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A COMPARISON OF ADVANCED COOLING TECHNIQUES FOR ROCKET THRUST CHAMBERS

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Abstract

In the early days of rocket propulsion, two primary methods were employed for cooling the walls of thrust chambers. These were uncooled metal chambers where the heat sink capacity of the chamber and nozzle wall materials limited the operating duration, and regeneratively cooled chambers where one of the propellants was circulated in a cooling jacket which constituted the chamber wall. Today, there are at least fourteen different methods with variations for cooling the combustion devices and nozzles of liquid propellant, solid propellant, and/or nuclear rocket propulsion engines. It is the intent of this paper to examine these methods, to describe for

A-7-SC 65-130

each the useful range of operating conditions, as well as present and likely future applications, to define their limitations and associated problems. Emphasis is primarily placed on liquid rocket engines.

Introduction

In a rocket thrust chamber a vigorous convective transfer of heat occurs from the hot combustion gases through a partially-insulating, low-velocity gaseous boundary layer to the walls of the chamber (Ref. 1). The relatively high resultant heat flux generally requires that wall be cooled to prevent failure. There are two basic processes which aid in the understanding and differentiation of the various cooling methods available: (1) the controlling mechanism for the absorption of the heat that is rejected from the hot gases to the walls and (2) a mechanism for reducing or limiting the amount of heat to be absorbed, by the introduction of a thermal resistance. Existing and practical solutions for the further definition of these two basic controlling parameters for each cooling method will be developed. Fourteen cooling types are described in this paper. They are all basic methods and not necessarily the only cooling methods for keeping rocket engine hardware from overheating, since there is overlap in several of these methods. For each of the fourteen cooling methods, the present discussion describes (1) the basic cooling mechanism, (2) the principal limitations, (3) the likely heat transfer rates, temperatures, etc., (4) means for extending the limits (e.g., increased heat flux), and (5) principal applications. The cooling methods will be described approximately in the order of increasing relative heat-flux rate; that is, the thermal energy transferred to the inner wall per unit of surface area and time. Table 1 shows an orientating comparison of the 14 proposed cooling methods, together with several typical applications and a simplified diagram of the temperature gradient from the hot-gas temperature (T_g) (about 4000 to 7000°F, depending on the particular propellant combination and other factors), to the temperature of the wall on the gas side (T_{wg}) and the cooler side (T_{wc}) to the temperature of the coolant (T_c). Table 2 illustrates applicable levels of heat flux for most of the methods.

1. Radiation Cooling

In radiation-cooled chambers, there is a steady-state heat-transfer condition with radiation to the surrounding environment from the hot wall. The wall may radiate at a maximum temperature of 2500 to 2900°F if refractory metals (Cb, W, Ta, and Mo) are used. The heat is transferred by convection from the hot gases to the chamber wall, conducted through the wall, and in turn it is rejected to its cooler surroundings by means of thermal radiation.

Refractory metal experiences recrystallization and brittle transition which is undesirable for a hard start when the wall is cold. Coatings are also necessary for preventing oxidation of the refractory metal walls by the combustion gases and ambient air; present disilicide coating materials fail at about 3000°F. Any scratches in the coating usually cause failure, and the coating life becomes a function of operating temperature level. Limits of cooling with cluster conditions (space factor in radiation equation) are not well established. Radiation heating to the vehicle's base requires either heavy radiation shields, thermal insulation, or special and somewhat complex vehicle internal cooling systems. Fabrication difficulties currently are many in making refractory metals (welding, forming, joining, etc.). Emissivities of current metals are poorly defined in terms of surface condition and coatings. Some refractory metals experience hydrogen embrittlement, and, since hydrogen is present in almost all rocket propellant exhaust gases, this can be a severe limitation. The chamber must be mounted external to the vehicle to permit adequate radiative heat rejection. Radiative cooling is not suitable for applications where the chamber is buried inside the vehicle, e.g., in a re-entry body.

The advantages of radiation cooling are that it provides a simple and light construction, it is throttleable and can operate for long firing durations. The principal applications for the radiation method are for small attitude-control liquid-propellant engines of space payloads or terminal space vehicle stages (usually less than 150-lb thrust and 50 psia chamber pressure) (Ref. 2), where the rocket is mounted externally to the spacecraft, and for uncooled nozzle skirt extensions of solid or liquid propellant chambers of all sizes where the chambers themselves are cooled by other means. The radiation cooled nozzle skirt becomes useful at locations where the heat flux and local stream pressure are low and where the nozzle protrudes beyond the vehicle external envelope to permit adequate heat rejection. Fig. 1 shows typical minimum attachment point area ratio with a molybdenum radiation skirt for several propellant combinations.

The discovery of a superior protective coating (against the corrosive hot gases and with a high emissivity) capable of going to higher wall temperatures would slightly increase the maximum heat flux (and chamber pressure) and increase the usefulness of this method of cooling technique. Film cooling and other supplementary cooling techniques (to be described in a later section) will also increase this capability.

2. Heat-Sink Cooling

The use of a single-wall metal combustion chamber and nozzle offers a simple, easy-to-make and inexpensive means of cooling (Ref. 3). Because of the transient nature of this cooling method, it is limited by the heat capacity of the wall as determined by the wall thickness, density, and specific heat. At high heat flux conditions, relatively short durations of only a few seconds may be possible. Failure occurs when the inside wall temperature exceeds the point at which either melting or erosion takes place. A high thermal conductivity of the wall material is important to reduce the temperature gradient and thus the gas side wall temperature. Often a material with a maximum value of the factor consisting of the product of density and specific heat, divided by the conductivity, is considered to be optimum. Materials such as copper and mild steel are typical choices. A protective or insulating coating or film cooling will increase the allowable chamber pressure and/or duration of this type of cooling.

In some repeated-use applications (many starts and stops), high thermal stresses on the inner wall cause the formation of gradually increasing axial cracks on the inside wall surfaces because the local tangential compression stresses exceed the yield point of the material and leave a residual tension stress after the wall cools down to the original ambient temperature. When a relatively cool propellant exhaust gas is used (e.g., very fuel-rich bipropellant mixtures and low flame temperature), the duration of this type of thrust chamber is substantially increased. The addition of extra metal to increase the thickness and, thus, the heat capacity of the wall is limited by weight considerations in flying vehicles but is not a serious drawback for experimental thrust chambers.

A principal area of application is for selected short-duration use in flying vehicles where this type of cooling can allow low-weight and low-cost thrust chambers. The maximum operational duration is a function of the propellant combustion gas characteristics, the chamber pressure, size (thrust level) and chamber geometry. For experimental investigations of new propellants, new injectors, or new chamber and nozzle geometries, an uncooled, non-flyable, heavy-duty metal thrust chamber will give useful and relatively inexpensive service.

For solid propellant motors, materials which exhibit good heat sink properties, together with high melting points, are high-density graphite, edge-grain pyrolytic graphite, and tungsten. For less severe conditions, tantalum, 90 tantalum/10 tungsten and molybdenum have been used.

3. Low Flame Temperature Metal Chamber

When the maximum allowable wall temperature is above the flame temperature, then there is really no cooling problem. This occurs, for example, with monopropellants such as H_2O_2 (1300°F) or N_2H_4 (up to 2000°F) used with conventional metals such as the stainless steels (melting point 2600°F). As long as the simple metal chamber structure is capable of taking the pressure and vibration loads at the flame temperature, a satisfactory, simple and low-cost thrust chamber can be designed. There is some radiation heat loss from the outside of the chamber wall, but this is relatively small. The process here is one of essentially thermal equilibrium, and the steady-stage heat transfer from the hot gases to the wall approaches zero. While the thrust chamber and propellant feed system design is simple, the performance is very low, and this type of rocket is not useful except in very special cases where cost, simplicity, and schedule considerations are of great importance.

4. <u>Turbine Exhaust Gas Cooling</u>

In the case of a turbopump-fed liquid propellant rocket engine (usually relatively high thrust and long duration), the turbine which drives the propellant pumps provides a small flow of turbine exhaust gas which can be used for some limited cooling applications. The turbine exhaust gases are typically one to three percent of the total propellant flow at temperatures of perhaps 800 to 1300°F, usually at a very fuel-rich mixture ratio and at relatively low pressures of usually less than 25 to 50 psia when emerging from the turbine. The cooling capacity of this gas is, therefore, limited not only by the low mass flow and the small available but also by the maximum temperature rise that can be absorbed without exceeding the maximum tolerable metal wall temperatures.

Materials such as special heat-resistant alloys, e.g., 347 stainless, Inconel 718 and Hastelloy are used to temperatures of 1900°F.

The best application known to the authors is nozzle skirt cooling in liquid propellant rocket engines, where the chamber and converging section, throat region and part of the supersonic diverging section are cooled by another cooling method such as regenerative cooling. The turbine exhaust gases can be injected as a film coolant along the gas side nozzle wall surface (where pressures are low enough to permit) and/or used as a coolant in a jacket along the nozzle surface and expelled at the chamber exit for the performance addition. The Rocketdyne J-2 engine (Ref. 4) utilizes exhaust gas as a film coolant. The Rocketdyne F-1 engine (Ref. 5) as shown in Fig. 2 utilizes the exhaust gas simultaneously as a convective and film coolant in the nozzle skirt. This cooling technique permits a radiation heat loss from the outer wall of the jacket, which is lower than that emitted from a radiation cooled nozzle skirt, thus minimizing the heat protection requirement for a vehicle boat-tail.

Using the exhaust gases for convective cooling as long as the gas pressure in the cooling jacket is above the free stream pressure of the main thrust chamber exhaust, it is permissible to have some leaks on the inner wall of the jacket, and, thus, the design does not have to be failure proof. This permits the designer added flexibility in coping with thermal stresses. Thermal stresses and deflections in this gas cooling jacket are severe because of the high operating temperature for both walls. Temperatures and thermal gradients, particularly during start-up, are quite severe. The heat transfer in turbine exhaust gas cooling is essentially a steady-state equilibrium process. Using the combined method with part of the gases for convective cooling and part for film cooling appears the most advantageous from wall temperature and performance standpoints.

5. Insulation Methods

The use of various types of insultation introduces thermal resistance into the heat flux path and, thus, reduces the amount of heat that must be absorbed or disposed of by the cooling mechanism. Additionally for a given cooling capacity, the added resistance permits an increase in the capability to handle hotter temperatures or higher chamber pressures. The application of insulation is not by itself a cooling method but an auxiliary means to enhance the capacity of other cooling methods. Insulators can be applied to the gas-side surface of a radiation cooled skirt, of a metal heat-sink wall, or the inside of the inner wall of a regenerative liquid-propellant-cooled jacket or a turbine gas-cooled jacket. In these applications the heat flux transferred to the wall is reduced. It can also be applied to the outside of the outer wall in a turbine gas-cooled jacket or a radiation-cooled refractory metal wall in order to reduce the heat rejection loss to the environment near the chamber such as in a "buried" installation inside a vehicle; however, this increases wall temperatures.

There are several types of insulation: coatings which are painted or dipped onto a metal surface and subsequently hardened or baked, such as certain silica, asbestos, or ablative containing formulations; thin ceramic coatings of refractories applied to metal surfaces, such as alumina, thoria or zirconia; special sleeves or separate walls made of ceramic, ablative, asbestos-filled rubber, graphite or other refractory material. There are numerous methods and materials in each category.

All of these insulators have possible deficiencies. They are not always resistant to the oxidation or chemical corrosion action of the hot reaction products, and they do not always readily resist the erosive action of the hot gases; many insulation layers cannot take the required high temperature (they become soft or peel off), and most of these materials cannot readily take thermal shock and stress. The coefficient of thermal expansion is generally very different from the base wall material (usually metal), and the resulting thermal expansion causes flaking or cracking failure of the insulating layer.

A special case is the use of pyrolytic graphite, which has high and low conductivity directions. Ingenious designs may permit the preferential conduction of absorbed heat to heat sinks (e.g., regenerative cooling jackets in special locations) allowing a reduced heat transfer problem in critical regions, such as the throat area. Pyrolytic graphite designs have not as yet found their best applications.

Another special case is that of a "contact resistance," where a controlled contact between two layers of the metal chamber wall introduces a high thermal resistance to the flow of heat. Differential thermal expansion between the hotter inner wall and the cooler outer wall and the problem of designing for a close, repeatable contact between two surfaces are not easily solved.

The insulation method is used extensively in solid propellant rocket applications. A rubber or plastic-type layer is often bonded to the metal case wall to form an insulating liner. This rubbery polymeric material often contains asbestos fibers or other insulators. The propellant, in turn, is bonded to this insulating liner material and cured inside the liner. Fiber-reinforced plastic material (phenolic resin impregnated with glass fibers) in the form of sheets, sleeves, cones, etc., is also effectively used in solid propellant applications; special glues and bonding methods are necessary to install these individual plastic insulating parts and keep them in place (without gaps) throughout a wide spectrum of environmental conditions (vibrations, moisture, ambient temperature cycling, etc.).

The nozzles used in solid propellant rocket motors and in uncooled liquid propellant chambers are, in most cases, assemblies of insulating materials designed to reduce the transmission of heat to the uncooled outer wall. Graphite, tungsten, molybdenum, and silicon carbide are favorite material for throat inserts. Under more severe thermal conditions, these are often backed up or potted into complex assemblies of other insulators so as to permit adequate thermal differential expansion, absorption of the heat, lightweight, dimensional stability and structural integrity.

Improving the performance of insulators requires devising of better insulator materials, not only with good thermal resistance, but also with an expansion that matches that of the base metal wall material. It should also exhibit good bonding characteristics and have good fabrication properties. With liquid propellants, insulators with combined film-regenerative methods appear very advantageous. For solid propellant applications, the insulator material can be improved to be chemically more inert to the solid propellant, particularly during curing and long-term storage.

5. Dump Cooling

This Rocketdyne-developed concept (Ref. 6) employs low-molecular-weight fuels such as liquid hydrogen or liquid ammonia in a cooling jacket passage which surrounds the thrust chamber. It is similar to regenerative cooling but, instead of recirculating the heated coolant back to the injector and into the combustion chamber where it will enter into the combustion process, in dump cooling this fuel is dumped overboard at the exit of the nozzle. Gasified coolant is ejected at supersonic velicity through a separate set of nozzles, thus adding to the total specific impulse of the vehicle. Because the coolant gases coming from the dump cooling jacket are not as hot as those coming directly out of the combustion chamber, the overall performance is somewhat diminished. The dump method of cooling is not suitable with propellants that form high-molecular-weight vapors.

Improvements in the material of construction to permit a higher wall temperature will improve the capabilities of dump cooling. Hydrogen is particularly an excellent fluid for this purpose because of its low molecular weight. Even when it is heated to relatively moderate gas temperatures, the specific impulse of the heated hydrogen coming from the cooling jacket is appreciable. In addition, the performance penalty is not substantial if the total flow through the dump cooling jacket is a very small fraction of the total fuel flow, say five to ten percent of the total fuel flow.

Dump cooling seems to be particularly suitable for high-area-ratio, upper-stage, relatively low-pressure rocket engines or in nozzle skirts of higher pressure engines where heat fluxes are at low levels.

6. Self-Cooling

The development of high-energy aluminized propellants with flame temperatures in excess of 6000°F has created the need for a new non-eroding nozzle throat insert material for solid rocket motors. The nozzle wall temperatures attained with advanced propellants during a firing period of reasonable duration exceed even the maximum practical operating temperature of uncooled tungsten. However, by filling or infiltrating porous tungsten with a second material which can be endothermically transformed into a vapor phase, it is possible to reduce the wall temperature to a tolerable value. This concept has been termed "self-cooling" by Schwarzkopf and Weisert (Ref. 7).

The composite insert initially acts as a heat sink until the surface temperature achieves the boiling or decomposition point of the coolant. As the coolant vaporizes, an interface forms which recedes so as to leave behind a porous tungsten zone through which the coolant vapor flows. The endothermic vaporization of the coolant increases the heat sink capacity of the insert, and the coolant vapors absorb additional heat from the porous tungsten, thus reducing the surface temperature.

~~This cooling technique has already been successfully applied in the Polaris missile system where a silver-infiltrated tungsten insert has replaced solid refractory metal throat inserts.~~ It appears to be most applicable to motors using high-energy metallized propellants at low to moderate chamber pressures where solid tungsten throat inserts are marginal. Extremely porous matrices can be fabricated to achieve a large coolant reservoir and thus extend the operational duration. The high-porosity zone is covered with a thin surface layer of a low-porosity tungsten composite which can satisfactorily resist erosion.

Analytical methods for predicting the thermal behavior of refractory metal composites have been developed and have been successfully used to help guide experimental and development studies. The analysis indicates that superior self-cooling inserts, as determined by surface temperature rise, will result from a low coolant boiling temperature, a slow interface recession rate achieved by a high latent heat of vaporization or heat of decomposition, and a high matrix thermal conductivity. The choice of coolant and matrix porosity depends upon the particular application. Coolants such as silver, copper, zinc, and lithium hydride have received the most attention.

Substantial progress has recently been made in conducting firing tests and developing fabrication methods for the self-cooling concept, but more work is necessary to develop satisfactory inserts in large sizes and to control matrix porosity gradients. Improved coolants are necessary if the method is to find application at high chamber pressures and for long firing durations. A variation of the self-cooling technique based on utilizing a relatively thin refractory metal throat liner backed by a pool of boiling metal is also under intensive investigation (Ref. 8).

7. <u>Ablative Cooling</u>

Ablative cooling involves the use of wall materials which undergo endothermic thermal degradation and consequent mass removal but which permit sufficient material to remain to preserve sufficient structural integrity.

Initially, as summarized by Adams in 1959 (Ref. 9), a great deal of effort was expended in the successful development of ablative materials for use as heat shields for ballistic missile and satellite re-entry applications. This work indicated potential applications for rocket thrust chambers. Unfortunately, much of the re-entry technology was not directly applicable to thrust chambers because: (1) The combustion gas temperatures in thrust chambers are lower by about a factor of three, (2) The combustion gas composition, and, therefore, the chemical behavior, is markedly different, (3) Surface radiation and boundary layer transpiration effects are usually negligible in thrust chambers, (4) Particle impingement effects may occur with certain solid motor applications. Accordingly, during the past few years a correspondingly large effort has been made to develop ablative cooling for thrust chambers (Ref. 10).

Char-forming plastics reinforced with organic or inorganic fibers have proved to be superior for thrust chamber applications. Commonly-used materials include phenolic and epoxy resins reinforced with oriented high-silica glass, carbon or graphite fibers. Initially a resin-bases ablative wall acts like an ordinary heat sink, but a low thermal conductivity causes the surface temperature to rise very rapidly. At about 1000°F, the resin decomposes endothermically into a porous carbonaceous char and pyrolysis gases. As the char increases in depth and temperature, the pyrolysis gases may undergo additional endothermic thermal cracking as they percolate through the char and into the combustion gas boundary layer. With silica reinforcement, as the surface temperature approaches 3000°F, a highly endothermic reaction with carbon in the char may take place, providing additional thermal protection. Above this temperature, the silica will melt to provide a very viscous film which may protect the char from chemical erosion, but which itself may be eventually subject to surface erosion by viscous shear. At even higher temperatures, additional silica reactions and vaporization may take place. With carbon and graphite fibers, melting does not occur. In this case, oxidizing atmospheres can cause rapid surface erosion. The relatively low thermal conductivity of the char layer restricts the transfer of heat to the virgin ablative material. Head soak-back effects can continue after engine shut-down.

In rocket engine applications where regenerative cooling cannot be easily applied and where limited dimensional changes are tolerable, the use of ablative materials is quite attractive. Thus, in low-thrust space engines where insufficient fuel coolant is available for regenerative cooling (Ref. 11), ablative materials are used in the combustion chamber and exit cone regions. With relatively non-oxidizing or low combustion temperature propellants, suitable ablative materials can even be used in the throat region without incurring a performance loss. They are used as liners for exit cones of solid rocket motors. For high-thrust liquid propellant engines, ablative materials may be used for skirt extensions.

The self-regulating type of thermal protection afforded by ablative materials makes them particulary suitable for throttleable engines. Ablative chambers have been particularly useful for pulsing or cycling operation, where individual pulses may be very short in duration, approaching a few milliseconds.

A number of potential problems must be considered when ablative cooling is to be used. It has sometimes been difficult to control the fabrication process and to obtain reproducible properties for the char and virgin material. Relatively minor variations in the manufacturing process may cause severe performance degradation due to spalling or tearing apart of the ablative materials induced by thermal stresses. Other potential problems include outgassing of the binder after engine shutdown, the effect of long-term exposure to vacuum, uneven erosion with combustion chamber liners due to injector nonuniformities, and severe flow of molten glass.

Fig. 4 shows a cross-sectional photograph of an ablative wall after firing. Both charring and surface are indicated. The required thickness of an ablative material to withstand a given environment depends mainly on the char rate and the surface erosion rate. Some of the major factors influencing these two rates have been considered from a design standpoint by Tick, et. al. (Ref. 12). They show that the two dominating properties are the thermal conductivity of the char layer and the density of the virgin material. The lightest design may involve a composite wall including a high density ablative of thickness just equal to the surface erosion loss, backed-up by a low density ablative with the best possible charring properties.

In the absence of appreciable surface erosion, the char rate for a given ablative material is mainly dependent on the propellant combination and to a much less extent on the local combustion gas pressure. Correlation of a large amount of experimental data has been satisfactorily achieved with the proportionality of the char depth to the square root of time.

Surface erosion rates depend strongly on the surface temperature and the combustion gas composition and pressure. Erosion for high flame temperature propellant systems appears to be controlled mainly by melting and partial vaporization of silica for refrasil phenolic. For phenolic-nylon with LOX/RP-1, erosion is controlled mainly by surface combustion of the char. With the F_2/H_2 propellant system, surface erosion may be negligible for carbonaceous chars from phenolic-nylon and carbon or graphite-reinforced phenolics.

The capabilities of ablative thrust chambers are continually being extended by: (1) the development of new resin-reinforcement combinations tailored for the particular application, (2) better fabrication methods and controls to improve quality, (3) development of improved thruat inserts, and (4) use of auxiliary film cooling to reduce ablative surface temperature and auxiliary regenrative cooling to minimize required wall thickness. In some cases, pre-charred ablative liners are particularly attractive.

8. Spray Cooling

This singular cooling method employs a single chamber wall where the outside wall surface is cooled by spraying certain liquids directly against the wall (Ref. 13). Liquid metals such as lithium and sodium are particularly suitable for spray cooling because of their low melting temperature characteristics (367 and 207°F, respectively) and an excellent heat of vaporization (greater than 10,000 Btu/lb) and a high thermal conductivity and low viscosity. By spraying these liquids against the wall surface, a high liquid-film coefficient is obtained, and, by coolant vaporization, additional heat can be absorbed.

The principal limitations are concerned with the use of materials capable of withstanding the liquid metal vaporization temperature (1650 to 2400°F), the added complication and weight of supplying and rejecting a third propellant, and the provision of a high-velocity spray technique with good surface coverage. The aforementioned problems may result in a compromise design with pure convection liquid metal self-cooling under some circumstances. Ingenuity in design could make it possible to find applications for this method since no current designs employ this method.

9. Regenerative Cooling

In the past, the regenerative cooling method has been the mainstay of all cooling types and is used in most large engine liquid propellant engines (Ref. 14). With this method the coolant is first circulated in a jacket with a thin metal wall .010 to .080 inches in thickness separating the coolant from the combustion gases and is then passed through the injector into the combustion chamber. Fuel is generally used as the coolant, principally because its total allowable heat capacity and heat transfer properties (Ref. 15) are generally the highest and because it presents a non-oxidizing atmosphere to the heated inner wall surface. In some cases the use of oxidizer cooling may be indicated particularly where a fuel exhibits exothermic decomposition (N_2H_4), where the oxidzer endothermically disassociates upon passage through the cooling jacket (N_2O_4) or where design mixture ratio is higher so as to provide only a small fuel flow. Coolants may be subcritical or supercritical in pressure and temperature, operating in a boiling or single-phase model. Wall surface temperature becomes limited by material considerations both from a surface deterioration and stress limitation standpoint. Commonly used wall materials are stainless steel, nickel, and Inconel with temperature limitations around 1600°F. Thermal conductivity dictates occasional use of copper or aluminum, but the minimal strength of these materials with increased temperature and weight considerations indicates low stress level applications. Thrust chamber walls are generally composed of a bundle of long coolant tubes which are formed to the nozzle contour. At the smaller thrust and pressure levels a milled channel or double-wall approach appears advantageous.

The principal high chamber pressure limitations become those of coolant pressure drop, wall thermal and pressure stress, and the heat conduction limits for defined minimum wall thicknesses. At low chamber pressures and thrust levels, the small amount of coolant relative to the large wall surface results in a large coolant bulk temperature rise with attendant coolant bulk decomposition and near achievement of the saturation temperature with high-pressure drop resulting from a small available wall-to-coolant temperature difference. Coolants such as N_2H_4 and RP-1 have allowable wall temperatures (400°F and 800°F, respectively) above which undesirable coolant decomposition at the wall surface occurs.

Lower thrust and pressure feasibility limits may be reduced by insulating wall coatings or by limited nozzle or combustion zone surface area. Upper feasibility limit increases in pressure and thrust require the development of high-strength, high-conductivity materials with high allowable wall temperatures. Candidate materials appear to be the refractory metal family (Mo, Cb, Ta, W). However, it is apparent that considerably material development is necessary before these materials offer an advantageous solution. Wall coatings previously mentioned, such as zirconia, thoria, and magnesia, provide both an allowable high wall surface temperature as well as a reduction of heat input to the regenerative coolant. These coatings are, however, subject to spalling and thermal stress failure due to the high temperature gradients involved.

Future applications of regenerative cooling for advanced chambers will probably involve supplementary cooling by some of the other methods described in this chapter.

10. Film Cooling

The film cooling method offers a unique approach with simple thrust chamber wall construction allowed. Coolant is injected along the gas-side wall surface by means of tangential or nearly tangential coolant holes, slots or louvers. If a small-length combustion chamber design is employed, injection can be provided at the injector face, and the film cooling effect will persist to the throat region. In a fully film-cooled design, injection points are located at incremental distances along the wall length. The wall temperature essentially assumes the film coolant value, increasing from the inlet coolant injection temperature to the final allowable wall temperature value. Judicious spacing of the injection locations can provide an optimum wall temperature distribution.

A typical lengthwise wall temperature distribution between slots may vary from -400°F to 1200°F with liquid hydrogen film coolant, for example. It can be seen that an inefficiency of cooling results from the overcooling of the wall at the injection region location. Similarly the large variation in wall temperature can contribute to severe thermal stresses. Generally, except for low thrust engines, a small coolant flow percentage (.5 to 4 percent of total propellant flow) is involved, and uniform circumferential injection of the film coolant becomes a difficult task. Injection velocity and injection manifold design become important because proper matching of film coolant to mainstream gas velocity can contribute to a coolant flow reduction as a result of improved efficiency. Unique designs for injection manifolds, or the use of non-structural temperature equalizing materials of high conductivity and high allowable temperature (copper or refractory metal) which may be provided as a chamber liner, become worthwhile approaches to this problem.

Cooling with liquid film coolants (Ref. 16) below critical pressure results in sensible heat absorption by the coolant to the critical temperature point where vaporization occurs. From this point, the heat of vaporization is utilized until complete gasification occurs. Further film cooling benefit can often be realized by using this gaseous region. Film cooling at supercritical pressure and/or supercritical temperature (Ref. 17) occurs by sensible heat absorption of the film coolant without a phase change.

A principal disadvantage to the film-cooled design is the performance loss associated with uncombusted coolant adjacent to the wall surface. If at high pressures supplementary film-cooling becomes necessary, the performance degradation caused by the coolant may result in the negation of the chamber-pressure-derived specific impulse advantage. Film coolants such as H_2 or NH_3 show promise because a low molecular weight will contribute to a high temperature-to-molecular-weight ratio without coolant combustion, and, hence, no appreciable performance degradation will occur. In consideration of the performance loss aspect, it appears that the application for film cooling is either for supplementation or in large thrust designs where a small percentage of the total propellant flow is required. Surface film stability under high turbulence, phase change along the cooled length, and endothermic or exothermic decomposition are problems which require further study.

The film cooling method was used early in rocket motor designs, notably the German V-2 engine. Similarly, film coolant is often provided by the injector design configuration to provide thrust chamber wall cooling in the combustion zone. Fig. 5 illustrates a current completely film cooled hydrogen-oxygen engine design.

11. Transpiration Cooling

Transpiration cooling can be differentiated from film cooling only by a fine line of distinction which lies principally in the injection technique. A transpiration-cooled surface is composed of either a porous sintered material or a fine-pored compressed multiple screen mesh. Coolant is diffused at low velocity through the surface into the hot combustion gas boundary layer with a blocking effect which can reduce the heat flux to the wall (Ref. 18). The transpiration method becomes much more efficient (three to five times less flow) than the film cooling method due to the axial and circumferential uniformity of the wall temperature and low injection velocity with resultant small mixing losses with the mainstream gases.

Surface temperatures again become limited by material considerations, with stainless steel and nickel being the common material choices. Pressure drop-flow relationships for commercially available materials indicate an unsuitability for high-pressure applications due to the large wall thickness required to limit coolant flow or a necessarily small pore size (10 to 25 micron). The latter feature illustrates drawbacks to this cooling method, either as a result of foreign matter plugging of the coolant passages or an unstable coolant flow behavior under small pressure variations (minimum wall thickness and small differential pressure).

Transpiration materials suffer additionally from high imposed thermal stress, since the wall on the coolant side is essentially at the coolant bulk temperature and the gas side is at a maximum.

Transpiration methods appear, however, as the only approach for ultra-high heat fluxes due to the minimal mass flow required as coolant. Limitations appear to be only in the fabrication techniques which are presently in infancy. Future application can be seen in large thrust (1M-6M lb) high-pressure (> 3000 psia) O_2-H_2 and nuclear H_2 fueled engine designs. With the development of refractory materials, wall temperatures can be maximized with a further flow and performance loss minimization. Transpiration-cooled injector faces are presently being employed in current hydrogen-fueled engines.

12. Sacrificial Solid Propellant Cooling

This method is similar in principle to dump cooling but employs solid propellant. A doughnut-shaped or a hollow cylindrical-shaped solid propellant grain is used consisting of a composition which gives a relatively low gas temperature, e.g., a nitrate-type solid propellant or a conventional solid propellant where additional fuel is used to reduce the gas temperature. This forms an annular gas layer of relatively cool temperature which protects the walls and the nozzle throat and diverging section from excessive heat transfer. As with dump cooling, there is a performance loss inherent in this type of cooling method primarily because the low temperature propellant does not have a comparable high specific impulse to the main combustion gas core. The main gas flow is hot and is generated by a conventional solid propellant of high-temperature and high performance. Possible use in a conventional liquid propellant rocket engine of a solid propellant sacrificial outer layer needs further study. Scale effect considering performance losses with varying thrust, duration, and chamber pressures have not yet been examined.

13. Injector Design Modifications

It is well known that changes in the injector pattern of a liquid propellant rocket motor can result in drastic changes in heat transfer and boundary layer characteristics. It is possible to inject additional fuel around the periphery of the injector face (most American practice uses a flat cylindrical injector face) which will give a cool, lower-mixture-ratio outer gas layer to reduce the wall heat transfer rates. The gases in this cool outer layer, however, eventually mix with the hotter gases from the core of the combustion zone and the cooler layer diminishes along the chamber length dependent upon the initial zone width and mixture ratio at the periphery.

The possible variations in injector design are many, and the effects on the wall heat transfer are dependent on the design configuration. Thrust scaling effects occur. Even slight variations in a very small motor have a profound influence as opposed to variations in the center of a large-size injector which have no appreciable effect on the heat transfer to the combustion chamber walls. A definite connection between heat transfer and performance can be seen to result. With higher performance of an injector, a higher heat transfer rate to the wall occurs since most design modifications in injectors change the outer gas layer. Reductions in outer zone mixture ratio will be accompanied by slight decrements in over-all thrust chamber performance.

14. Combined Methods

Two or more cooling methods can be combined to advantage at a given location (Ref. 19). The most prominent possibility appears to be the combined film-regenerative method. With this arrangement, the total heat flux loading normally imposed on the wall surface can be fractioned with a percentage absorbed by a reduced film coolant flow and the remainder by a reduced velocity regenerative coolant. In such a manner, turbopump power or tank pressures are diminished with a compromise on specific impulse as a result of a small film coolant flow. With such approaches, longevity and reliability may be greatly enhanced, since both wall temperature and heat flux level are reduced. Limitations of this method may be shown in the areas of injection method design techniques and in the description of the performance losses associated with the film coolant flow. It is apparent that higher allowable surface temperatures enhance the capabilities of both the regenerative and film method; consequently, material limitation evaluations are of utmost concern.

Combining film cooling with radiative cooling, particularly in low-pressure, low-thrust designs, is of interest. Single-wall refractory metal chambers result with the film coolant boundary providing both a reducing atmosphere to prevent wall surface oxidation and a tolerable radiative wall temperature.

Combined ablative backed by regenerative cooling is particularly attractive in small thrust engines where a small coolant flow is available to cool a large wall surface area and regenerative cooling cannot strongly be employed but can be used for absorption of a fraction of the wall heat input. The ablator in this case would initially serve as a sacrificial coolant and, upon depletion, as an insulator. Combining film cooling with ablation can result in a materially reduced char rate if some performance degradation can be tolerated. With the film-ablative combination, surface oxidation and erosion can be prevented with a considerable improvement provided in the areas of longevity and reliability.

Conclusions

The basic approaches for improving thrust chamber technology have been indicated (e.g., reduce weight and size, increase reliability, performance, chamber pressure, thrust, etc.), and they, in turn, are related to specific cooling problems which arise. For example, a very high-thrust (6M lb), high-pressure (3000 psia) liquid-propellant booster rocket with estimated heat transfer rates of 40 to 60 Btu/in.^{2}sec, as shown in Fig. 6, will probably use regenerative cooling augmented in the areas of high heat flux by film cooling or transpiration cooling. It is concluded that design requirements for future advanced thrust chambers will dictate the use of a combination of two or more cooling methods. Much research and development yet needs to be done to overcome the current design limitations imposed by heat transfer and to extend the capability to cool under more severe conditions. Selection of the most advantageous cooling technique will depend largely on future study and novel design approaches to be developed.

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Figures to be Included

Table 1	Tabular comparison of cooling methods (grouping-resistance, mass transfer, convective, temperature distribution).
Table 2	Heat flux regime vs. cooling spectrum
Figure 1	Radiation cooling limits graph
Figure 2	F-1 chamber (regenerative cooling with turbine exhaust gas cooled extension) (photo)
Figure 3	Cutaway sketch of advanced solid propellant nozzle with heat sink walls and throat insert
Figure 4	Cutaway sketch of ablative chamber with insert
Figure 5	X-8 experimental high pressure film-cooled engine (photo)
Figure 6	Heat Flux Level with chamber pressure

TABLE 2 SPECTRUM OF THRUST CHAMBER COOLING LIMITS

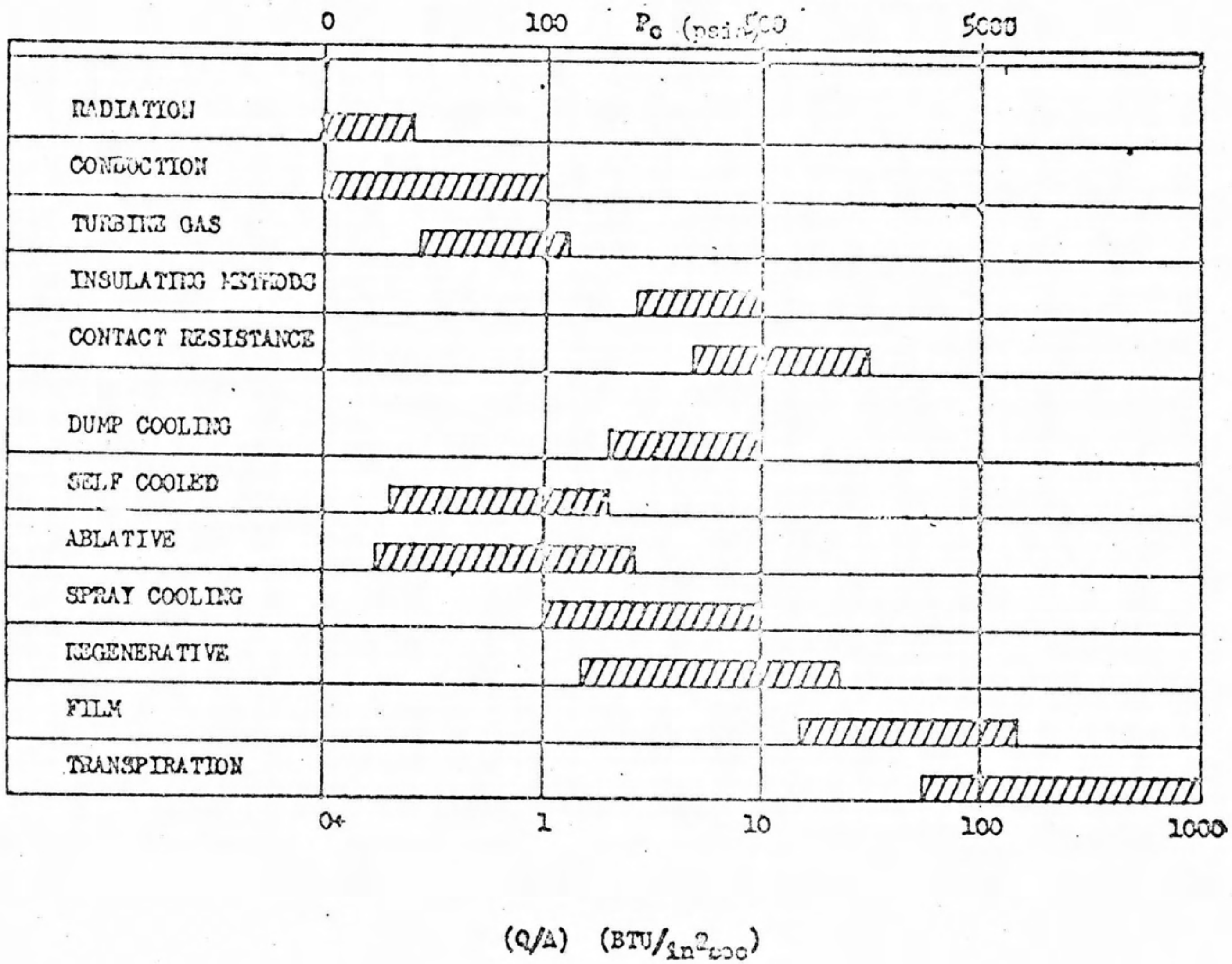


Table #1
figure 3
not included

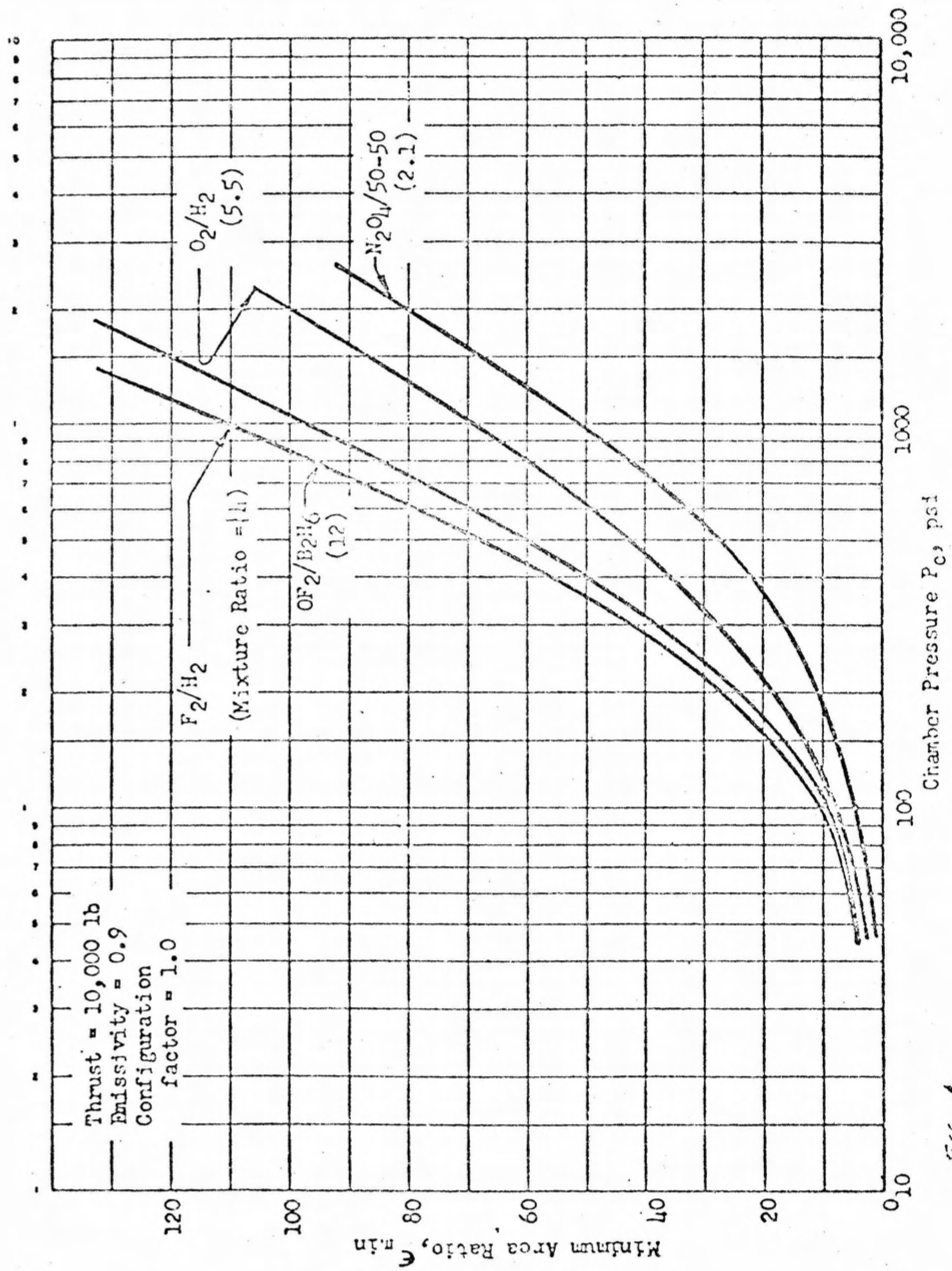
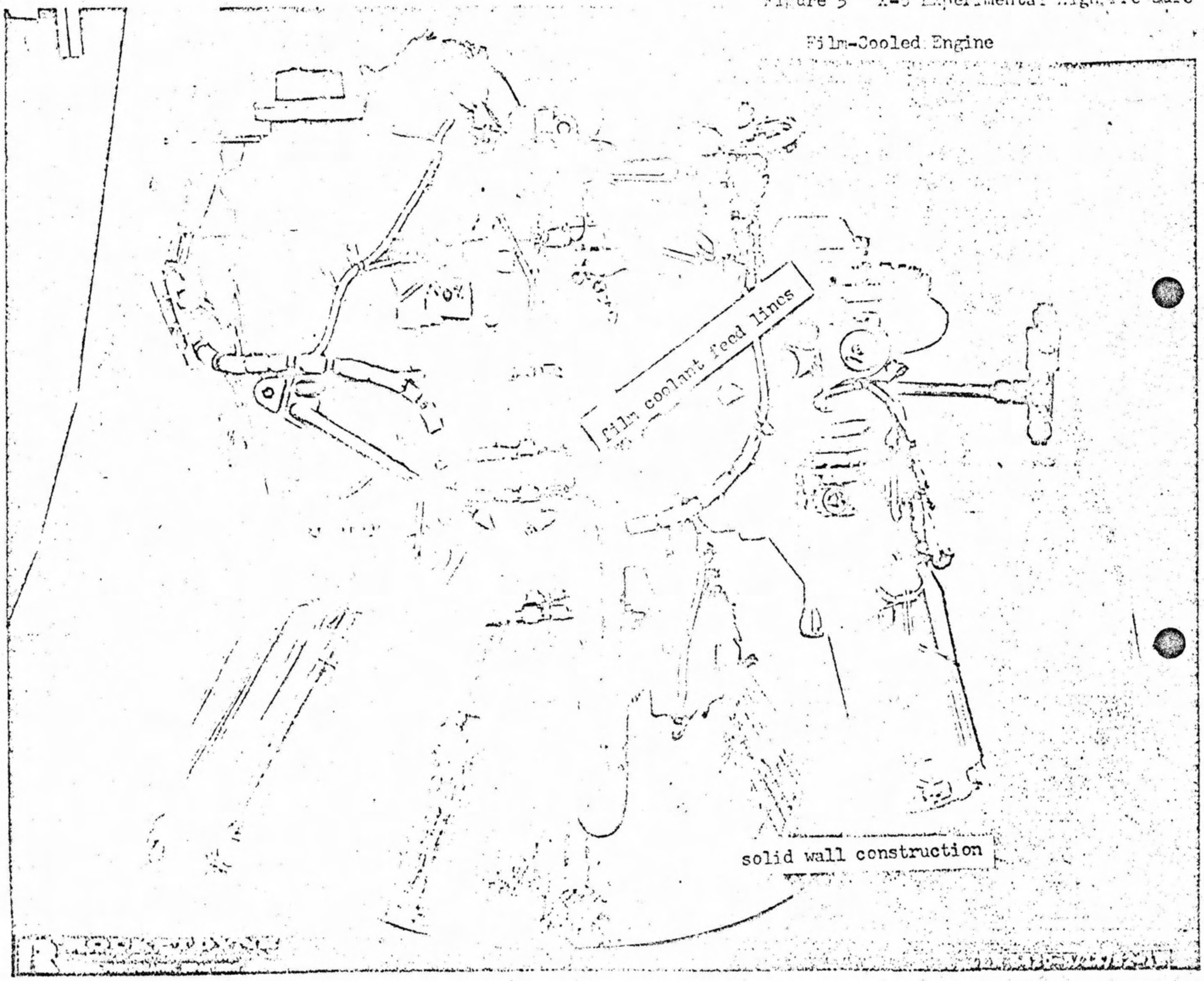


Fig. 1. Minimum Area Ratio for Attaching a Molybdenum Skirt (2500°F)

Figure 5 X-8 Experimental High Pressure

Film-Cooled Engine



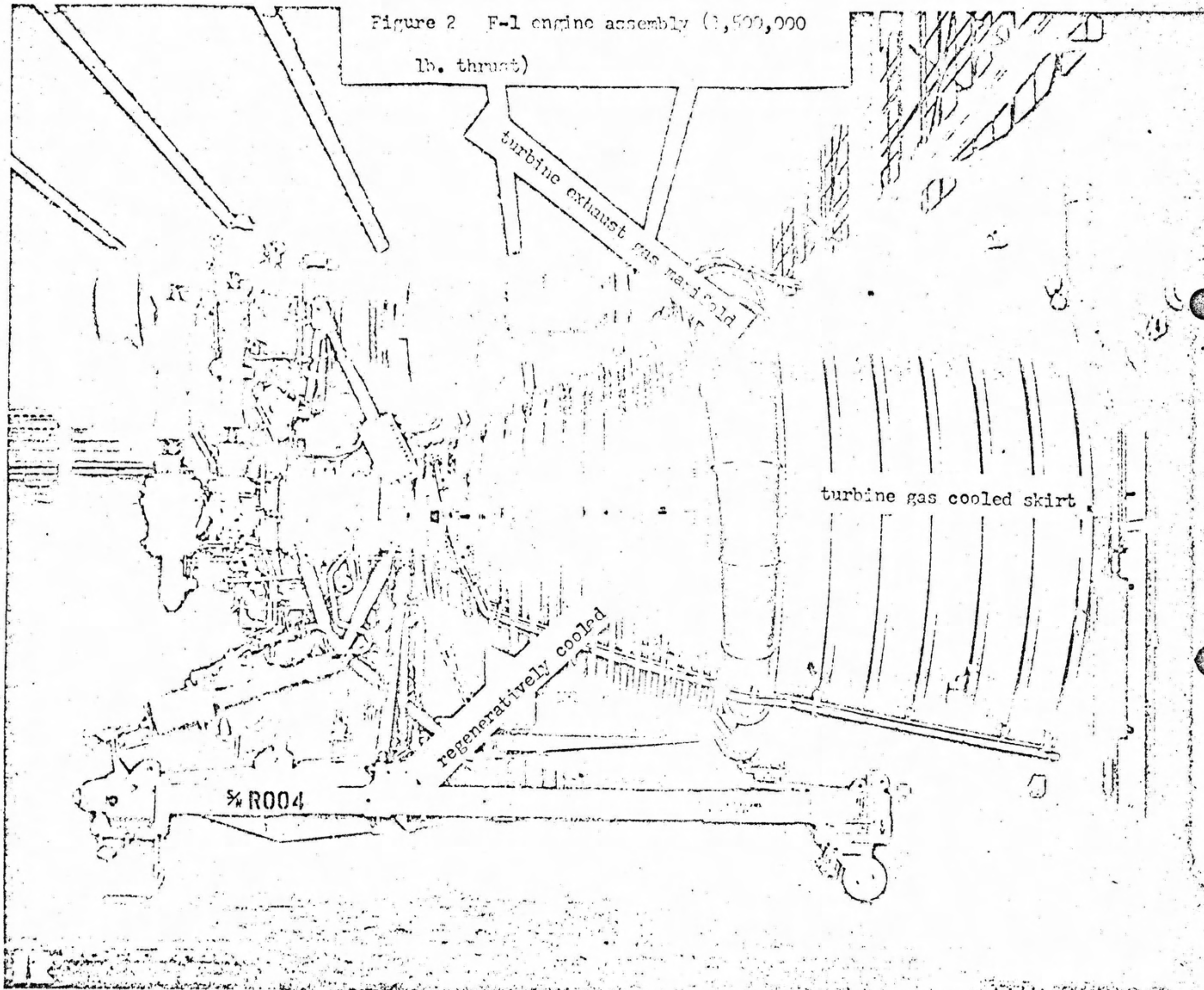
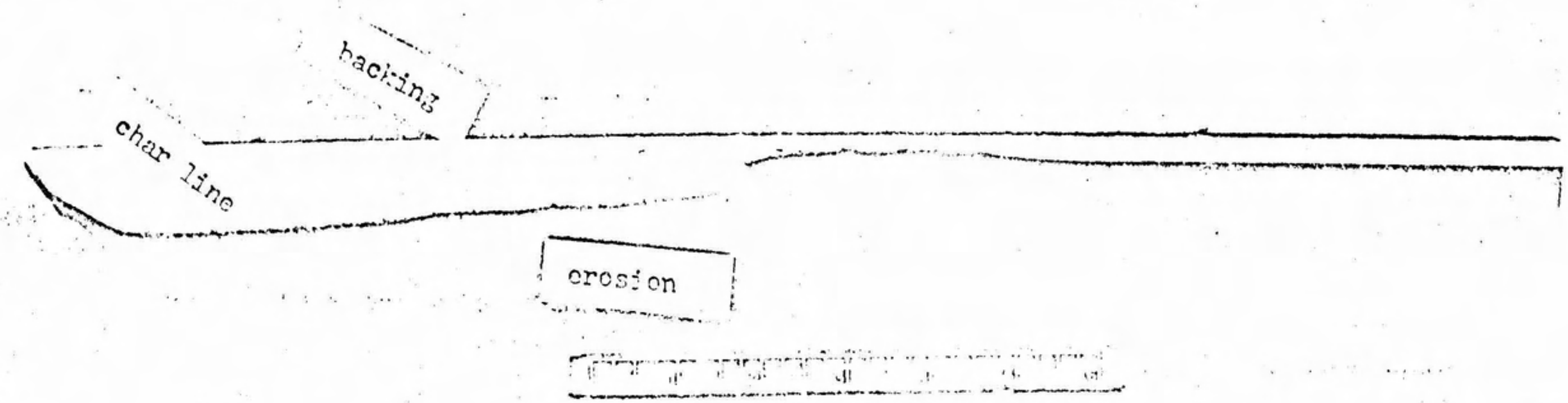


Figure 2 F-1 engine assembly (1,500,000 lb. thrust)

Figure 4   Cutaway Photo of Ablative

Chamber with Throat Insert



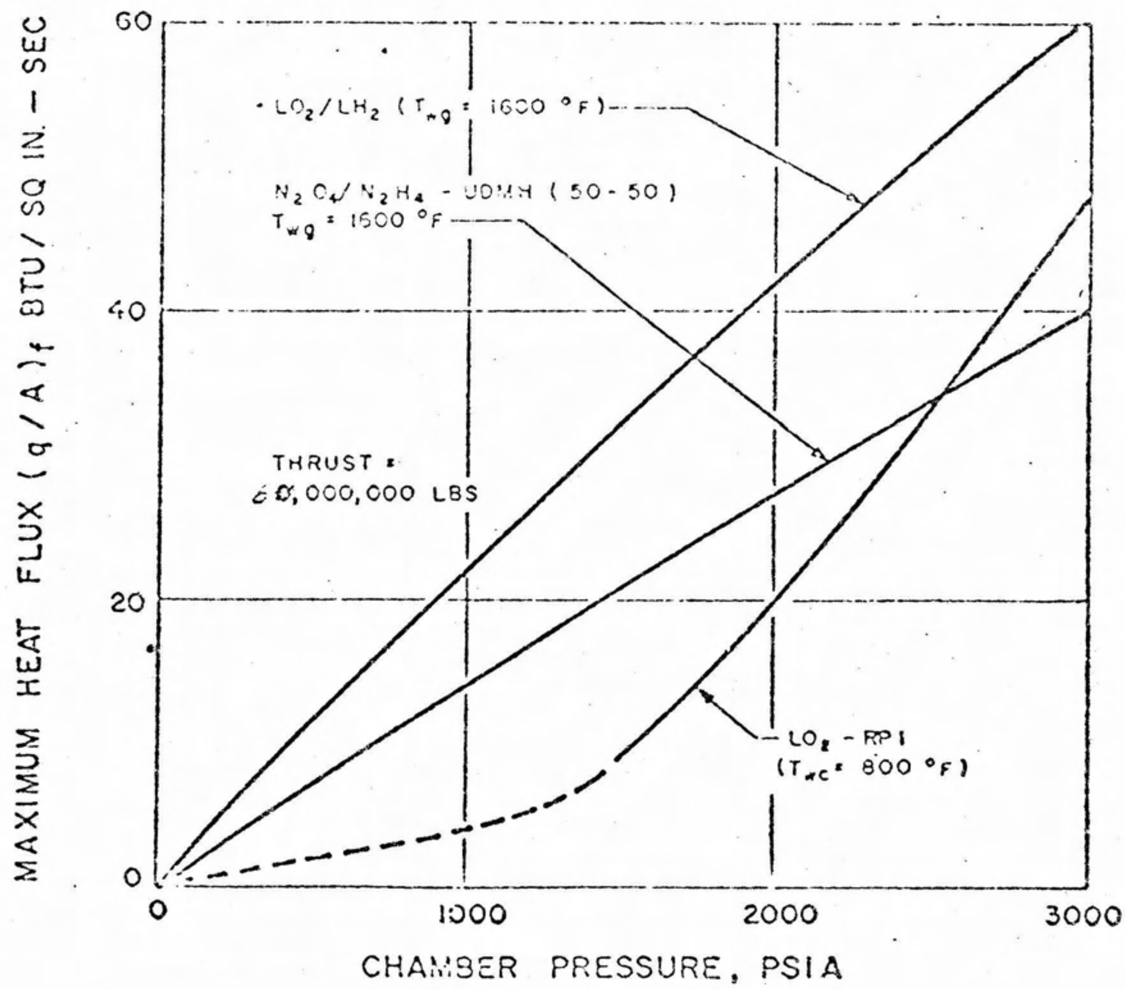


FIG. 6

Maximum Heat Flux vs. Chamber Pressure for $LO_2/RP\text{-}1$, LO_2/LH_2 and N_2O_4/50-50 UDMH-N_2H_4

1-65-19

TABLE 1

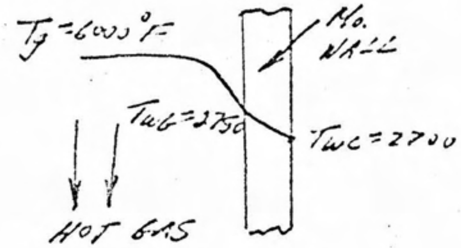
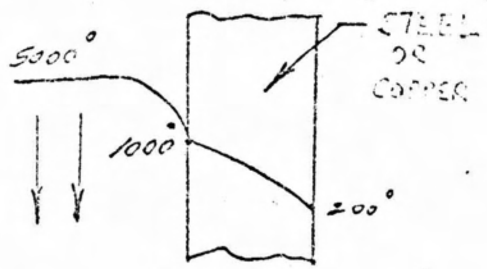
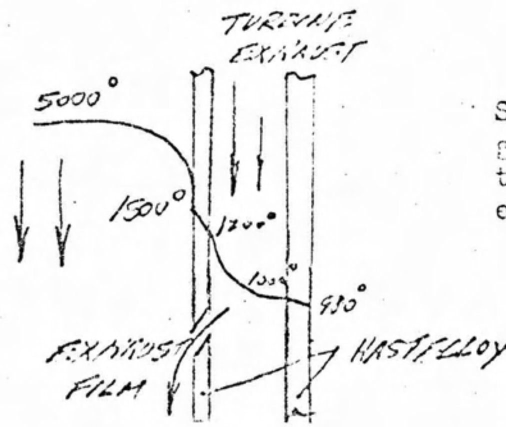
	Name	Description	Temperature Diagram	Application
1)	Radiation Cooling	High-temperature, light-weight, refractory-metal pressure vessel which glows white hot and rejects heat by thermal radiation (reaches thermal equilibrium)		Low-thrust, low chamber-pressure unit for attitude control (e.g., Apollo service module) Uncooled nozzle exit skirt extensions
2)	Heat Sink	Heavy-walled metal shell (does not reach thermal equilibrium)		Short duration, low-thrust, low-pressure units (typical experimental unit for evaluating propellants or feed system features, JATO's, or retrorockets)
3)	Turbine Exhaust	Turbine exhaust gas (900 to 1500°F) is used as a coolant in a cooling jacket and/or as a film cooling fluid		Small amount of available gas and heat capacity limits this to nozzle exit skirt extensions.

TABLE 1 (Cont.)

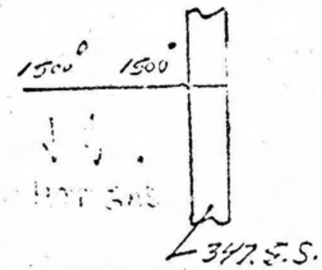
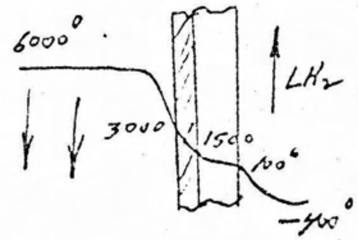
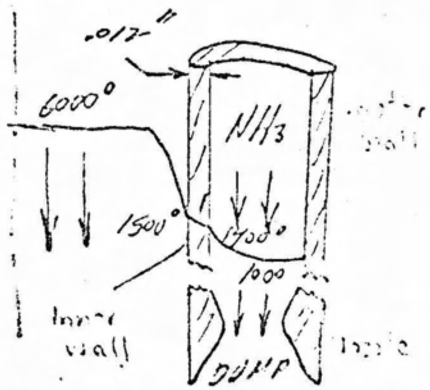
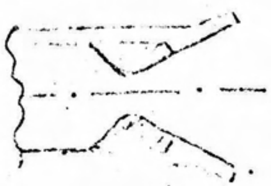
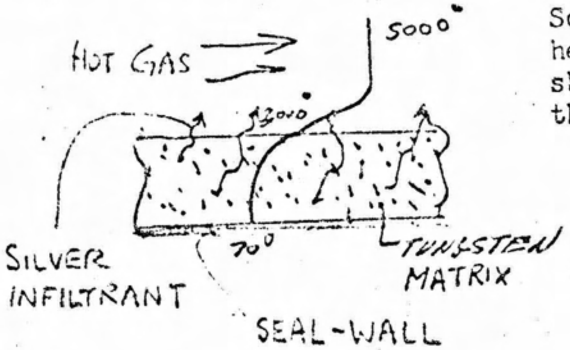
Name	Description	Temperature Diagram	Application
4) Low Flame Temperature	Wall temperature achieves temperature of combustion. Equilibrium condition.		Low performance propellants with low flame temperature, ~~low-thrust, and low pressure~~ ~~low-cost applications~~ (Nitrate solid propellant or H_2O_2)
5) Insulation Method	Low conductivity coating or layer on wall surface for heat flux reduction		High pressure, high performance solid or propellants for high heat flux conditions. ~~Thermal stress limited.~~
6) Dump Cooling 	Fraction of propellant is used as coolant in cooling jacket and eventually dumped overboard for performance gain.		~~Low-pressure~~ systems for ~~reduction of~~ cooling pressure drop, nozzle extension and ~~high-area-ratio~~ points. Generally H_2 or NH_3 fuels.
7) Self Cooling 	Porous refractory metal matrix filled with vaporizing material which transpires through to the gas surface.		Solid propellant designs where heat fluxes are severe or where shear or char erosion is large, throat inserts.

TABLE 1 (Cont.)

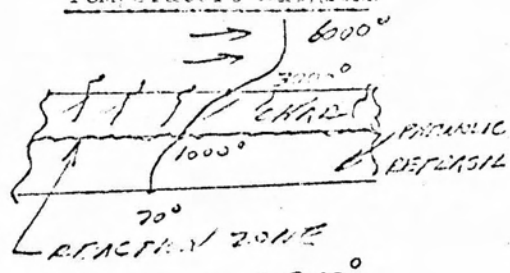
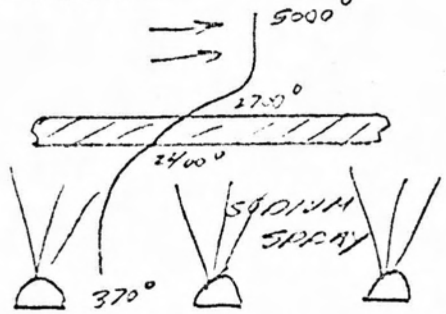
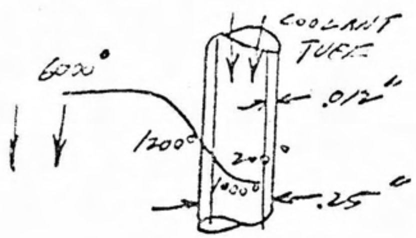
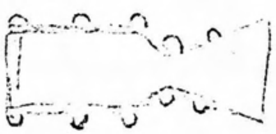
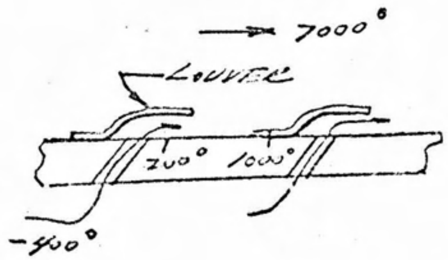
Name	Description	Temperature Diagram	Application
8) Ablative Cooling	Progressive endothermic decomposition of wall surface forming insulating char for passage of pyrolysis gases.		Moderate and low pressure, limited duration designs for combustion chamber and nozzle, skirt extensions, throttleable space engines, ~~[illegible]~~, etc. or attitude control space engine
9) Spray Cooling	Liquid metal (Na, Li) spray with vaporization and heat absorption. Discharge of vapor overboard.		No present applications; possible for solid propellant nozzle throat inserts or high heat flux application.
10) Regenerative Cooling 	One of propellants (fuel) circulated in wall cooling jacket and passed into injector for combustion. Most commonly used method.		Most liquid propellant applications; low and high thrust to a moderate chamber pressure; Atlas, Titan, Thor, Saturn, etc.
11) Film Cooling 	Coolant film injected as a heat absorption boundary between combustion gas and wall surface.		Very high pressure and large thrust applications ~~where low propellant percentages are required~~.

TABLE 1 (Cont.)

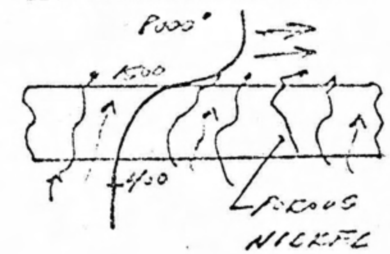
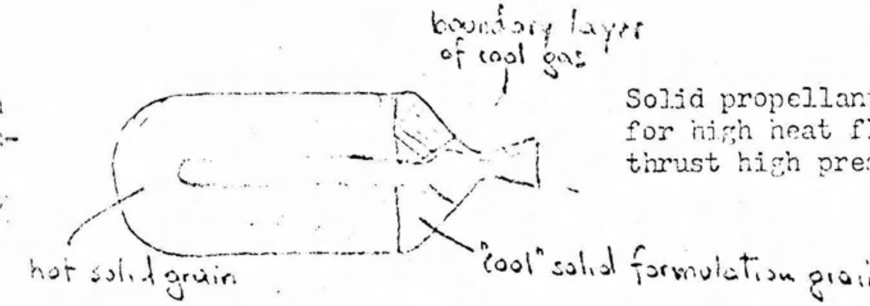
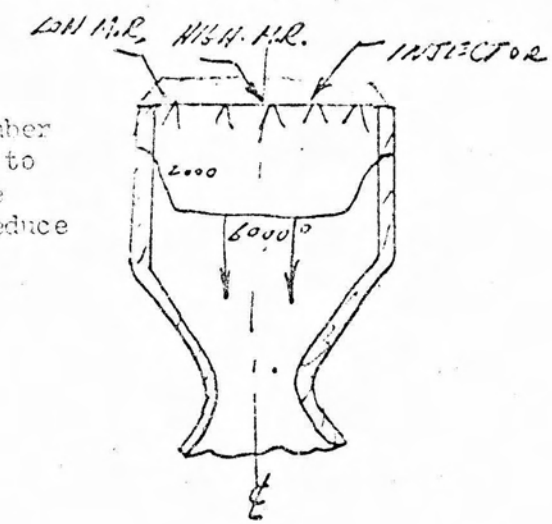
	Name	Description	Temperature Diagram	Application
12)	Transpiration Cooling	Diffusion of mass transfer coolant through thick porous or mesh wall with low velocity. Absorption of heat through wall and blockage on surface.		Nuclear, F_2-H_2 O_2-H_2 fueled designs for ultra high heat flux application with low performance loss
13)	Sacrifical Solid Propellant Cooling	Solid propellant grain composition of low combustion temperature forms gas film coolant layer		Solid propellant nozzle protection applications for high heat flux; large thrust high pressure designs.
14)	Injector Design Modification	Alteration in main chamber gas core mixture ratio to produce low temperature at wall periphery to reduce wall heat input.		Most current large thrust liquid propellant designs employ reduced mixture ratio conditions at the injector periphery.

TABLE 1 (Cont.)

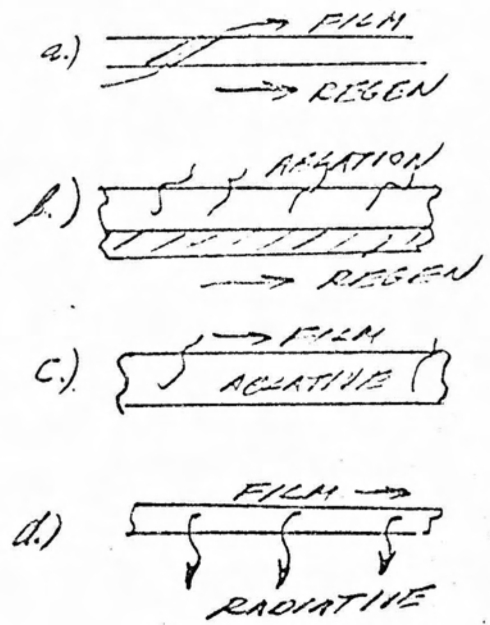
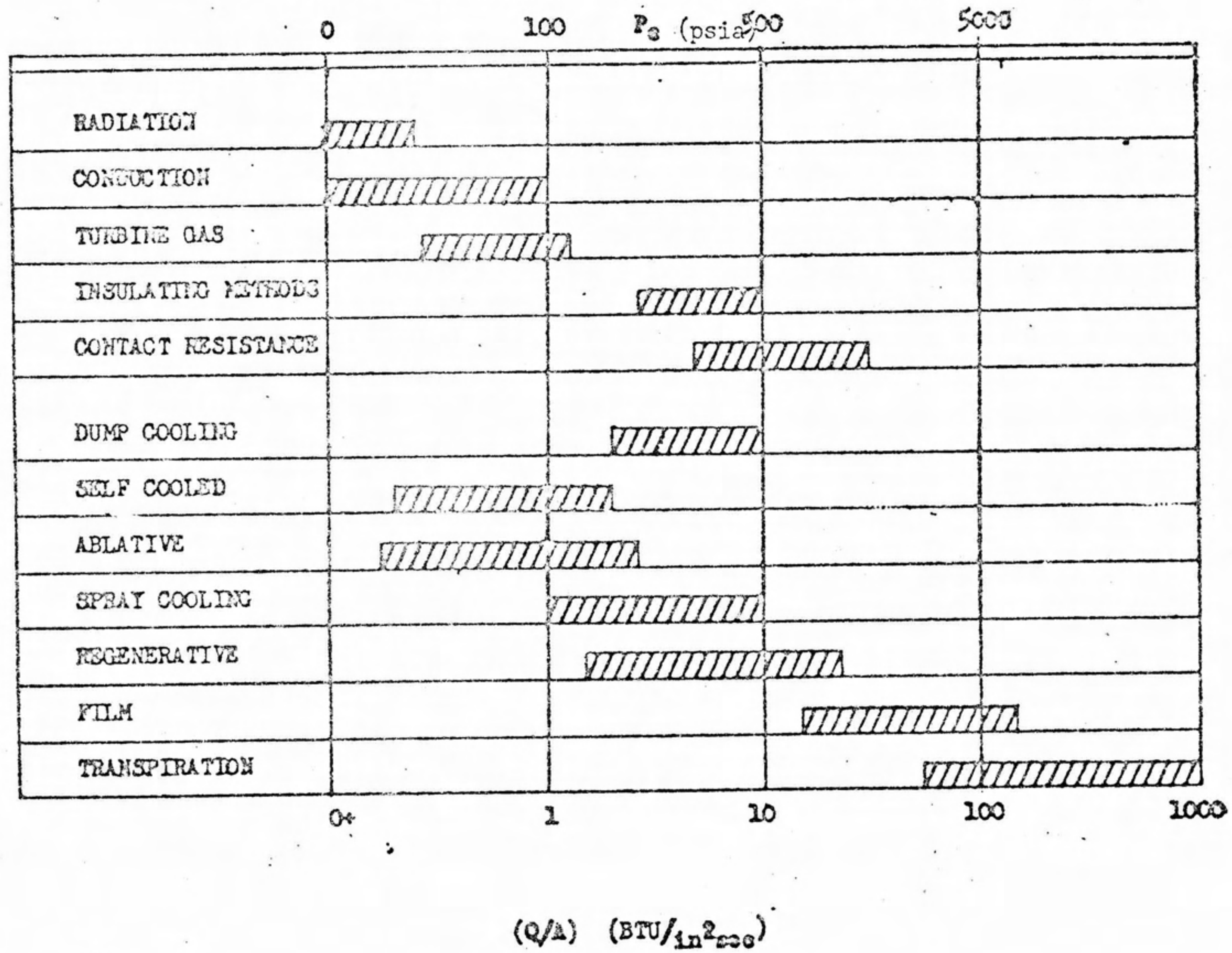
Name	Description	Temperature Diagram	Application
15) Combined Methods	Use of two or more methods to reduce cooling pressure drop, to increase performance, duration, reduce weight, etc.		High pressure, low pressure-thrust, etc., special applications for heat transfer optimization.

TABLE 2 SPECTRUM OF THRUST CHAMBER COOLING LIMITS



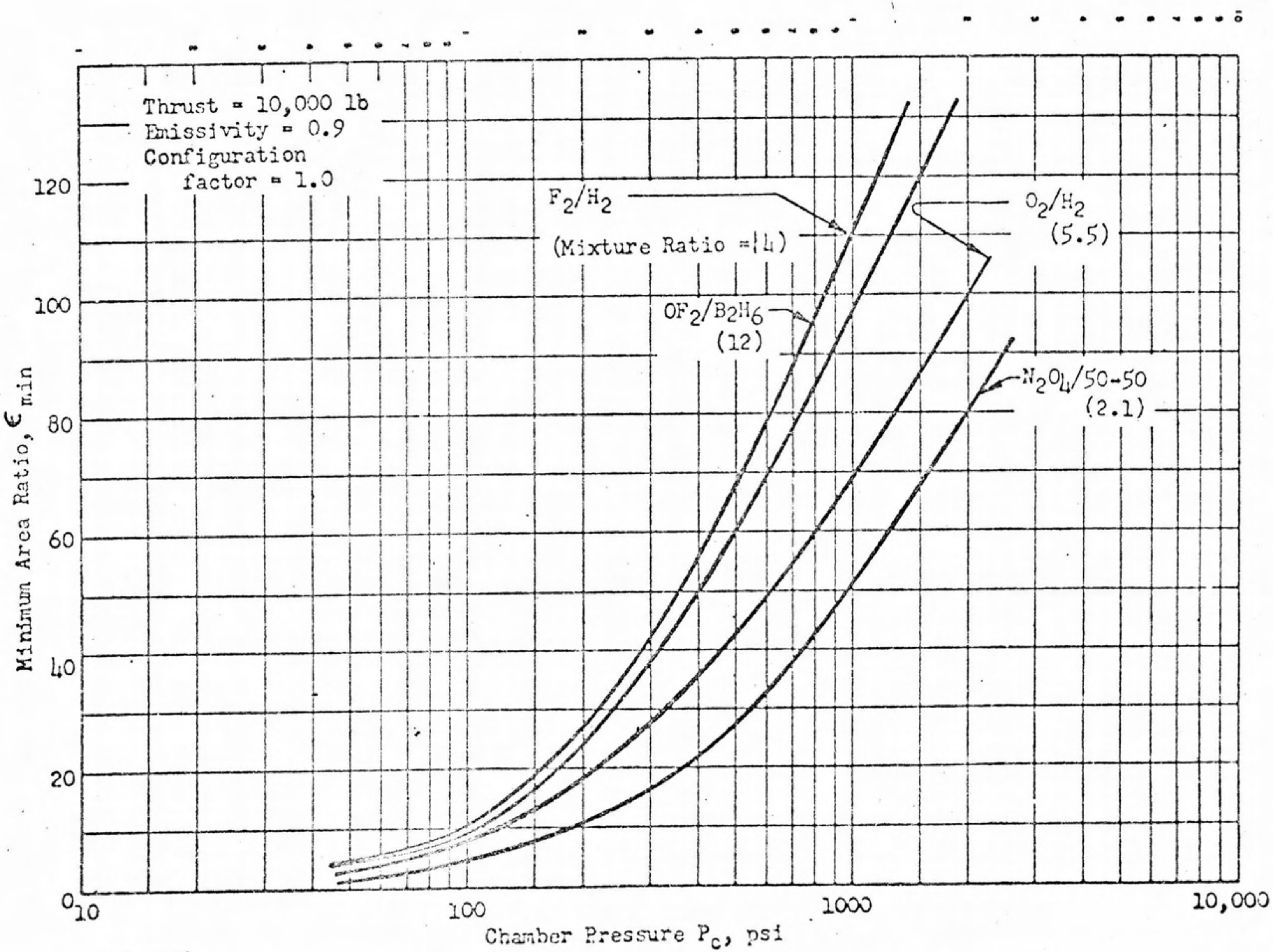


FIG. 1

~~Fig. 19~~. Minimum Area Ratio for Attaching a Molybdenum Skirt (2500°F)

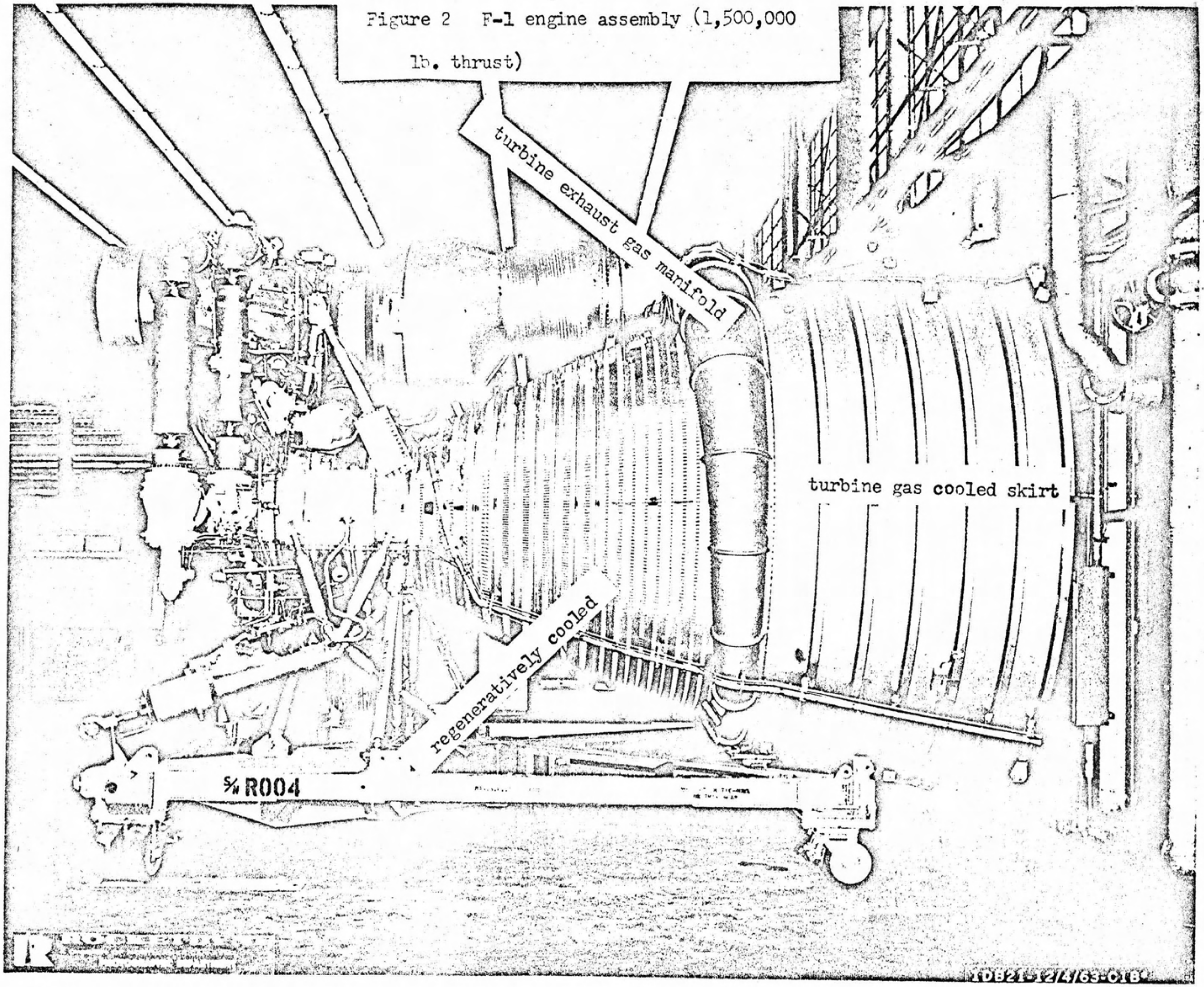


Figure 2 F-1 engine assembly (1,500,000 lb. thrust)

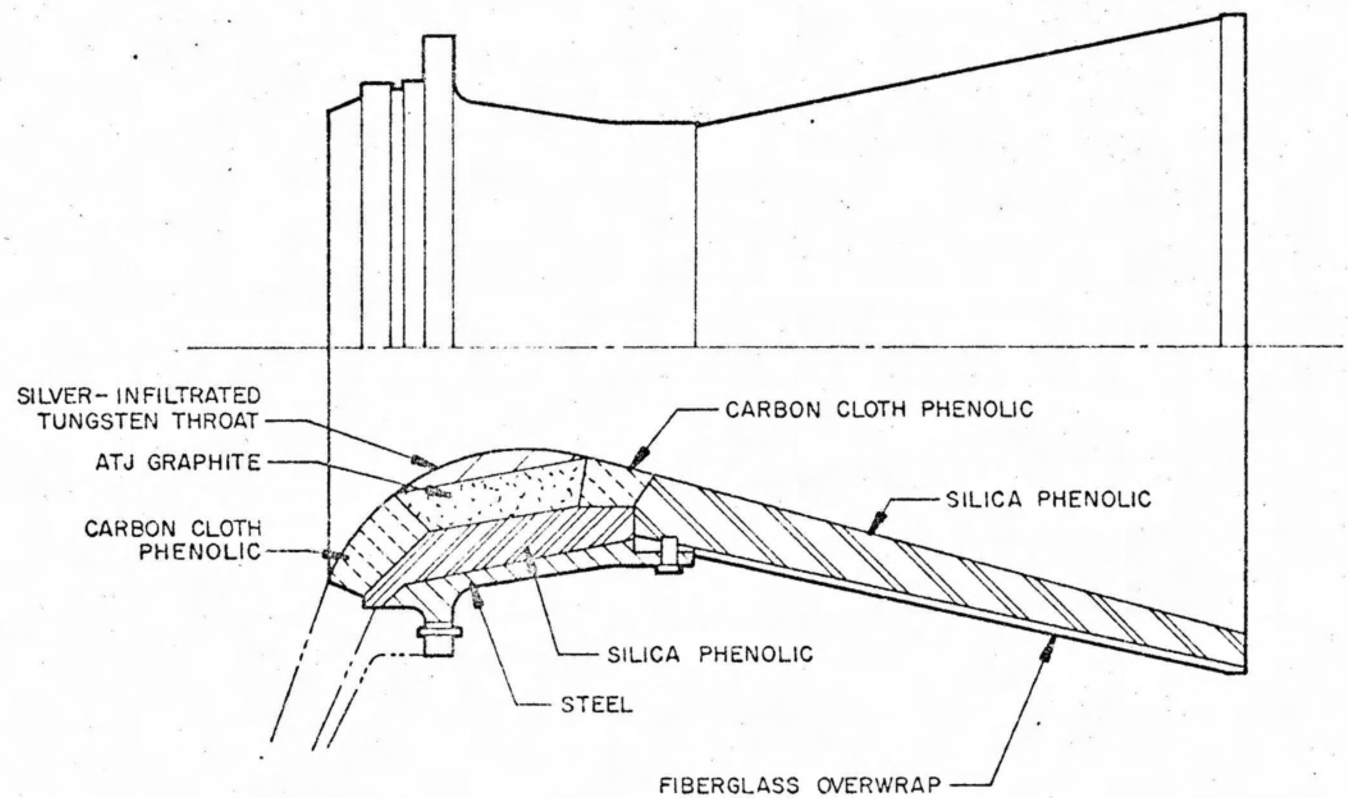


Figure 3. Cutaway sketch of Advanced Solid Propellant Nozzle with Heat Sink Walls and Throat Insert

Figure 4   Cutaway Photo of Ablative

Chamber with Throat Insert

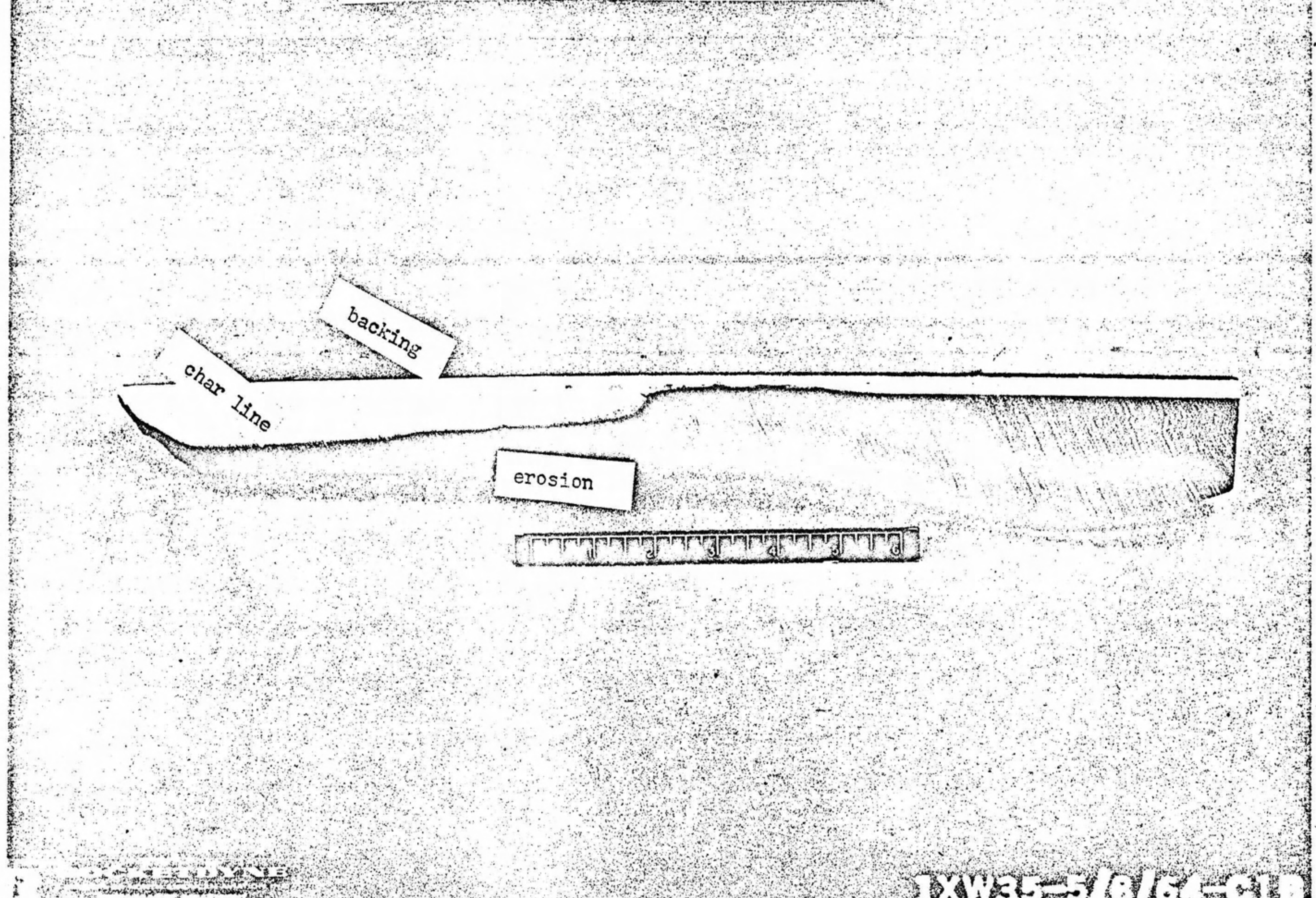
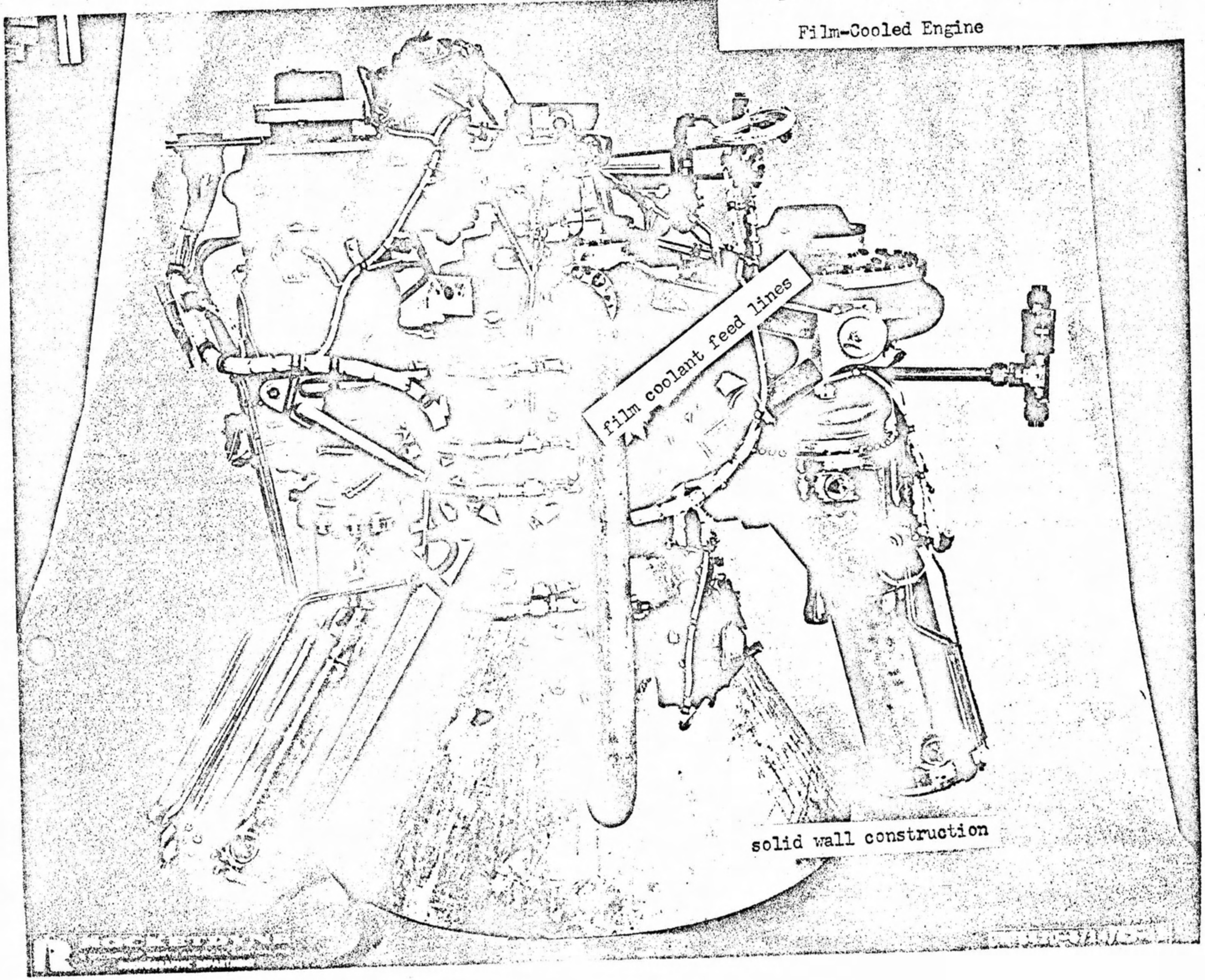


Figure 5 X-8 Experimental High Pressure

Film-Cooled Engine



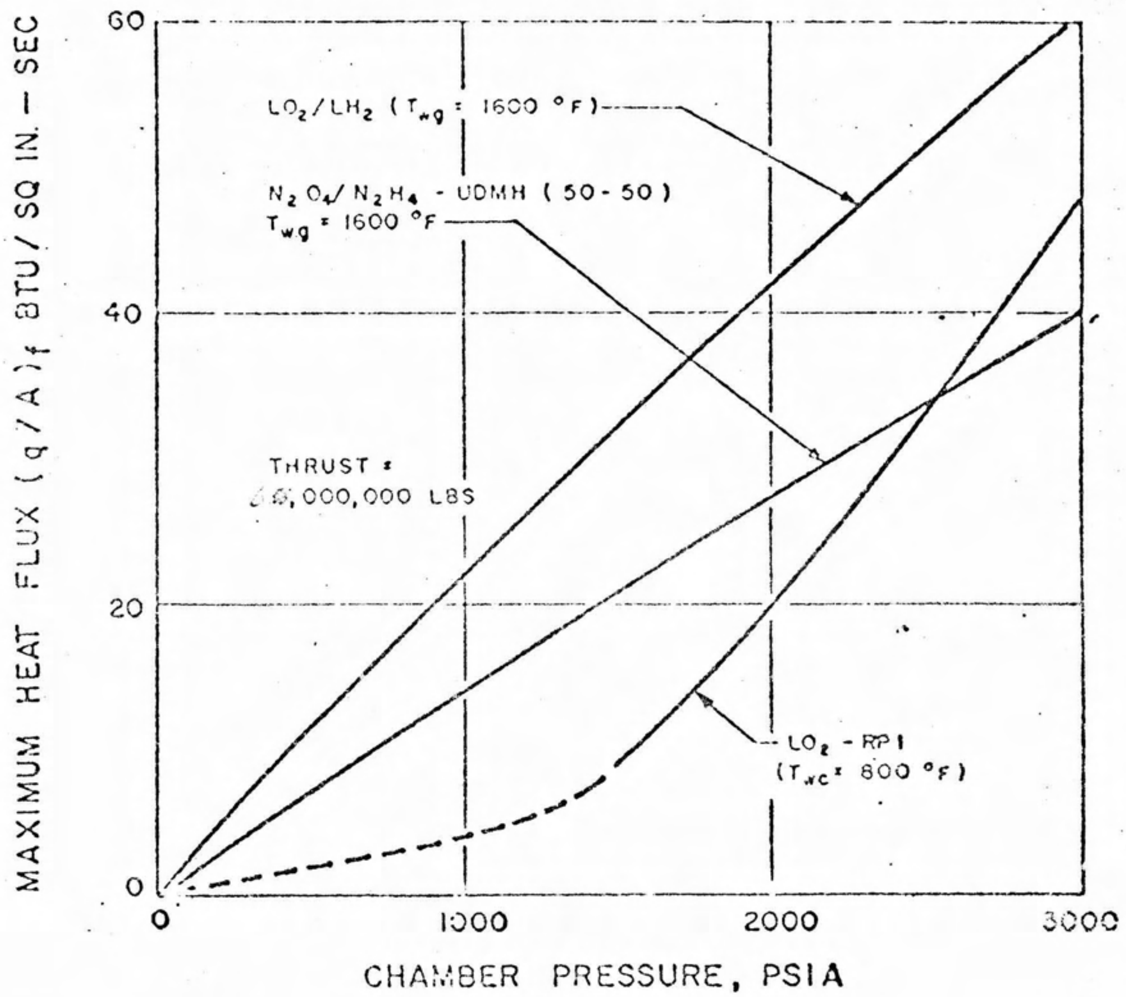


FIG.6 ~~FIG. 5~~- Maximum Heat Flux vs. Chamber Pressure for LO_2/RP-1, LO_2/LH_2 and N_2O_4/50-50 UDMH-N_2H_4