Lewis Research Center at Cleveland, Ohio. Schematics for such systems at some engine contractors test facilities, here at AEDC, and the Naval Air Turbine Test Station at Trenton, New Jersey would have a basic similarity. It is not intended that any time should be spent in belaboring details of such a system but certain significant features are of interest.

Air and Exhaust System Components

Compressors, exhausters and drive motors. - The combustion air compressors indicated on the flow schematic (fig. 27) are large multistage centrifugal machines. Three casings are coupled to a 16, 500 horsepower synchronous drive motor to provide an output of 120 pounds per second at 40 psi.

If higher pressure is desired the 40 psi air is routed through a fourth casing driven by an 18, 000 horsepower motor to provide 150 psi air.

The so-called exhausters are double sided centrifugal compressors with a pressure ratio of 1. 9. In order to provide the desired altitude static pressure at the test chambers, staging of the many casings can be accomplished for maximum weight flow and operating economy. The first stage consits of 2 casings in parallel driven by a 10, 000 horsepower synchronous drive motor. The second, third, and fourth stages are driven by a 16, 500 horsepower motor. Manual disconnect couplings provide for operation of the second, third, or fourth stage. Additional staging requirements for minimum pressure in the system is accomplished by routing the exhaust gas through two machines normally used as compressors. This provides a sixth and seventh stage.

Exhaust coolers and interstage coolers. - In order to provide the cooling necessary for the exhauster machinery to function properly it is necessary to lower the exhaust gas temperature for the test engine from as high as $3500^{\circ}$ F to $120^{\circ}$ F. This is done in two steps, $3500^{\circ}$ to $600^{\circ}$ F and $600^{\circ}$ to $120^{\circ}$ F in the two coolers located near the altitude chambers. The particular coolers in this system are of shell and tube construction. Other cooling schemes are feasible such as direct water spray cooling or combinations of both. When spray cooling is used the additional mass flow of water has to be handled by the exhauster machinery.

So-called interstage and after coolers are used in conjunction with the various stages of pumping within the central machinery complex for proper operation of subsequent stages and to assure mechanical integrity and performance efficiency.

Air dryers. - Many test requirements are predicated on the use of dry air. Several schemes are provided in this particular system. One such system utilizes chilled water sprays to cool the air to $33^{\circ}$ F at 40 psi. For this condition the air is saturated with 7 to 8 grains of moisture per pound of dry air. Further drying is accomplished by passing this air over refrigeration coils so that its temperature is reduced to $0^{\circ}$ F and its moisture content is 1 to 2 grains per pound of dry air. This system is required for use in an expansion turbine which will provide down to $-100^{\circ}$ F outlet temperature. These systems are continuous use systems. A third system utilizes an alumina bed for providing completely dry air for a period of up to 9 hours after which the bed must be dried for further use.

Air heaters. - In this system three natural gas fired air heaters (heat exchanger type) are used to provide $650^{\circ}$ F air at 150 psi to the test chambers.

This corresponds to the requirements for approximately Mach 3 stagnation conditions at the inlet of the test hardware in the altitude chamber.

Expansion turbine. - A double entry centrifugal compressor operated as an expansion turbine and coupled mechanically into the compressor drive line can provide 100 pounds per second of air at near ambient pressure and $-100^{\circ}$ F outlet temperature. This outlet temperature is controlled by controlling the pressure ratio across the turbine so that high altitude low Mach number stagnation temperatures can be provided to the test chamber. Up to 5000 horsepower is fed back into the compressor drive line during this operation.

## Supersonic Tunnels

Many problems of supersonic propulsion systems and their components involve interaction of internal and external flow. Investigation of these problems and those associated with the external interaction between the powerplant installation and the airframe require external flow simulation such as is provided by supersonic tunnels.

Modestly sized tunnels are used successfully for studies of some of these problems but many of the problems require large tunnels capable of accommodating full-scale engines and large-scale engine-airframe configurations. The 8×6 foot and 10×10 foot tunnels at NASA Lewis Research Center, and the 16×16 foot transonic and supersonic tunnels at AEDC, are examples of such facilities useful for investigating such problems. These tunnels are all designed to handle combustion products.

Detailed studies of inlet and induction system and exhaust nozzle components comprise a large part of the investigations conducted in supersonic tunnels.

Although inlet problems can be studied in combination with complete propulsion systems, even the large tunnels are often unable to test complete engine installations without strong shock reflections from the tunnel walls on the rear of the model. Consequently, nozzle research is conducted on models that simulate only the afterbody of the installation.

A typical jet nozzle and ejector model is shown in figure 28. The model shell which represents the aircraft external surface is mounted on a force balance so that all body forces can be measured directly. High pressure air is brought through the strut and simulates the engine air flow. Besides the usual nozzle problems which can be investigated, a variety of interaction problems can also be studied; interactions between aircraft components (such as the tail surfaces) and the jet or between free stream and the jet. As illustrated in the bottom rear half of the configuration, ejector nozzle performance may also be studied along with possible auxiliary inlet configurations designed to supply secondary air to the ejector as efficiently as possible.

A typical installation of a turbojet engine in a nacelle in a supersonic tunnel is illustrated in the photograph (fig. 29). A schematic arrangement of a nacelle test configuration is shown in figure 30. In order to reduce the number of engine test hours at supersonic operating conditions and in order to obtain basic data relating to the effect of the engine on the inlet characteristics, much of the program can be run with the engine removed, as shown in the bottom sketch. In this case, a duct liner was installed within the nacelle, and a remotely actuated plug provided to simulate the engine back pressure.

## Propulsion Cycle

The Lewis Research Center 8×6 foot wind tunnel is a continuous operation return or nonreturn transonic tunnel with a controlled Mach number range from 2. 1 to a lower limit of 0. 5 or as determined by model blockage and shock reflection. The tunnel can be operated on either a propulsion cycle (open circuit) or an aerodynamic cycle (closed circuit).

The propulsion cycle is illustrated in figure 31(a). Here the air enters the alumina air dryer where up to a ton of water per minute can be absorbed on a summer day with $67^{o}$ saturated air. A dewpoint of $-40^{o}$ F is maintained. The air dryer is reactivated after each run, which required about 9 hours. The air then enters the plenum chamber and the seven-stage axial flow compressor. The compressor is driven by three electric motors totaling 87,000 horsepower. The speed is controlled from 745 to 880 rpm. The maximum airflow is 2,000, 000 cubic feet per minute, and the maximum compression ratio is 1. 8. From the compressor the air passes through the flexible-wall nozzle, which can be varied while the tunnel is operating to control the test-section Mach number. The flexible nozzle is made of 1-inch-thick stainless steel plates 35 feet long, 8 feet high, and 6 feet wide. The first half of the test section is the supersonic portion, and the last half, which is perforated on all four sides, is the transonic portion. After leaving the test section, the air flows through the diffuser and the accoustical muffler and is exhausted to atmosphere. This cycle is used for all burning-type model installations.

## Aerodynamic Cycle

The aerodynamic or return cycle is used for all nonburning model installations. As shown in figure 31(b), the operation is similar to that of the propulsion cycle except that the air is not exhausted to the atmosphere but cooled by a cooler located in the duct and returned to the dryer building. The rest of the cycle remains the same as for the propulsion cycle.

The Lewis Unitary Plan Wind Tunnel better known perhaps as the 10×10 foot Supersonic Tunnel has a Mach range of 2 to 3. 5 and can be operated throughout this range either on an aerodynamic cycle at various air densities or on a propulsion cycle. Although it is basically similar to the 8×6 foot tunnel just described it varies in operational flexibility as well as certain features that are apparent from the sketch in figure 32.

## Component Test Facilities

As mentioned earlier in the text the discussion has centered chiefly around full engine or propulsion system test facilities, but the subject would not be complete without mention of component test facilities. In this context components such as the compressor, combustor, etc. are meant to be full scale hardware or nearly so. The figures that follow are meant to provide an insight into the associated complexity necessary for component testing that certainly is not unique solely to fully assembled engine testing. In many ways it is oftentimes more difficult to set up a suitable component facility than it is to provide for a complete engine test.

### Compressor Facility

Figure 33 is a schematic of a compressor test facility. Air and exhaust piping provide the necessary environmental conditions desired for test simulation. The compressor is coupled to a 15, 000 horsepower variable speed drive motor so that rotative speeds can also be properly simulated.

### Combustor Facility

Figure 34 is a schematic of rather unique full scale combustor test facility.

The main feature of this facility is the utilization of two jet engines operating on natural gas to provide the high temperature air for the heat exchangers. Combustion air is routed through the heat exchangers ahead of the research combustor and heated to a maximum of $1200^{\circ}$ F simulating compressor discharge temperatures of an SST engine at nominally Mach 3 flight conditions. Figure 35 is a photograph of jet engines and heat exchangers and figure 36 is a photograph of the test section used for combustor hardware.

### Turbine Facility

The photograph of figure 37 shows the significant features of a turbine component test facility. The turbine is driven by high pressure air from the central air system and discharged into the altitude exhaust system. The inlet air may be heated up to $300^{\circ}$ F in order to prevent icing problems as it expands through the turbine. Power is absorbed by the 5000 horsepower eddy current dyamometers.

### Nozzle Facility

A typical nozzle test facility is shown in figure 38. This facility will handle hardware of approximately one-quarter scale. The test nozzle is installed on a mounting pipe located inside a test chamber. The test chamber is connected to central air and exhaust services to provide the desired pressure ratio across the nozzle. In addition to flow requirements, the inlet air must be dried or heated sufficiently to prevent condensation shocks from forming within the nozzle. The jet thrust of the test nozzle is determined from a complete force balance on the nozzle and mounting pipe.

## CONCLUSION

As with any discussion on ground test facilities whether for airbreathing engines, rocket propulsion, or space simulation, there are no doubt many unanswered questions and many areas left undescribed. Certainly this paper is no different in this regard.

In planning this presentation for its intended use it was felt the format chosen would be more meaningful in that it places the facility in the proper relationship to the propulsion system and its likely problem areas. Time has limited the degree of design and operational details presented on the facilities but it is hoped those presented are informative and useful.

Finally, the author has drawn primarily on the experience and facilities of the NASA and its predecessor the NACA, but recognizes that a more comprehensive treatment would include the fine work and description of facilities of other government agencies and industry both in the United States and foreign countries.

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E-3791

## BASIC TURBOJET ENGINE

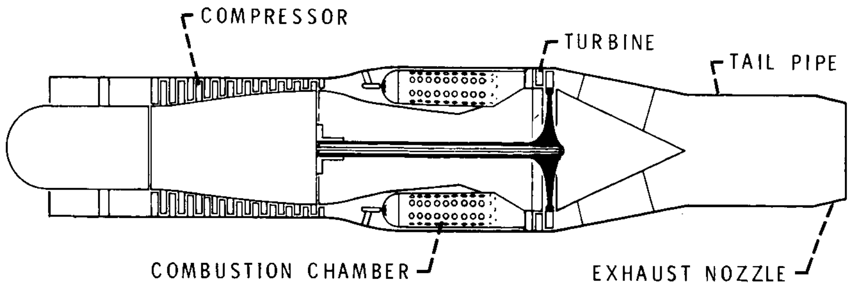



Fig. 1

CS-42067

## TURBOJET ENGINE WITH AFTERBURNER

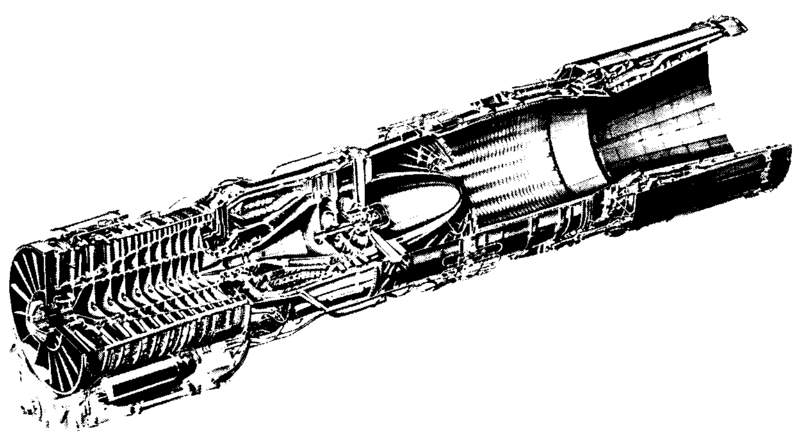
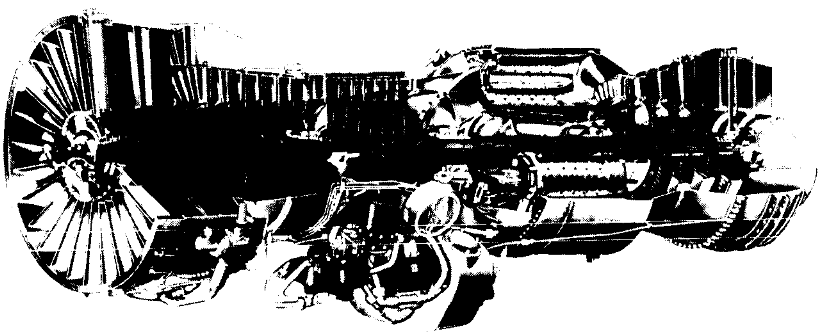


Fig. 2

CS-42094

TURBOFAN ENGINE



CS-42096

Fig. 3

TURBOPROP ENGINE

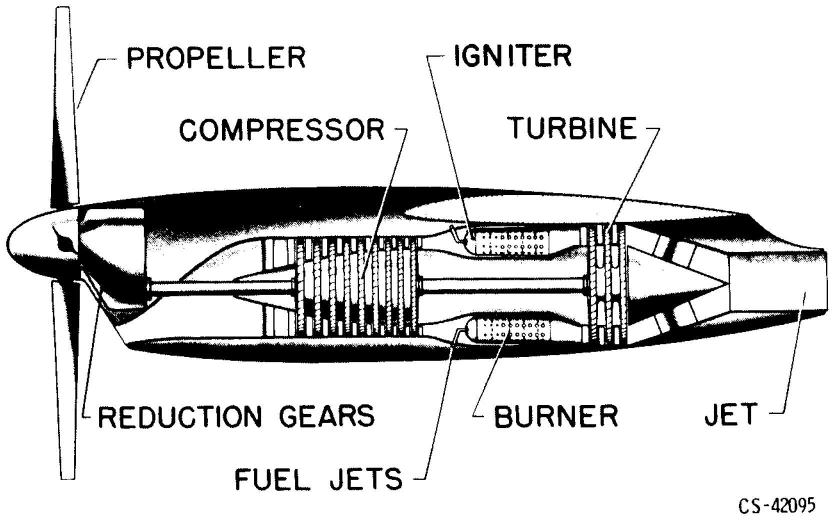



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Fig. 4

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## RAM JET ENGINE

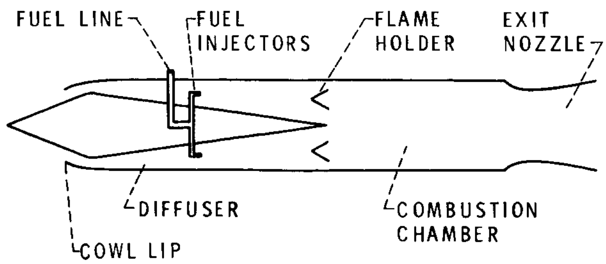


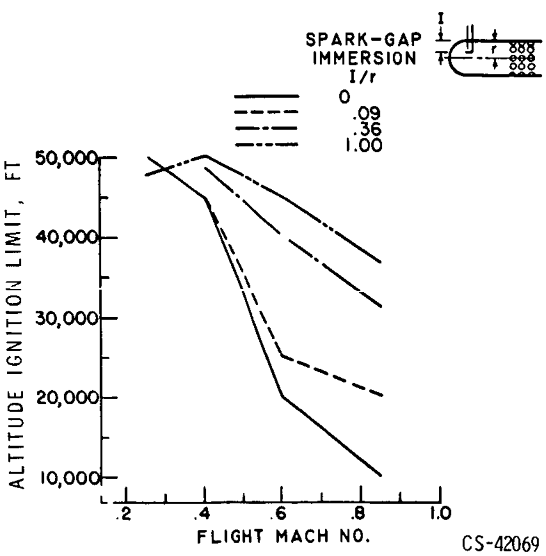

Fig. 5 CS-42091

## PROPULSION SYSTEM PROBLEM AREAS

1. SYSTEM PERFORMANCE EVALUATION AND CALIBRATION
2. ALTITUDE OPERATING CHARACTERISTICS
3. AFTERBURNER DEVELOPMENT
4. COMPRESSOR INLET FLOW DISTORTION
5. ENGINE DYNAMICS AND CONTROLS
6. INLET PRESSURE AND TEMP TRANSIENTS

Fig. 6 CS-42068

## EFFECT OF SPARK-GAP IMMERSION ON ALTITUDE IGNITION LIMITS




CS-42069

Fig. 7

## TYPICAL ALTITUDE STARTING LIMITS

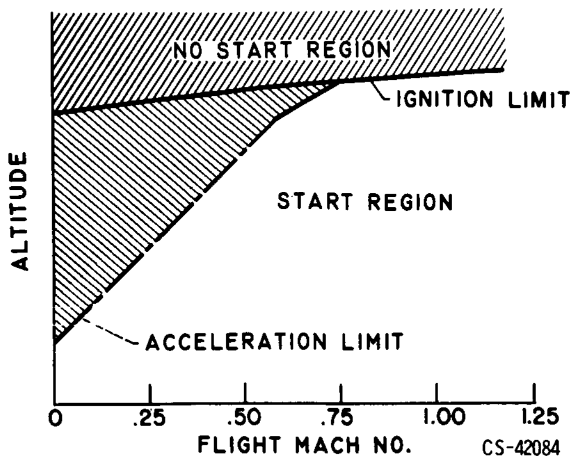



Fig. 8

## AFTERBURNER RESEARCH PROBLEMS

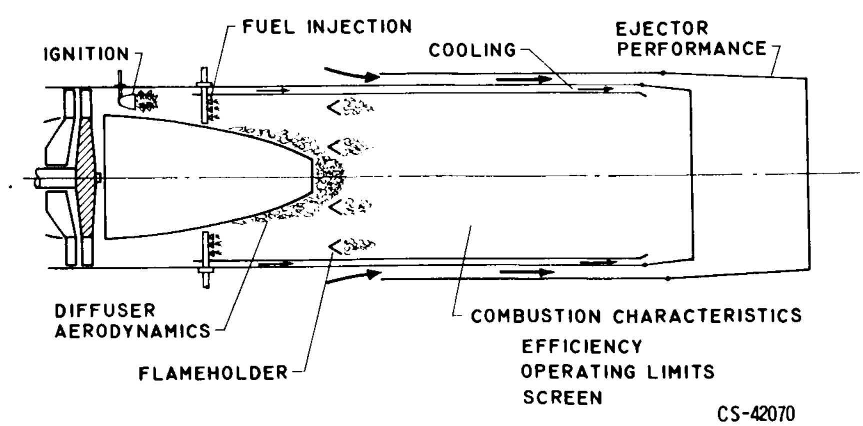
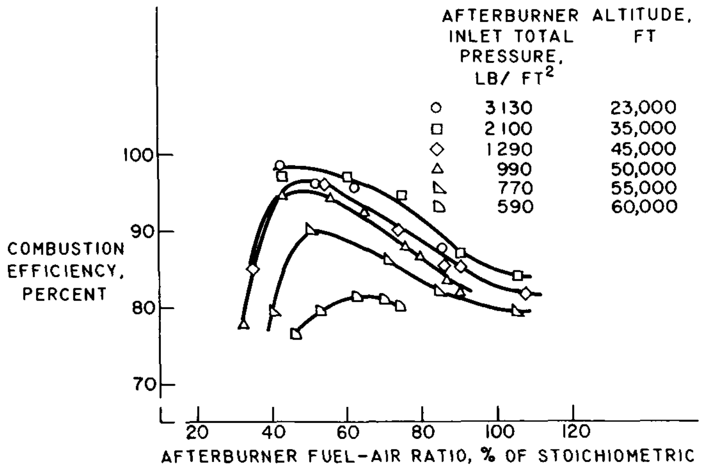



Fig. 9

E-3791

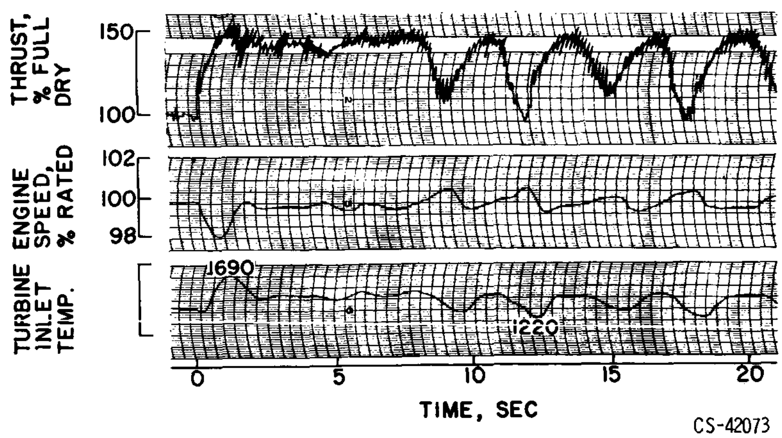
# EFFECT OF ALTITUDE ON AFTERBURNER PERFORMANCE




CS-42071

Fig. 10

# BEHAVIOR OF AFTERBURNER CONTROL




CS-42073

Fig. 11

## GENERALIZED PERFORMANCE VARIABLES

$$\delta = \frac{\text{OBSERVED ABSOLUTE PRESSURE}}{29.92\text{" Hg (OR 2116 PSFA)}}$$

$$\theta = \frac{\text{OBSERVED TEMPERATURE (IN }^{0}\text{R)}}{519^{0}\text{ R}}$$

AIRFLOW = $W_a\sqrt{\theta}/\delta$

VELOCITY = $V/\sqrt{\theta}$

RPM = $N/\sqrt{\theta}$

THRUST = $F/\delta$

FUEL FLOW = $W_F/\delta\sqrt{\theta}$

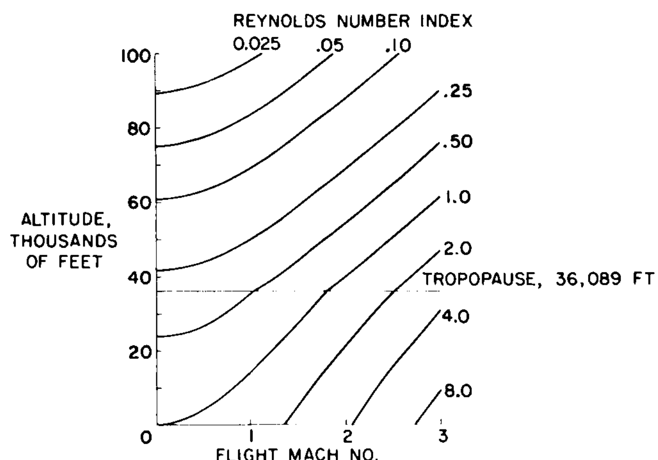
PRESSURE = $P/\delta$

TEMPERATURE = $T/\theta$

CS-42072

Fig. 12

## VARIATION OF REYNOLDS NUMBER INDICES

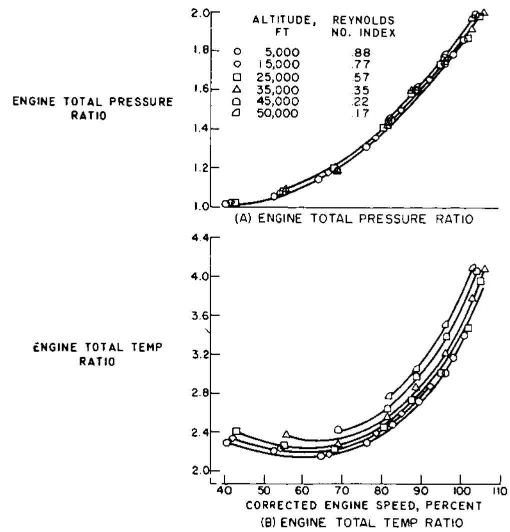



CS-42074

Fig. 13

E-3791

## TURBOJET PUMPING CHARACTERISTICS FOR A RANGE OF REYNOLDS NUMBER INDICES




CS-42075

Fig. 14

## EFFECT OF ALTITUDE ON GENERALIZED ENGINE PERFORMANCE

FLIGHT MACH NO. 0.2

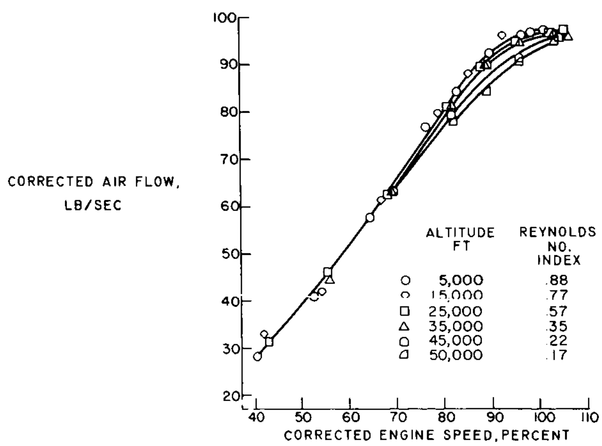



Fig. 15

CS-42076

## CORRELATION OF COMBUSTION EFFICIENCY WITH COMBUSTION PARAMETER

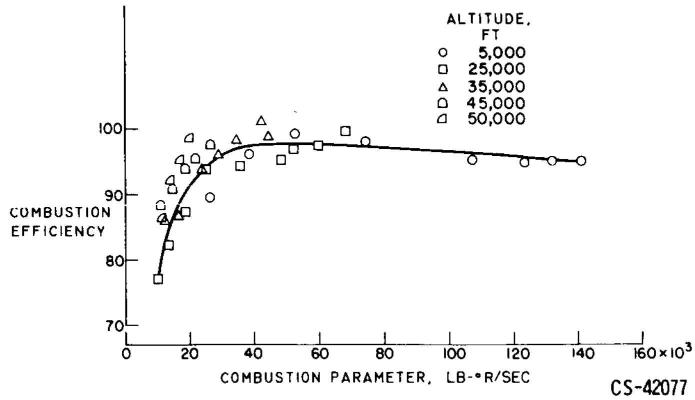



Fig. 16

## OVER-ALL PERFORMANCE CURVES GENERATED FROM TYPICAL ENGINE PUMPING CHARACTERISTICS

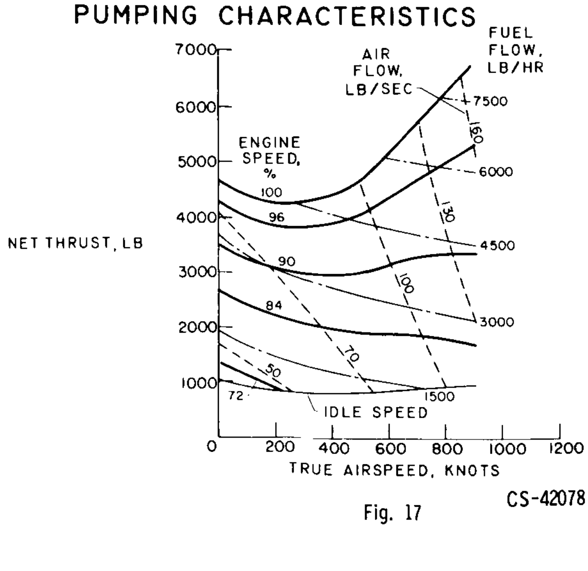
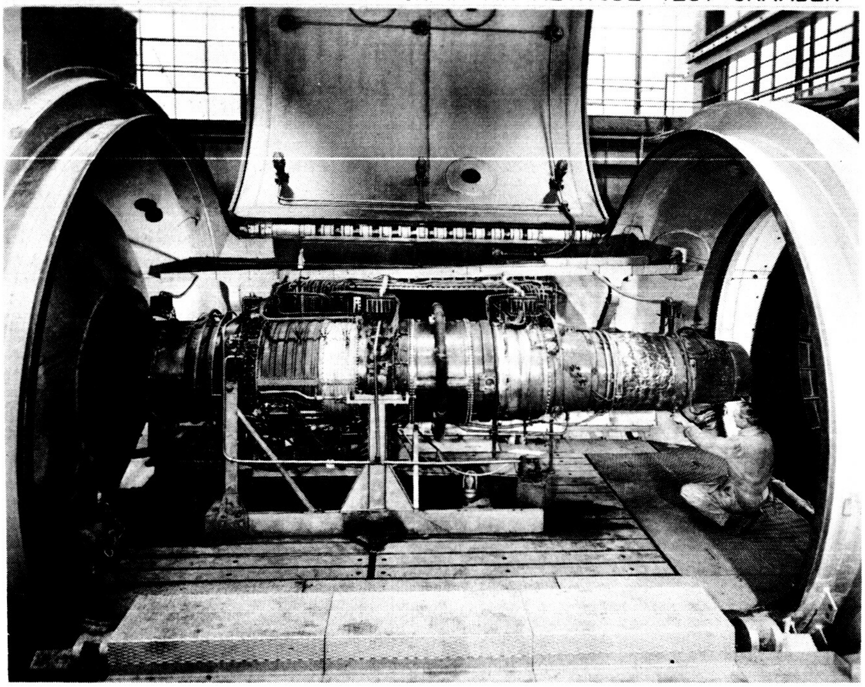



Fig. 17

E-3791

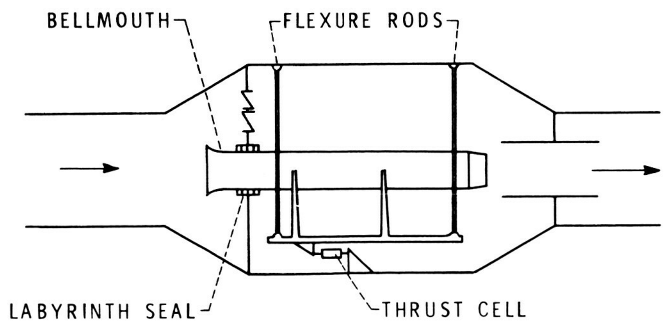
TYPICAL TURBOJET INSTALLATION IN AN ALTITUDE TEST CHAMBER



CS-42103

Fig. 18

SCHEMATIC OF TURBOJET ENGINE IN AN ALTITUDE TEST CHAMBER




CS-42093

Fig. 19

SCHEMATIC OF AUTOMATIC ENGINE INLET PRESSURE CONTROL SYSTEM IN ALTITUDE TEST CHAMBER

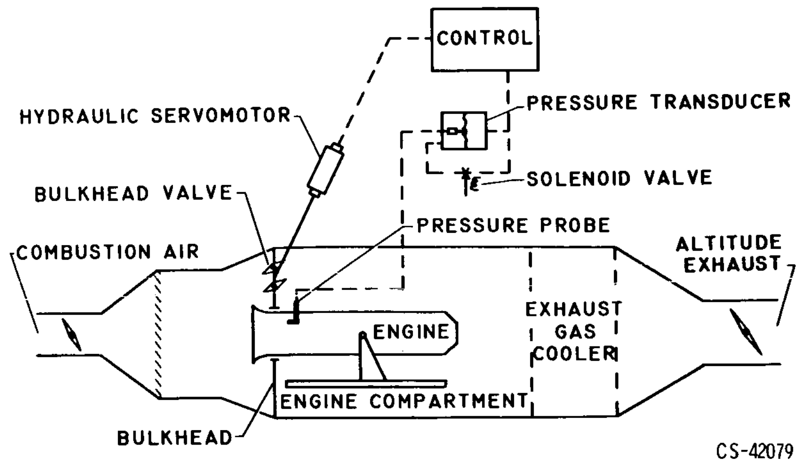



Fig. 20

TYPICAL SCREEN ARRANGEMENTS AND INLET FLOW DISTORTIONS

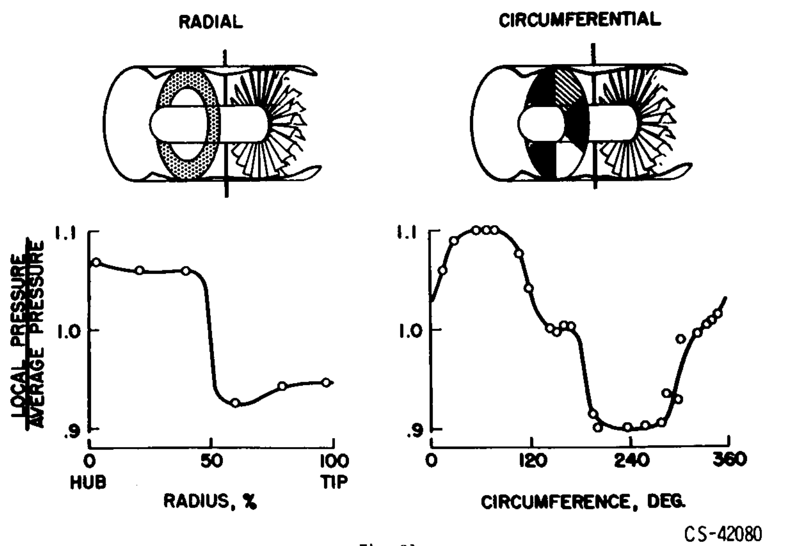



Fig. 21

E-3791

INLET PRESSURE PULSE GENERATING DEVICE

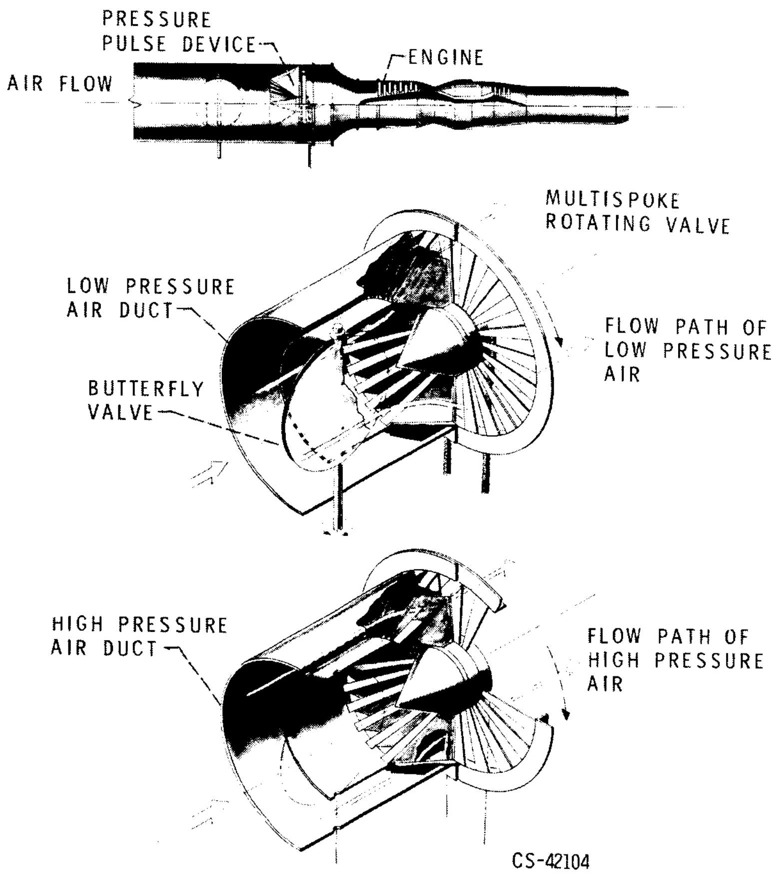



Fig. 22

## INLET TRANSIENT TEMPERATURE DEVICE

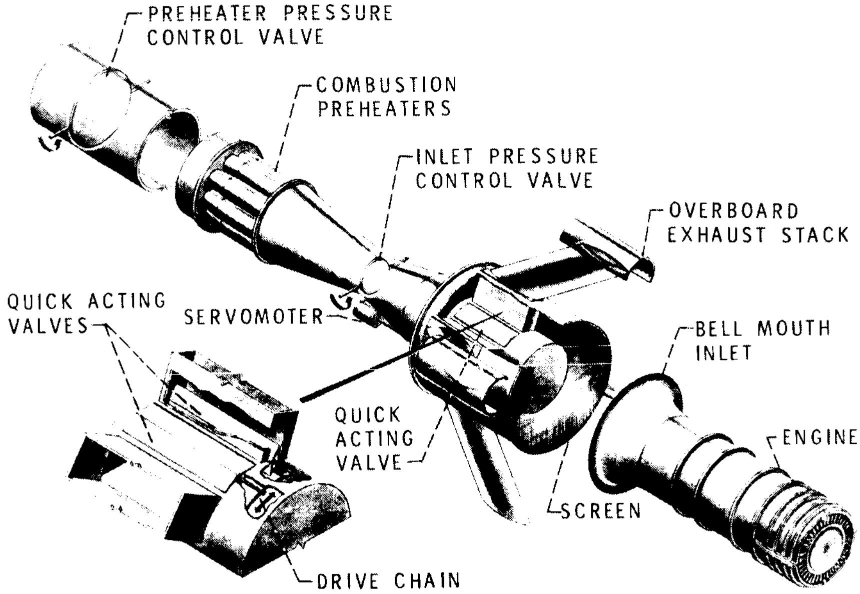
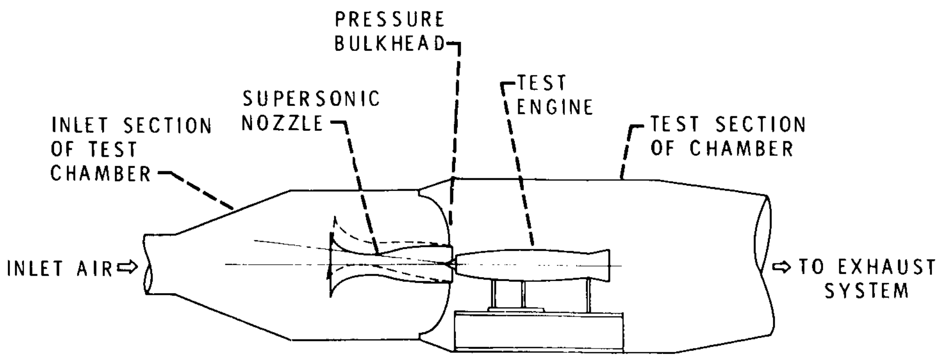



Fig. 23 CS-42100

## SCHEMATIC OF FREE-JET INSTALLATION OF RAM JET ENGINE IN ALTITUDE TEST CHAMBER

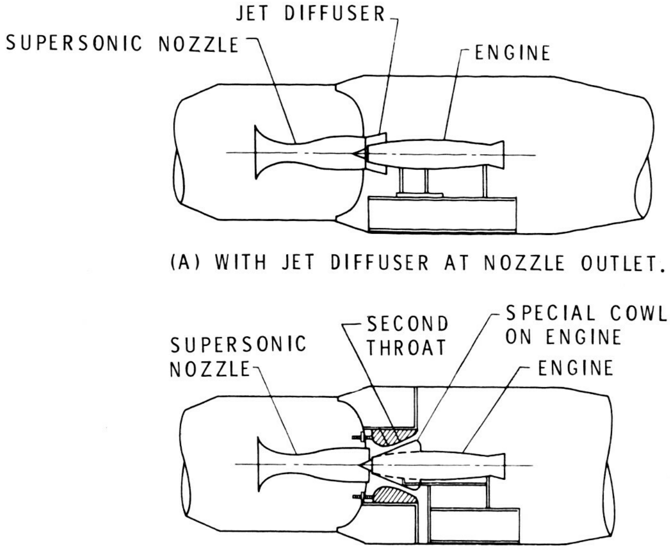



CS-42081

Fig. 24

E-3791

SCHEMATIC OF FREE JET INSTALLATION WITH JET DIFFUSER AND SECOND THROAT FOR PRESSURE RECOVERY




(A) WITH JET DIFFUSER AT NOZZLE OUTLET.

(B) WITH SECOND THROAT AROUND ENGINE COWL.

CS-42082

Fig. 25

28" RAMJET INSTALLATION IN ALTITUDE TEST CHAMBER

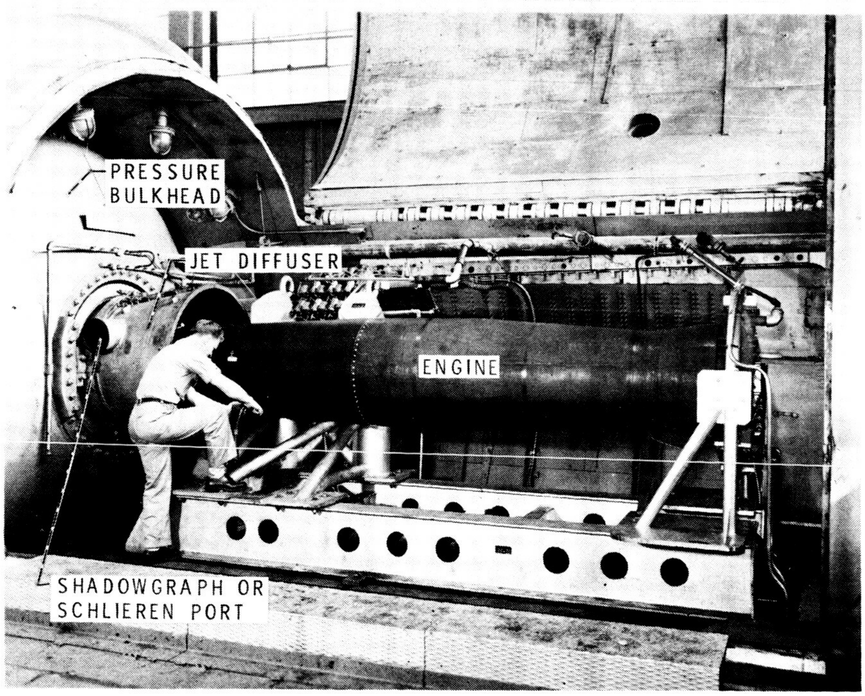
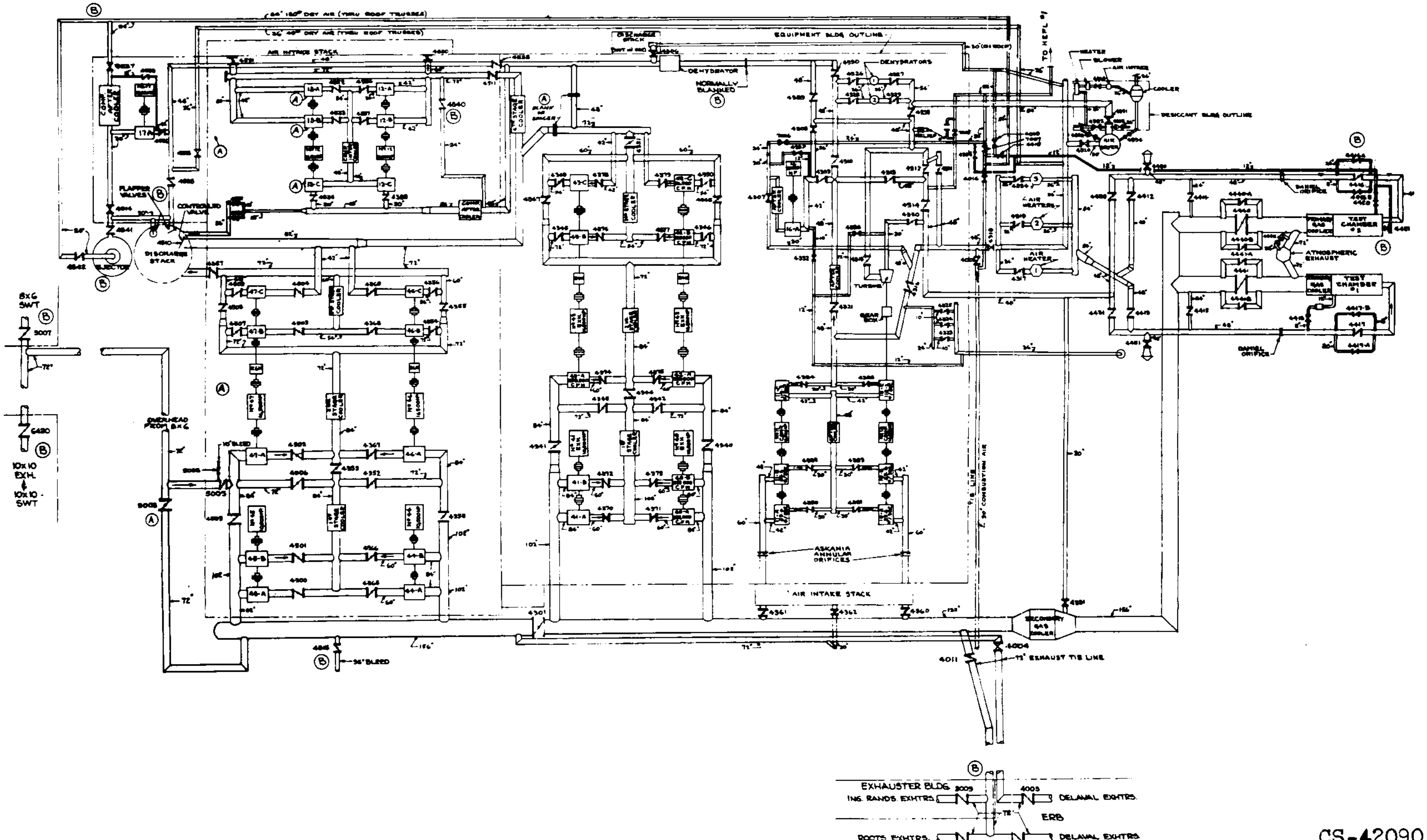



Fig. 26

CS-42098

SCHEMATIC OF COMBUSTION AIR AND EXHAUST SYSTEMS

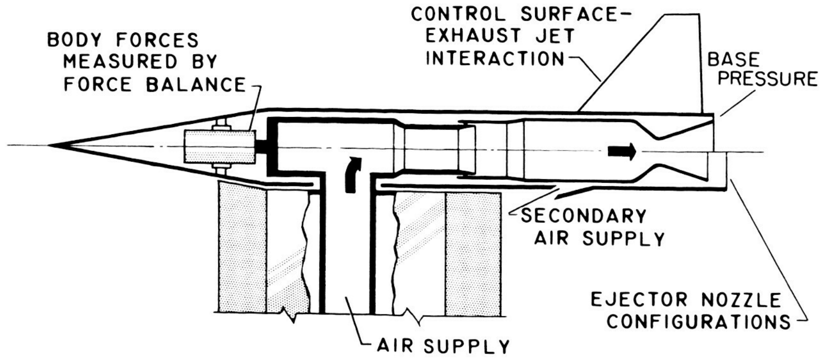



CS-42090

Fig. 27

E-3791

SCHEMATIC OF JET EXIT RESEARCH MODEL




CS-42089

Fig. 28

NACELLE INSTALLATION IN SUPERSONIC TUNNEL

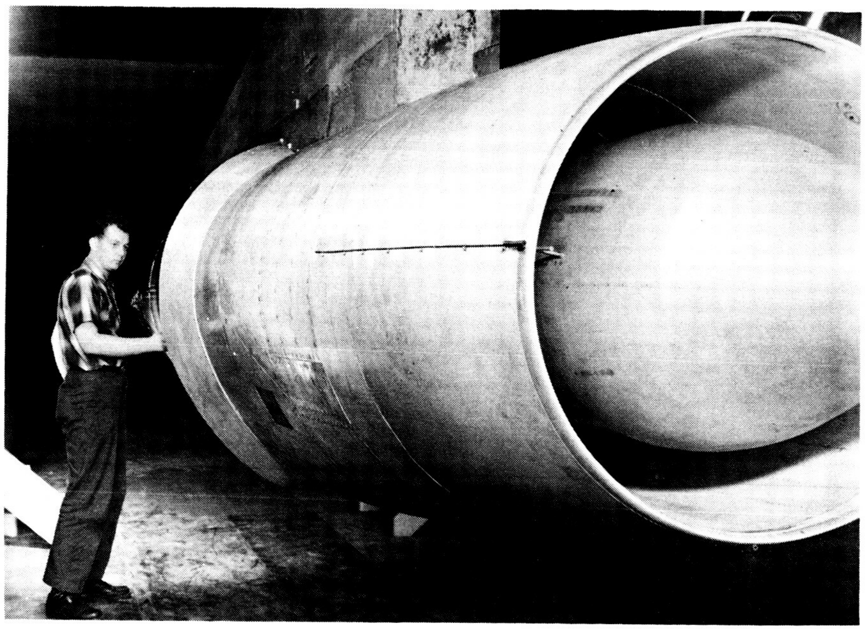


Fig. 29

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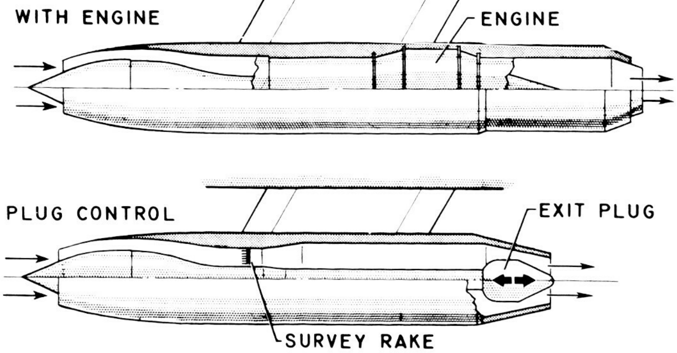
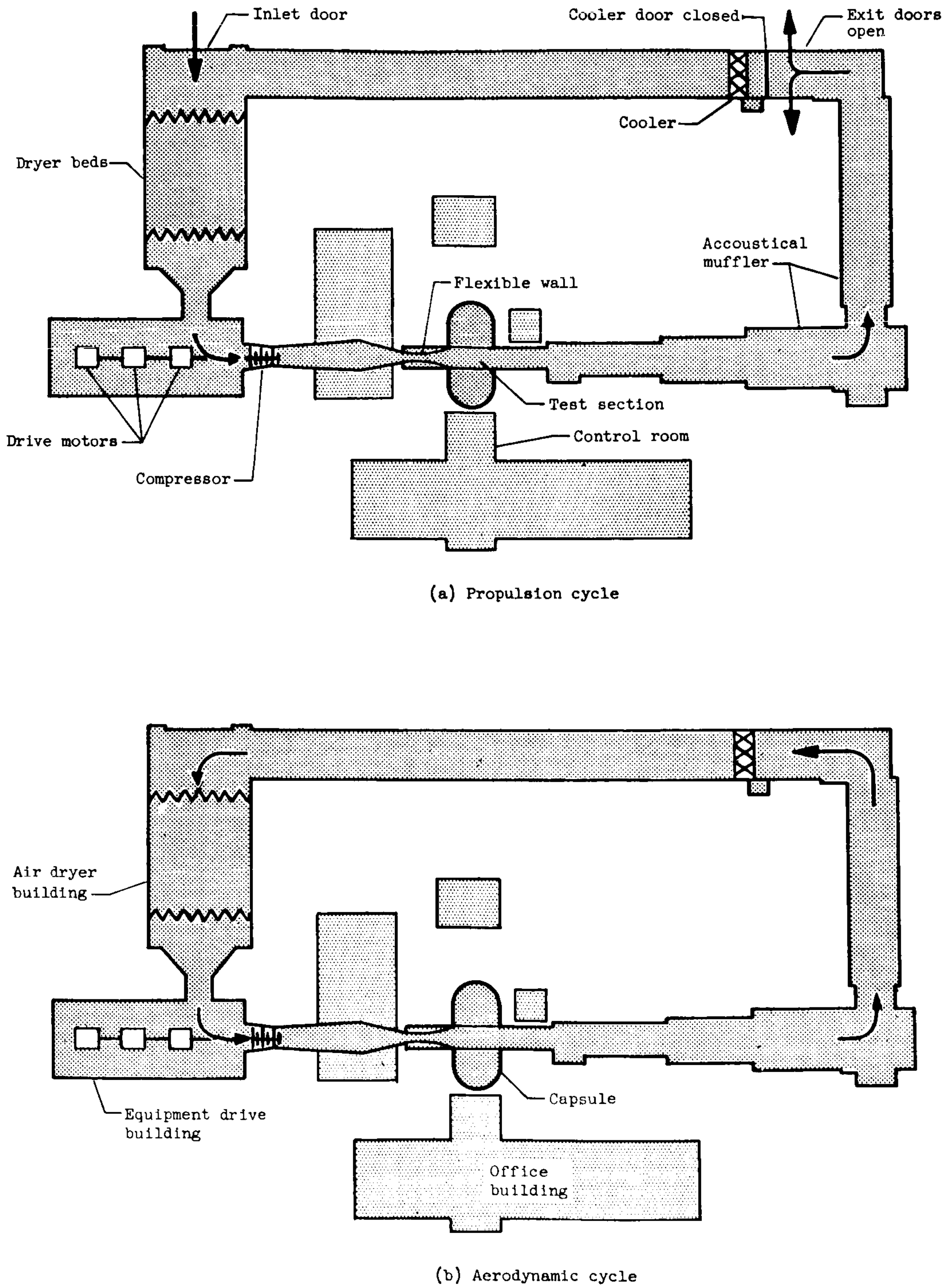
SCHEMATIC OF TURBOJET INSTALLATION




CS-42088

Fig. 30

# 8X6 FT. SUPERSONIC TUNNEL OPERATING CYCLES

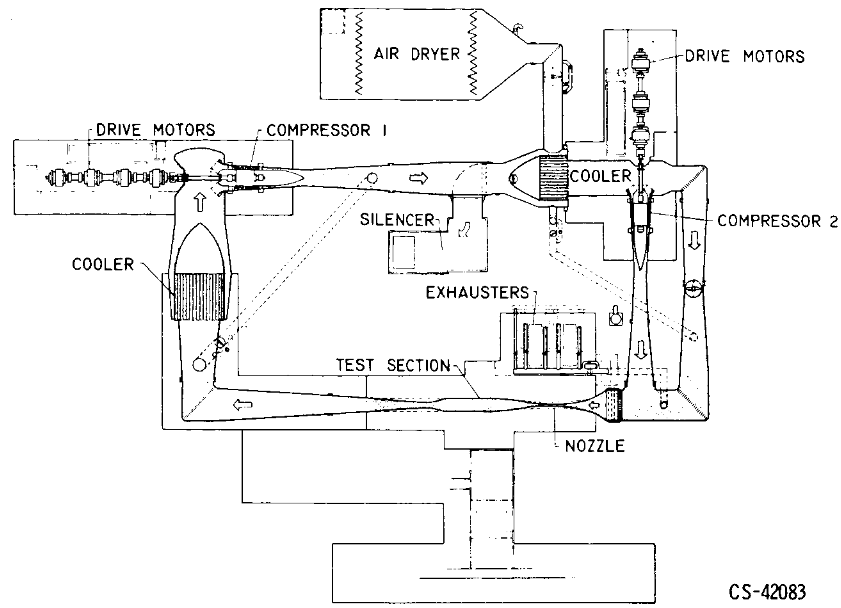



CS-42086

Fig. 31

E-3791

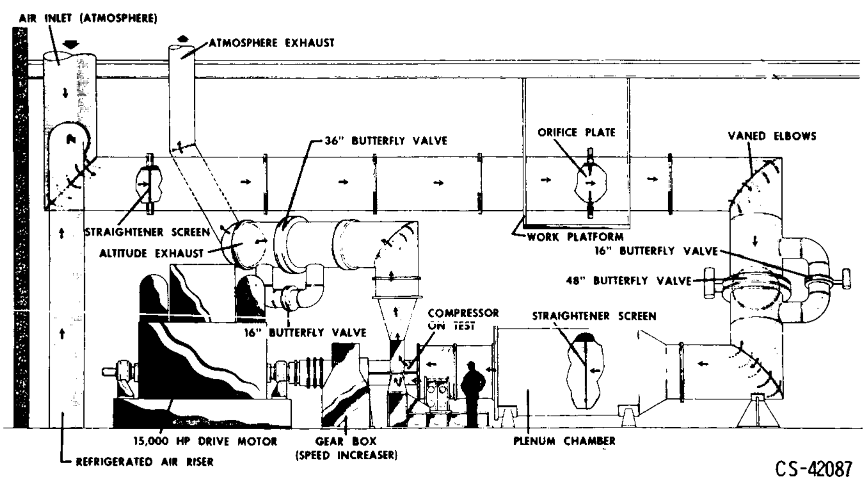
PLAN VIEW OF UNITARY PLAN WIND TUNNEL




CS-42083

Fig. 32

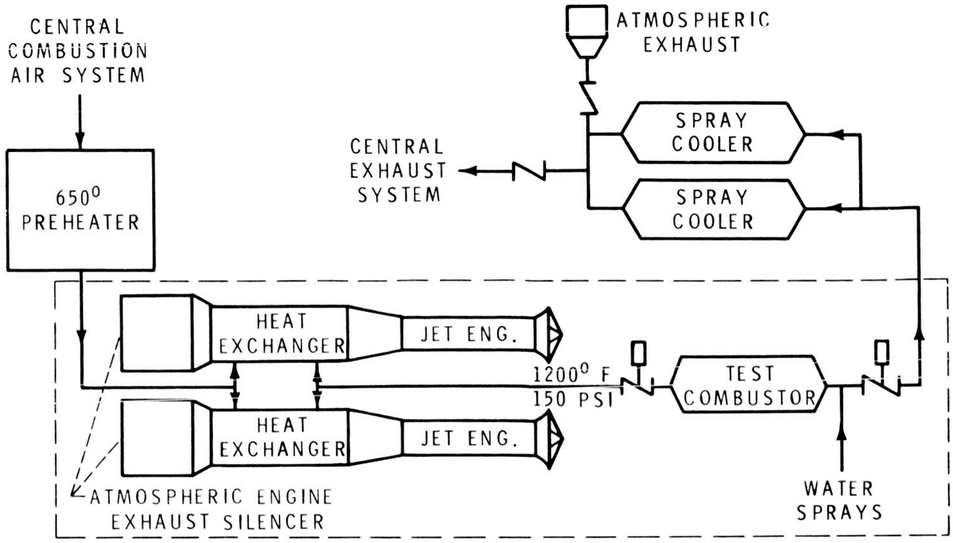
ADVANCED COMPRESSOR INSTALLATION




CS-42087

Fig. 33

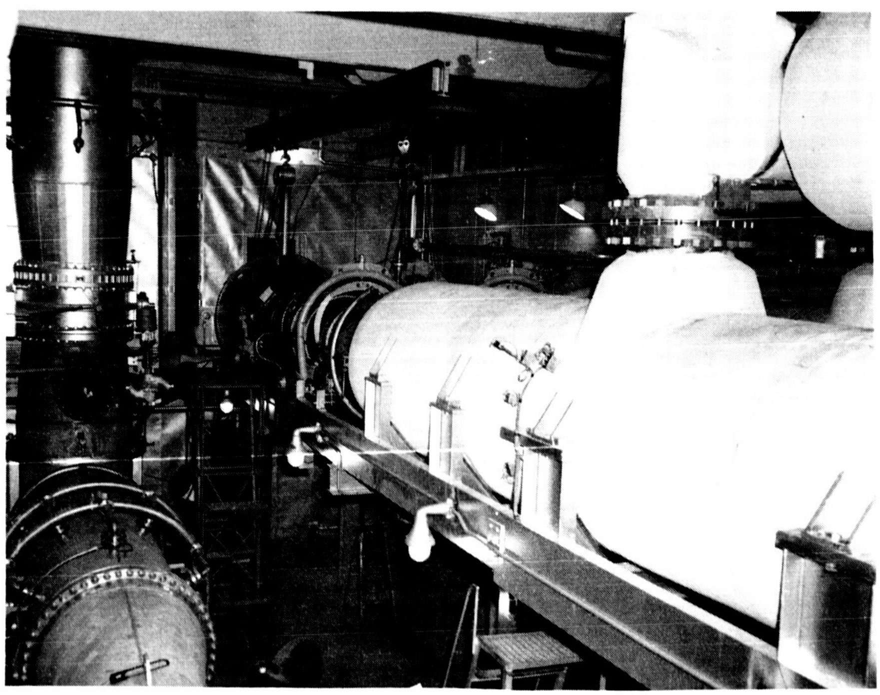
SCHEMATIC OF HIGH TEMPERATURE COMBUSTOR FACILITY




CS-42092

Fig. 34

JET ENGINE AND HEAT EXCHANGER FOR COMBUSTION FACILITY



CS-42102

Fig. 35

E-3791

TEST SECTION OF COMBUSTION FACILITY

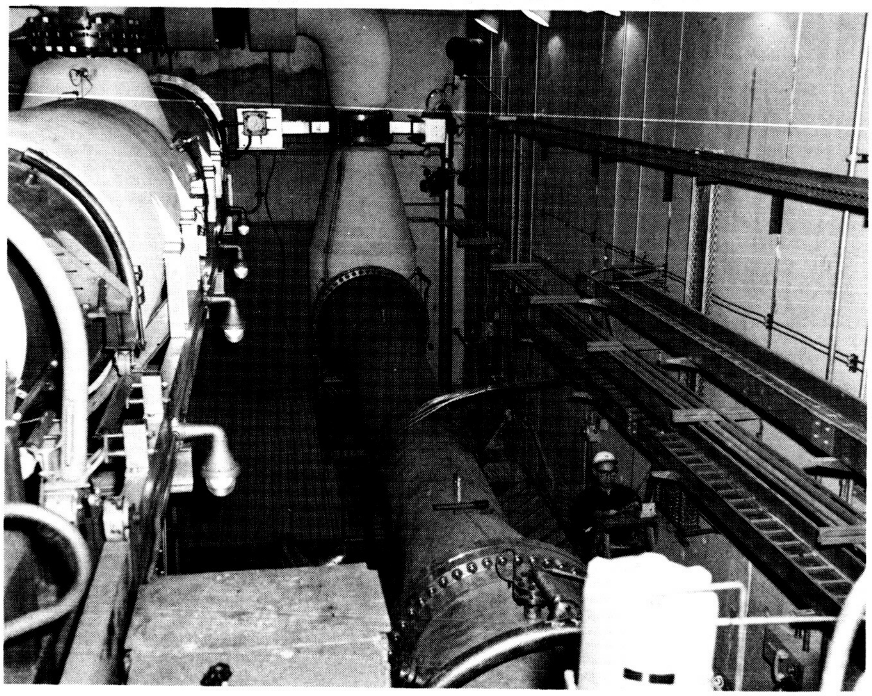


Fig. 36 CS-42101

TURBINE TEST FACILITY

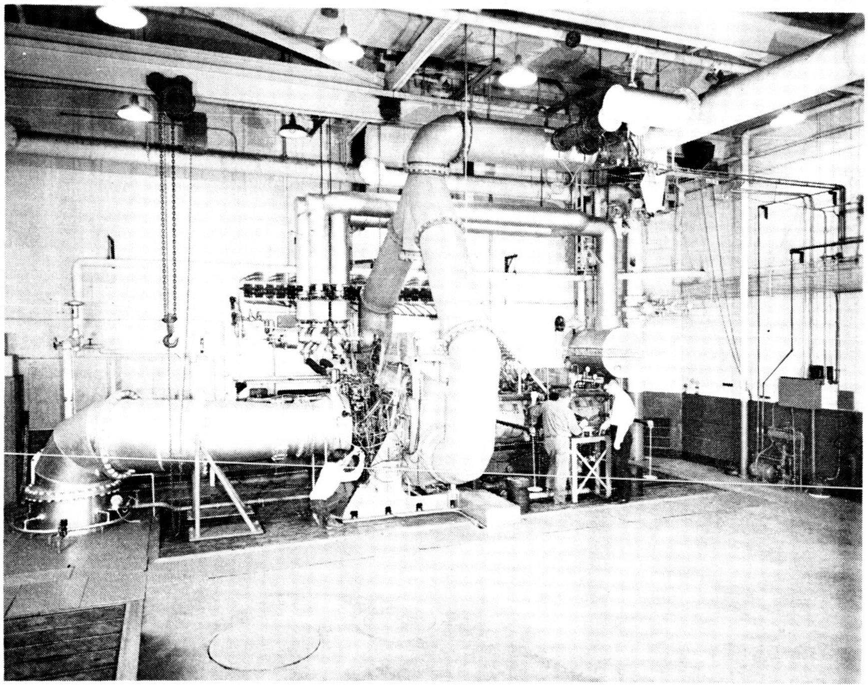


Fig. 37 CS-42099

SCHEMATIC OF NOZZLE TEST FACILITY

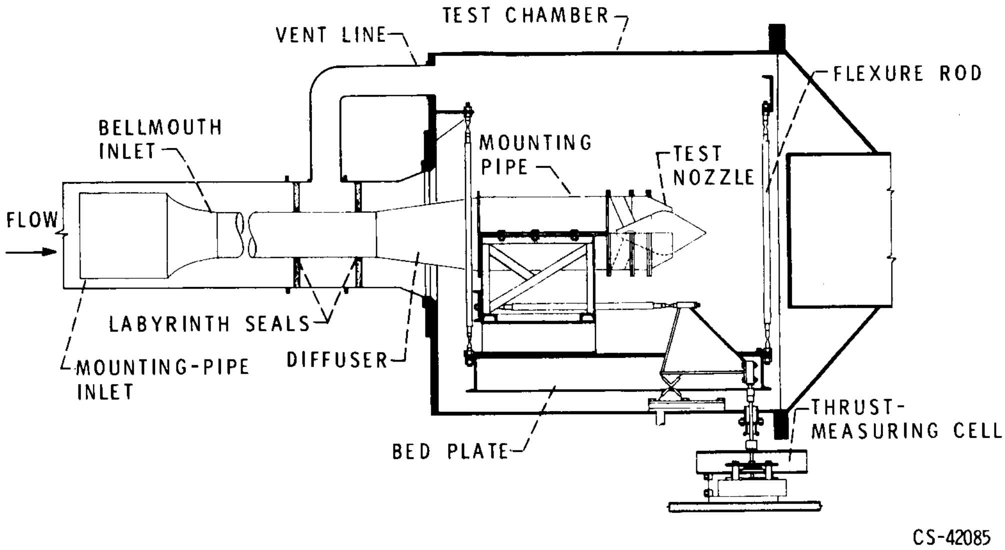



Fig. 38