STAT

-0013
COPY 1 OF 3

Report X-394 Copy No. 1

FEASIBILITY STUDY

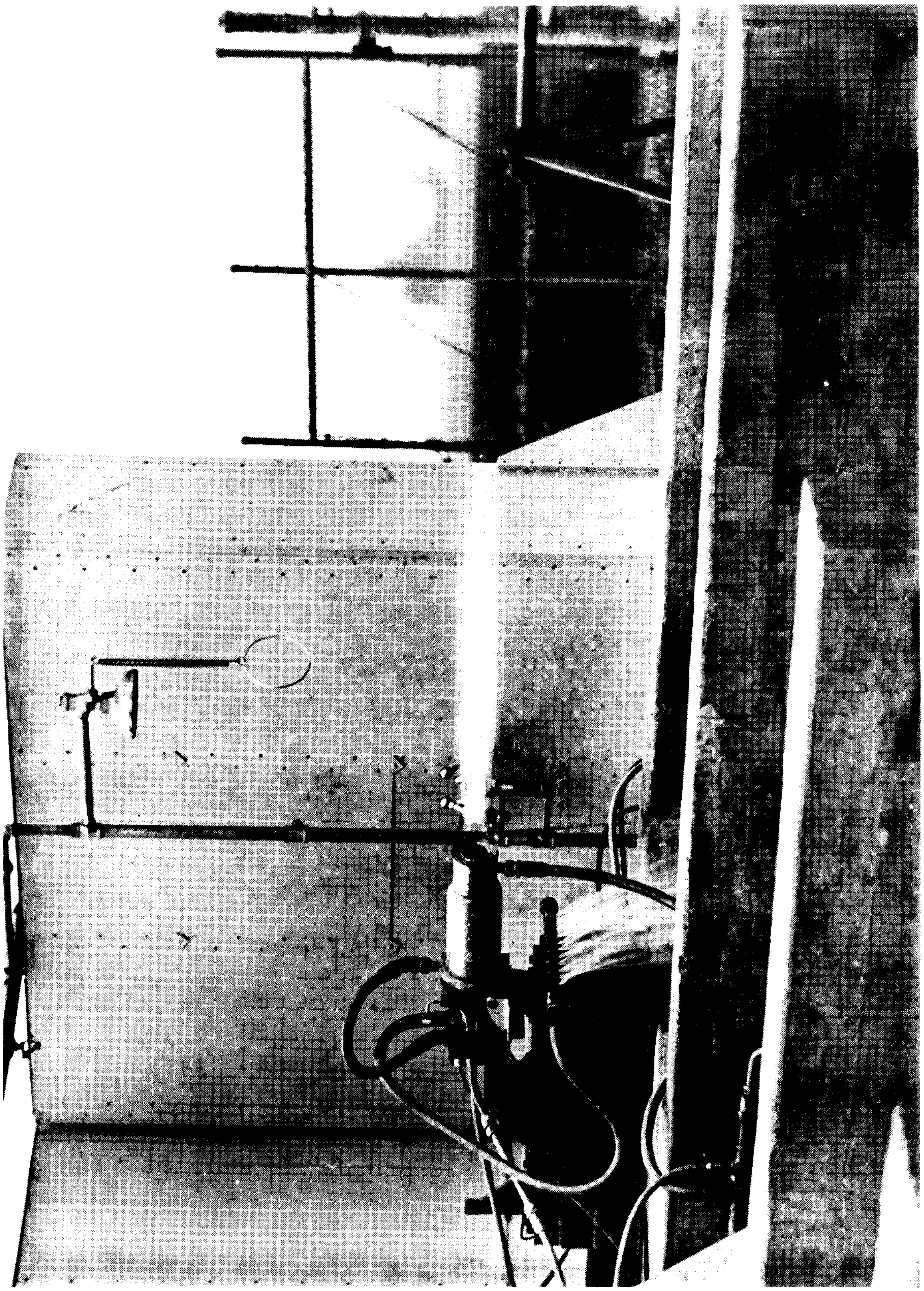
THROTTLEABLE LIQUID BIPROPELLANT ROCKET ENGINE

20 March 1959

STAT

STAT

HUGHES TOOL COMPANY--AIRCRAFT DIVISION
Culver City, California



Frontispiece. Static Firing of Model 315 Test Engine on RFNA and N_2H_4

TABLE OF CONTENTS

LIST OF ILLUSTRATIONS

SUMMARY

The feasibility of a throttleable liquid bipropellant rocket engine suitable for a long range manned vehicle was investigated. Results of the study indicate that such an engine is indeed within the present state of the art and could be developed to qualification status in a two-year period. The detection of such an engine by present ground based IR systems would be extremely difficult, if not impossible. The experimental phase of this program, in which several of the capabilities of this type of engine were to have been demonstrated was terminated after an accumulated five minutes of firing time on the test engine. This termination was necessitated by the limitations on program time and funds.

I. INTRODUCTION

From a vehicle standpoint, the choice of a propulsion system depends upon its performance, reliability, availability, and cost. It is the purpose of this study to establish the status of these factors for a liquid bipropellant rocket engine, so that a vehicle designed around such an engine can be compared with vehicles designed around other types of engines, specifically, airbreathers.

The vehicle comparison is beyond the scope of this study, but in general the basis for vehicle performance comparisons is range. In this regard, rocket-powered vehicles are at a disadvantage in travel through that part of the atmosphere that can support combustion. In the vehicle comparison at hand, however, the performance criterion is not solely range -- there is also an implied requirement for escape from detection. Here, the rocket-powered vehicle is at an advantage in comparison with an airbreather. This is so because the very parameter that a vehicle seeks to maximize to escape detection, namely, altitude, is that which imposes detection compromises in the case of an airbreathing engine.

Thus, the Mach 3 or faster airbreathers capable of operation at altitudes over 100,000 feet must resort to use of pyrophoric fuels to sustain combustion. The exhaust products of such fuels, bearing as they do solid constituents in relatively large amounts, can serve as rather efficient energy radiators in the IR spectrum. This condition is not necessarily so for a liquid rocket engine, inasmuch as its fuel constituent is not compromised by the necessity of combustion in thin air. Thus, rocket propellants can be tailored for minimum exhaust emissivity, and as a result a rocket-powered vehicle could enjoy a performance advantage over a vehicle powered by an airbreathing engine. This situation depends, of course, on the importance of the detection requirement, an evaluation that is, as previously stated, beyond the scope of this study.

II. PRELIMINARY DESIGN OF ENGINE

The following criteria were set up as basic requirements for the liquid rocket engine in this feasibility study:

A. Storable, high performance propellants

B. Thrust controllability throughout a large turn-down range

C. Long burning duration

D. Capability of development in two-year time period from present

E. Nonemissivity of exhaust products.

These requirements in toto define an engine that is, at present, nonexistent. However, as will be discussed, this nonexistence is at present the result of lack of stated need rather than of fundamental developmental problems. Individually, each of these requirements has been satisfied in various operational or developmental engines. Since none of these requirements are mutually exclusive, it is definitely within the state of the art to design a liquid rocket engine satisfying each and all of these requirements.

A. PROPELLANT SPECIFICATION

The storability requirement rules out cryogenic propellants and restricts the choice of liquid propellant oxidizers to two oxidizers: nitric acid and nitrogen tetroxide. In the performance study of the subject system that follows, nitric acid was selected over nitrogen tetroxide on the basis of greater development background and superior characteristics as a coolant. However, nitrogen tetroxide is becoming increasingly acceptable as a storable oxidizer, and actually is somewhat superior on a performance basis. The performance study thus reflects a certain degree of conservatism as regards range as a result of the use of nitric acid performance rather than nitrogen tetroxide (about 5 percent).

The choice of storable fuels is more extensive, but the restrictive requirements in this regard relate to hypergolicity and emissivity of exhaust products. Since it is considered that the advantage of hypergolicity in safe and reliable engine operation outweighs the hazards associated with tankage integrity, hypergolicity of propellants has been set out as a further requirement. This decision rules out hydrocarbon fuels and focuses attention on the hydrazine family. Considerable development background exists on both RFNA-N_2H_4 and RFNA-UDMH systems, and there is little to choose between them on a performance basis. However, considerations of exhaust emissivity may favor the choice of N_2H_4 over the more easily handled UDMH.

B. THRUST CONTROLLABILITY

This requirement is probably the one creating the most problems from an engineering standpoint. The developmental effort required for a controllable thrust rocket engine has in the past motivated system designers to accept compromise solutions involving multiple chambers, or preprogrammed boost-sustain thrust patterns.

For the subject application, however, in which thrust must be capable of matching drag throughout a relatively long period during which vehicle mass is constantly decreasing, the aforementioned design compromises are not acceptable. A truly modulating thrust control is now required. Fortunately such an engine presents no fundamental problems incapable of solution[1]. The main design compromise occurs in optimizing the feed system pressure drop-cooling flow relationship for the extremes of the thrust range. That is, the combustion chamber

1 The Naval Ordnance Test Station, Inyokern reports such an engine in development.

coolant passages must be small enough to create coolant velocities sufficient to maintain safe wall temperatures at the low thrust flow rates, and yet be large enough to allow acceptable pressure drops at the high thrust flow rates.

C. BURNING DURATION

The subject vehicle trajectory requires burning durations of approximately 25 minutes. This requirement, as well as that of (B) above, represents an extension to the present state of the art. It is an extension, however, only in that no requirement of this duration has heretofore existed. An increase in burning duration in itself represents no fundamental difficulty, inasmuch as present day regeneratively cooled thrust chambers operate in thermal equilibrium. Their burning duration is limited solely by their available propellant supply.

D. DEVELOPMENT PERIOD

Restricting the development period to two years defines the propellant choice more than it affects the hardware design. The hardware design itself appears to be within the state of the art; at least, no hardware compromises are envisaged because of the development period restriction. On the other hand, performance increases, which are theoretically available if laboratory type propellants were considered, are excluded by this requirement. Specifically, storable high density halogen oxidizers appear to show promise for this application -- promise that could be realized operationally in a few years if investigative and evaluative work were begun now. However, since propellant development normally requires long lead times, the performance characteristics of the engine have been limited to those available today.

E. EXHAUST EMISSIVITY

Operational systems in the past have used carbon bearing fuels, which, when used with either acid or LOX, give rise to long tail-plumes containing incandescent carbon particles. The emissivity of the incandescent carbon particles, being close to that of black body radiation, is 10,000 times greater at 6,000° K than the emissivity of the gaseous products of combustion. The inference in such a comparative figure is that there should be no carbon atoms (or at least no atoms producing solid particles in the exhaust) in the propellant system, in order to minize IR radiant energy. Thus, from an emissivity standpoint, propellants could be qualitatively rated in the following order (high intensity to low intensity):

(1) Propellants containing light metals

(2) Propellants containing chlorides

(3) Propellants containing carbon

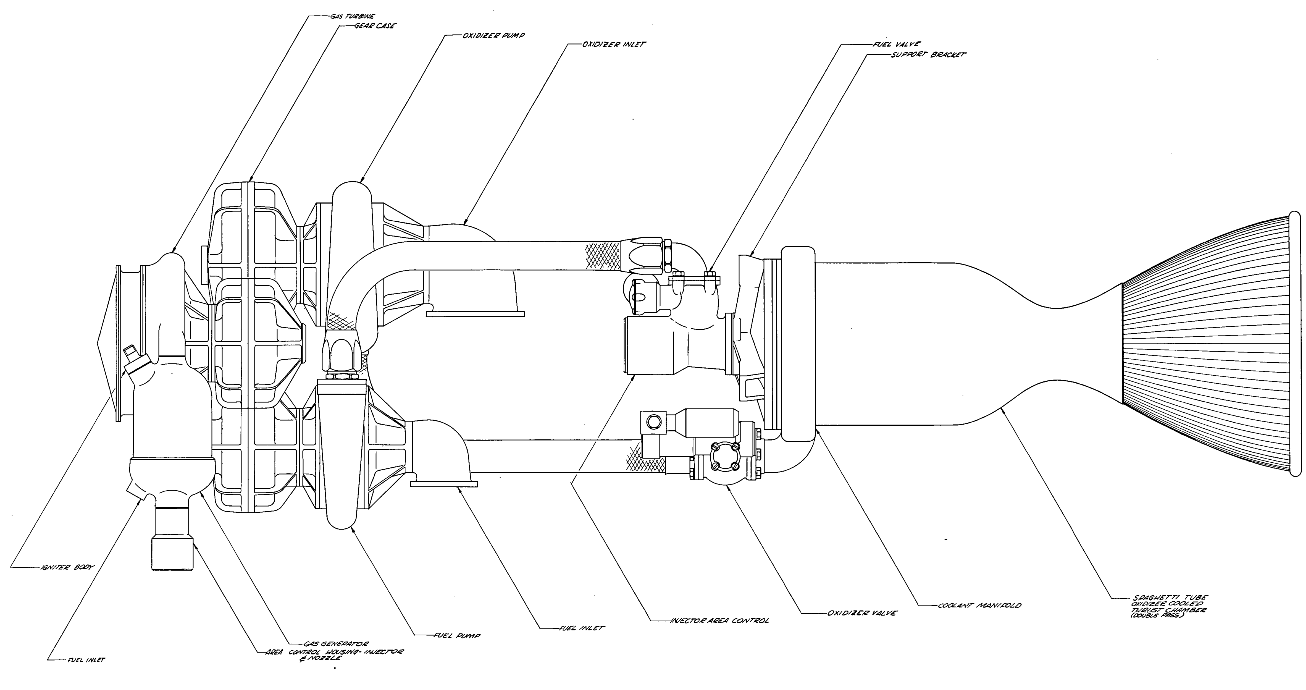
(4) Propellants yielding exhaust molecules containing only $N_2$and H_2O (RFNA-N_2H_4).

From the point of view of the possibility of detecting the exhaust emission, much depends on the location of the detector. Projects to achieve very sensitive IR detectors are now current for space environments. That is, the detectors must be in space as well as the IR source, and, if so, detection as far as 1,000 miles out is hoped for. This subject is dealt with in more detail in Section III. However, it can be stated here that for ground-located IR detectors the IR radiation from gaseous molecules in space is attenuated by the earth's atmosphere to such an extent that it would seem impossible to locate a high-altitude source of the subject magnitude.

F. PERFORMANCE OF PRELIMINARY DESIGN ENGINE

The foregoing has summarized the effect of the vehicle requirements on the powerplant. These requirements have been taken into consideration in the determination of design specifications for an applicable engine. The general geometry and essential features of such an engine are shown in Figure 1. Design specifications for two such engines whose turn-down ratios (4:1 and 2 1/2:1) span the area of interest are shown in Figure 2. It can be noted that their performance is essentially the same, although certainly the attainment of the lesser turn-down ratio would constitute a simpler development problem.

For evaluation of this engine's performance in a vehicle, Figure 3 is presented. Here, duration of powered flight is shown as a function of vehicle mass ratio and vehicle L/D for propellant Isp's of 275 and 300 seconds. For the sake of illustration, a range scale is superimposed on the time scale, in which the equivalence is based on a vehicle velocity of Mach 4. (In any particular vehicle there is, of course, a functional relationship between L/D and vehicle velocity, and therefore to preserve the generality of the graph a family of range scales should be envisaged.) It should be appreciated also that the vehicle range denoted by this range scale is solely that of the powered portion of the flight. The full flight profile would include the glide phase, which extends range considerably at the altitudes and flight speeds of interest.



DESIGN SPECIFICATIONS:

THRUST CHAMBER

THRUST AT DESIGN ALTITUDE	1250-5000 LBS.
CHAMBER PRESSURE	300-1200 PSIA.
CHAMBER COOLING	REGENERATIVE - RFNA.
PROPELLANTS	RFNA - UDMH.
MIXTURE RATIO	2.6:1
INJECTOR HEAD	VARIABLE AREA
NOZZLE THROAT AREA	3.0 SQ. IN.
NOZZLE AREA RATIO	25:1
CHARACTERISTIC LENGTH - L*	75 IN.
TYPE OF IGNITION	HYPERGOLIC PROPELLANTS
CONTROLLABILITY	THROTTLEABLE - 1250 TO 5000 LB.

TURBOPUMP

GAS GENERATOR PROPELLANT	ETHYLENE OXIDE
GAS GENERATOR CONTROL	VARIABLE AREA INJECTOR & NOZZLE
GAS FLOW	.06-.87 LB/SEC.
TURBINE INLET PRESSURE	500 PSIA.
TURBINE SPEED	12,500-25,000 RPM.
PUMP FLOW: RFNA	3.25 - 13.0 LB/SEC.
UDMH	1.25 - 5.0 LB/SEC.
PUMP DISCHARGE PRESSURE	400 - 1450 PSIA.
INLET PRESSURE	20 PSIA TO 50 PSIA.
POWER REQUIRED	10 - 145 HP.

PERFORMANCE

I_{sp} (ROCKET CHAMBER)	278 SECS.
I_{sp} (GAS GENERATOR)	150 SECS.
I_{sp} (EFFECTIVE)	265-274 SECS.

WEIGHT

THRUST CHAMBER	100 LBS.
TURBOPUMP	100 LBS.
TOTAL	200 LBS.

SPACE ENVELOPE

LENGTH	48 IN.
DIAMETER	18 IN.

PROPOSAL
LIQUID ROCKET ENGINE - R19
275-0517

Figure 1. Throttleable Liquid Rocket Engine, R19

Sanitized Copy Approved for Release 2011/01/11 : CIA-RDP89B00709R000400840007-6

DESIGN SPECIFICATIONS

	R19A (4:1 THRUST TURNDOWN)	R19B (2.5:1 THRUST TURNDOWN)
THRUST CHAMBER		
THRUST AT DESIGN ALTITUDE	1250 - 5000 LB	1000 - 2500 LB
CHAMBER PRESSURE	300 - 1200 PSIA	210 - 525 PSIA
THRUST CHAMBER COOLING	REGENERATIVE (RFNA)	REGENERATIVE (RFNA)
PROPELLANTS	RFNA - UDMH	RFNA - UDMH
MIXTURE RATIO	2.6:1	2.6:1
INJECTOR HEAD	VARIABLE AREA	VARIABLE AREA
NOZZLE THROAT AREA	3.0 SQ IN.	2.85 SQ IN.
NOZZLE AREA RATIO	25:1	25:1
CHARACTERISTIC LENGTH L*	75 IN.	75 IN.
TYPE OF IGNITION	HYPERGOLIC PROPELLANTS	HYPERGOLIC PROPELLANTS
CONTROLLABILITY	THROTTLEABLE, 1250 - 5000 LB	THROTTLEABLE, 1000 - 2500 LB
TURBOPUMP		
GAS GENERATOR CONTROL	VARIABLE AREA INJECTOR VARIABLE AREA NOZZLE	VARIABLE AREA INJECTOR VARIABLE AREA NOZZLE
GAS FLOW	0.06 - 0.87 LB/SEC	0.04 - 0.22 LB/SEC
TURBINE INLET PRESSURE	500 PSI	500 PSI
TURBINE RPM	12,500 - 25,000 RPM	12,500 - 18,700 RPM
PUMP FLOW		
RFNA	3.25 - 13.0 LB/SEC	2.6 - 6.5 LB/SEC
UDMH	1.25 - 5.0 LB/SEC	1.0 - 2.5 LB/SEC
PUMP DISCHARGE PRESSURE	400 - 1450 PSI	300 - 675 PSI
PUMP INLET PRESSURE	20 - 50 PSI	20 - 40 PSI
POWER REQUIRED	10 - 145 HP	6 - 34 HP
PERFORMANCE AT ALTITUDE		
Isp (ROCKET CHAMBER)	278 SEC	278 SEC
Isp (GAS GENERATOR)	150 SEC	150 SEC
Isp (EFFECTIVE)	265 - 274 SEC	271 - 275 SEC
WEIGHTS		
THRUST CHAMBER	100 LB	60 LB
TURBOPUMP	100 LB	60 LB
TOTAL ENGINE	200 LB	120 LB
SPACE ENVELOPE		
LENGTH	48 IN.	48 IN.
DIAMETER	18 IN.	18 IN.

Figure 2. Design Specifications, R19A and R19B

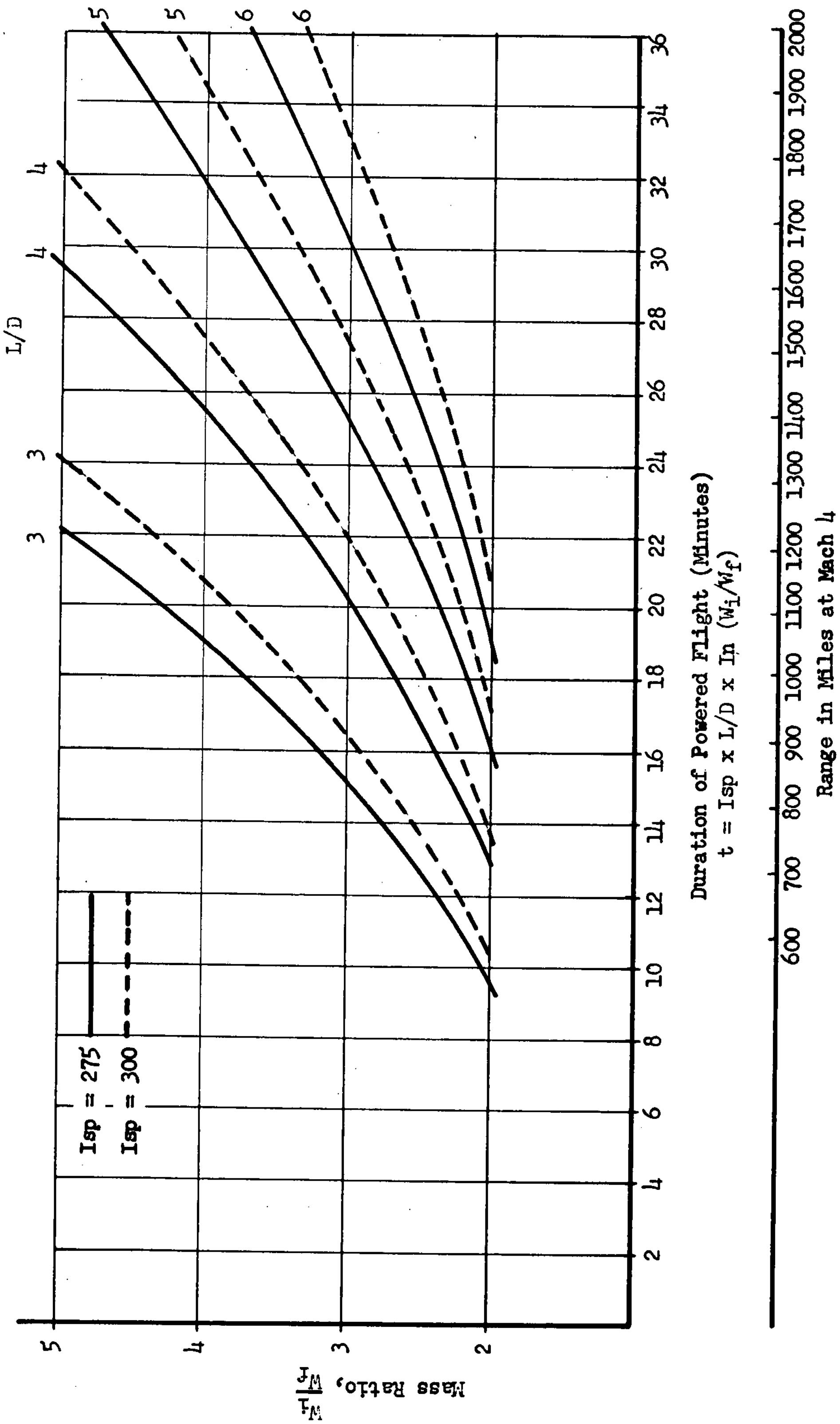


Figure 3. Vehicle Performance With R19 Rocket Engine

III. INFRARED DETECTION OF EXHAUST FLAMES

Exhaust flames containing incandescent carbon radiate with nearly black body emissivity. The hot exhaust gases containing associated radicals such as OH radiate considerably upon reassociation of these gases into their parent molecules. Some polar molecules such as CO_2 and H_2O radiate discreetly in the IR region. (The strong CO_2 and H_2O bands are at 4.25 microns and 3.0 microns respectively.) The polar molecules have lower emissivities than do carbon particles and, since they do not radiate in a continuum, their total intensity is less than that of a black body radiating at the same temperature.

The present state of the art shows the development of infrared detection devices that will seek out exhaust jets from a distance of several miles in the rarified atmosphere above the earth. At the present time, the most sensitive of these instruments can detect total intensities in the order of 10^{-11} to 10^{-12} watts/cm^2. However, infrared detection instrumentation research toward improving this range is now under way in several places. These research centers are working on outer atmosphere infrared-sensitive devices (selenium sulfide cells, and so forth) that will detect discreet radiation from CO_2 and H_2O at distances in excess of 100 miles.

The detection of discreet radiation through atmosphere containing water vapor and carbon dioxide is quite difficult, since these molecules in the air absorb this radiation and allow very little to pass through to the detector. However, the effect of pressure broadening of the discreet bands has helped in the problem of infrared detection in the atmosphere. An example might be given to show the radiation intensity of a flame at a temperature of 2000° F, one foot in diameter, radiating in a vacuum at a distance of 100 miles. The intensity would be approximately 10^{-12} watts. This approaches the maximum limits of present day sensitivity and would be considerably less in transmission through the atmosphere.

The infrared spectra may be generally considered to arise from the vibrational and rotational energies of a molecule. Vibrations in a molecule arise from a motion of the atoms along a line common to both in harmonic motion, or, in some cases, anharmonic oscillation. The vibrational energy, including its interaction with electronic energy, is found in the spectral region from 1 to 20 μ Rotational spectra arise from the rotation of the whole molecule about the center of gravity of the molecular structure. The energy levels of the rotational aspects of infrared spectra are weak and are found in regions beyond 20 μ and may extend and be detected as far as 600 μ, according to most recent work. For the purposes of infrared detection of exhaust flames it is only the vibrational energy levels that are important. For molecules possessing dipole moments absorption of energy may occur if the incident energy corresponds to the frequency of mechanical vibration. On the other hand, emission of radiant energy may occur if there is a transition in the quantum level from a higher to a lower plane. Absorption of energy will occur when there is excitation from a lower to a higher quantum level.

The following equations define the interrelationship of wave length (λ), wave number ($\tilde{\nu}$), frequency (ν), and speed of light (c):

$$\frac{1}{\lambda} = \qquad = \frac{\nu}{c}$$

The problem of employing linear units of frequency or linear units of wave langth has not been settled as yet and the above relationship provides a means of interconversion between units. As may be seen, the frequency divided by the speed of light will give the reciprocal centimeter or wave number. The range may cover 10,000 wave numbers (1 micron) to 10 wave numbers (1000 microns).

As it was previously stated, absorption of energy raises the molecule to an excited state and when the molecule returns to the ground state it re-emits the absorbed energy. The absorption or radiation of energy is characteristic of the atomic moieties undergoing vibration and will occur at definite wave lengths. However, there may be broadening of the spectrum of a characteristic vibration for gases by the so-called pressure broadening or collision broadening effect. In these cases there are statistical frequency perturbations caused by the presence of extraneous atoms, through the medium of van der Waal forces and phase perturbations produced by collisions. Schematically the problem is described by assuming a time during which there is a constant frequency of radiation with no perturbation and then assuming a time constant for perturbations creating phase changes. The effect of pressure broadening is of considerable importance in infrared detection of the exhaust of missiles. For CO_2 the 4.25 μ band is shifted to 4.1 μ to 4.5 μ. The effect of broadening apparently is sufficient for the radiation to get through atmospheric water vapor, which possesses various absorption bands at 2.5 μ, 3.0 μ, and 5.0 to 7.0 μ.

There is also a Doppler displacement of wave length that is characteristic of a source of radiation that moves with speeds of significance in comparison to the speed of light. This is a quadratic Doppler effect and the change in wave length with velocity of source may be described as follows:

$$\Delta\lambda = \lambda_o\left(-\frac{V}{c}\cos\theta + \frac{V^2}{2c^2}\right)$$

where: θ = angle of observation, V = velocity of source, c = speed of light.

For ordinary speeds, however, the Doppler shift is linear with velocity. The Doppler shift may be significant in detection of CO_2 in exhaust gases of a rocket engine.

Radiant heat may be considered as a stream of energy from a radiating body traveling with the same velocity as the speed of light, or 3×10^{10} cm·sec^{-1} (186,000 mi·sec^{-1}). The radiant energy is stopped only when it comes in contact with a solid, liquid, or gaseous body, being either absorbed or reflected. The radiant energy from a point source or a spherical body spreads out equally in all directions, following the simple geometrical law that states that the areas of similar solids increase as the square of their linear dimensions. Thus the intensity of radiation, or the amount of energy that passes each second through a plane of unit area perpendicular to the plane of flow, varies inversely as the square of the distance from the radiating source.

The ability to radiate heat is called emissive power and is a function of the temperature of a radiating body and its nature, especially that of the surface. It is mathematically equivalent to the total energy emitted per second per square centimeter of radiation body. The ratio between the emissive power of a given surface and that of a perfect radiator or black body is called the emissivity of that surface. The emissive power of a black body is 100 percent, or 1, so that the emissivity or the ratio of emissive power of any radiating surface to that of a black body radiation must necessarily be less than 1.

$$\varepsilon = \frac{E^1}{E} \quad \begin{array}{l} = \text{emissive power of body} \\ = \text{emissive power of black body} \end{array}$$

Stefan formulated a law which states that the total rate of radiation emitted by a unit area of a black body is proportional to the fourth power of its absolute temperature:

$$Q = \sigma T^4$$

σ = constant equal to 5.77×10^{-5} $\mathrm{erg \cdot sec^{-1}\ cm^{-2} \cdot degree^4}$

Q = $\mathrm{ergs \cdot sec^{-1} \cdot cm^{-2}}$ radiated

T = absolute temperature.

For bodies that are not perfect radiators the emissivity enters into the relationship that depends upon the nature of the radiating body and its surface, but for gases the emissivity is not even constant. Thus, the formula now becomes:

$$Q = \varepsilon \sigma T^4$$

where ε = emissivity.

For a hot gas that is in thermal equilibrium the emitted radiation at any given wave length is less than the emissive power of a black body at the same temperature. However, for exhaust flames of a rocket engine equilibrium conditions are not in evidence. Discontinuous radiation of exhaust flames of a rocket are expected to occur and the characteristic radiation spectrum of CO_2 will be evident. Although the emissivities of gases at laboratory conditions are optically thin and deviate considerably from black body radiation, rocket exhaust gases may approach the black body emission conditions.

The emissivity of a hot gas is a function of the wave length of the emitted radiation and therefore will have a series of values according to the wave length of emission. Another term is sometimes introduced, namely, the engineering emissivity, which is the total of the partial emissivities at the various wave lengths.

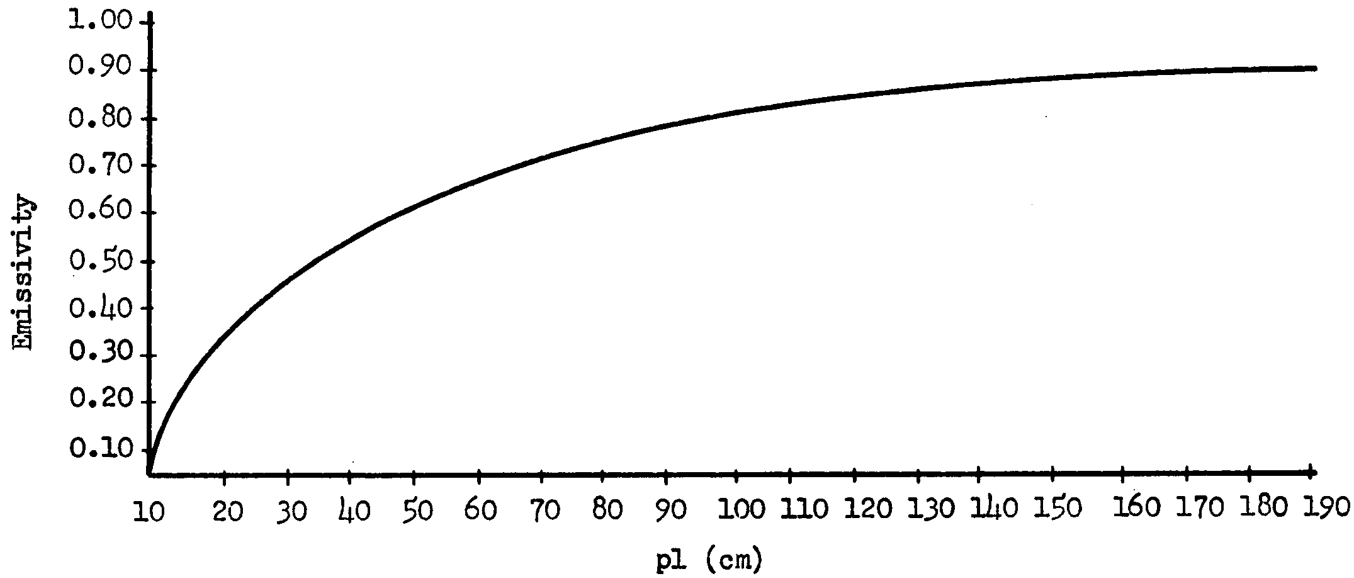
It must be recalled that it was stated that the emissivity of a gas is not constant. It depends upon the optical density of a gas in addition to the temperature. The optical density may be defined as the product of the partial pressure of the gas in atmospheres or centimeters of mercury and the geometric length expressed in linear measure such as cm or feet. Some data concerning emissivities of CO_2 are shown below as a function of optical density at a temperature of 600° K and a wave number 2349 cm^{-1}.

pl cm-atmos	pl ft-atmos	Emissivity Total
0.1	0.0033	0.019
0.5	0.033	0.040
1.0	0.033	0.040
5.0	0.164	0.057
15	0.492	0.062
50	1.64	0.063
100	3.28	0.065
200	6.56	0.067

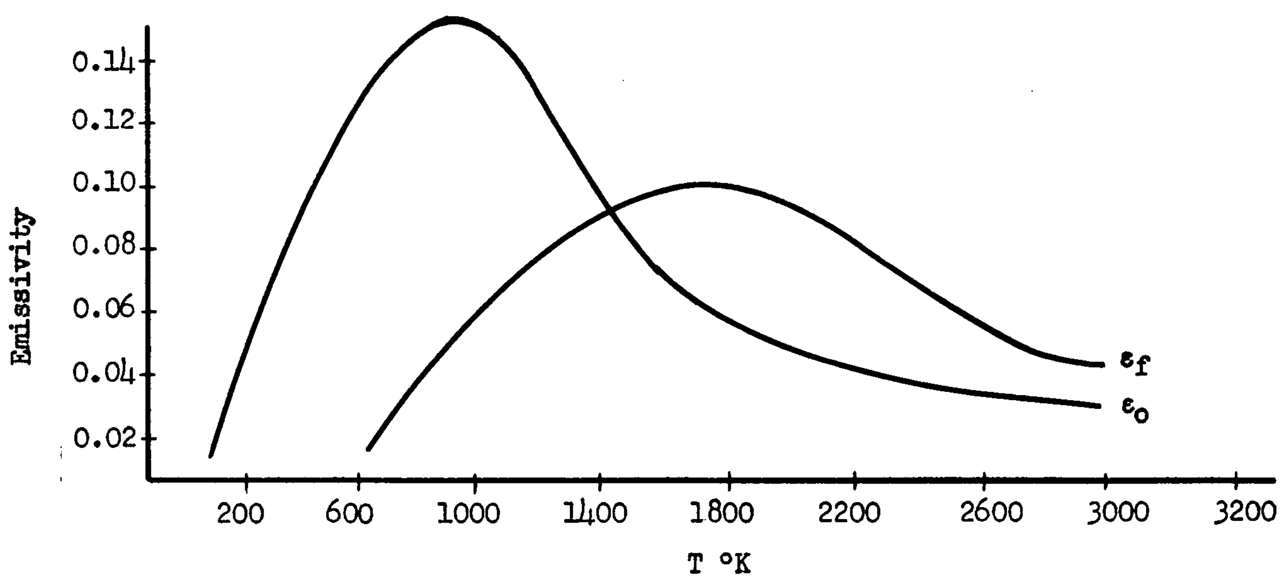
p = partial pressure

l = depth

Broadly speaking it may be said that the emissivity of a gas is decreased as the temperature or pressure is decreased. However, there is a limiting emissivity that is characteristic of a gaseous radiation at any temperature and optical density. For example, the limiting emissivity of CO_2 at 300° K is 0.4 and will increase very slowly even with high values of pl (optical density). The shape of the curve may be seen, on the following page, for emissivity values of H_2 at 12,000° K and P of 20 atmospheres.



The limiting emissivity of CO as a function of temperature may be seen in the following graph.



ε_o = emissivity of the first overtone

ε_f = emissivity of the first fundamental.

Emissivity data are also given by Lester Lees for air in equilibrium as a function of density and temperature. However, the data are based on samples of optically thin slabs.

$\varepsilon \cdot cm^{-1}$	Temperature ° K	Density (Relative to sea level air)
10^{-4} to 10^{-3}	4,000	0.10 to 1.0
10^{-4} to $10^{-2.5}$	5,000	0.04 to 1.0
10^{-4} to 10^{-2}	6,000	0.008 to 1.0
10^{-4} to $10^{-1.5}$	7,000	0.001 to 1.0
10^{-4} to $10^{-1.2}$	8,000	0.001 to 1.0
10^{-4} to 10^{-1}	12,000	0.001 to 0.11

In addition, the emissivity of atomic hydrogen is shown below as a function of temperature and pressure at $l = 50$ cm.

P atmos	8400° K	9200° K	11,300° K	12,000° K
10	0.014	0.037	0.26	0.48
20	0.030	0.071	0.35	0.66
40	0.063	0.14	0.57	0.83
70	0.11	0.24	0.74	0.92
100	0.15	0.32	0.82	0.96
150	0.22	0.44	0.89	
200	0.30	0.54		

To solve the problem of the total radiant energy emitted by the exhaust plume of a rocket engine, we may employ the formula (considering radiation to take place in a vacuum):

$$Q = \varepsilon\sigma T^4$$

ε = emissivity

σ = Stefan Boltzmann constant = 5.672×10^{-5} erg$\cdot$cm$^{-2}\cdot$deg$^4\cdot$sec^{-1}

T = absolute temperature

Q = ergs$\cdot$sec$^{-1}\cdot$cm^{-2}

We shall make an approximation of the emissivity of CO_2, which is higher than that of CO, following the limiting emissivity of CO_2 given by Penner at 300° K as 0.4. We shall next make an approximation of the size of the radiating sphere of hot exhaust gas as two feet in diameter. The hot gas sphere is considered as a stationary point source. Therefore, the formula is as follows considering the radiation from a sphere of one foot radius:

$$Q = 4\pi r^2 \varepsilon\sigma T^4$$

However, the radiation from the point source is now distributed at 100 miles away on the surface of the earth at an area of $4\pi R^2$. Therefore, we have

$$Q = \frac{4\pi r^2 \varepsilon\sigma T^4}{4\pi R^2}$$

where R = distance to the missile flame source from the surface of the earth

r = the radius of the emitting source.

All calculations are in the cgs system and conditions are stated as follows:

Flame plume: 2 feet in diameter or (2 x 30.48 cm)

Flame temperature: 2000° F or 820° K

Distance of flame: 100 miles or (1609.35 x 100) meters.

Thus:

$$Q = \frac{4\pi r^2 \varepsilon \sigma T^4}{4\pi R^2}$$

$$Q = \frac{r^2 \varepsilon \sigma T^4}{R^2}$$

$$Q = \frac{(30.48 \times 1)^2 (0.50)(5.67 \times 10^{-5})(820)^4}{(100 \times 1609.35 \times 100)^2} = 4.59 \times 10^{-5} \text{ ergs}\cdot\text{sec}^{-1}\cdot\text{cm}^{-2}$$

$$\frac{4.59 \times 10^{-5}}{10^7} = 4.59 \times 10^{-12} \text{ watts}\cdot\text{cm}^{-2}$$

(Note: One watt is equivalent to 10^7 ergs·sec^{-1}.)

This energy is detectable by present day devices.

The calculations have assumed a radiation in a vacuum and have ignored the significant absorption properties of the layers of water vapor in the atmosphere. In the infrared and red portions of the spectrum quantitatively the absorptions due to H_2O and O_2 are most evident. There is also the absorption due to CO_2. For example, the vibration spectrum of H_2O completely obscures the solar spectrum above 24 μ and almost completely above 16 μ.

As it was previously stated, the pressure broadening effect on CO_2 gas apparently enables the CO_2 radiation spectrum to get through the atmosphere. At the present time there is disagreement in the field of high altitude detection. Reliability of detection of infrared radiation at 100 miles altitude is theoretically possible, as seen by the calculated intensity. However, the feasibility of detection would be greatly enhanced on an air-based platform at a height of about 40,000 feet. From the ground the reliability of detection is poor and depends upon weather conditions, including the degree of water vapor saturation.

From purely theoretical considerations then, it is apparent that the CO_2 radiation spectra may get through to the surface of the earth, especially when the broadening effect is considered. The order of transmitted power from our theoretical radiation source was found to be 10^{-12} watts.cm^{-2}. This then must be correlated with the available detectors presently in use. To date tiny crystals that change in conductivity when excited by infrared radiation are known. The signal variables in a fixed bridge circuit are fed to servo movements that guide the missile to its target.

At present, the photoconductor materials available may be tabulated as follows:

Crystal	Wave Length (μ)	Temperature ° K	Minimum Detection Power watts.cm^{-2}
Lead sulfide	2 to 5	193	10^{-11} to 10^{-12}
Lead sulfide	2 to 4.8	Uncooled	10^{-10} to 10^{-11}
Lead telluride	2 to 7	90	10^{-10}
Lead selenide	2 to 8.5	90	10^{-9} to 10^{-10}
Indium antimonide	2 to 9.5	90	10^{-9} to $10^{-9.5}$
Germanium (specially treated)	2 to 11	Cooled	10^{-9}
Thermal detectors	2		10^{-8}

It may be observed that the crystal detectors do not have the infrared total wave length response characteristic of the thermal detectors; however, their sensitivity in narrow wave lengths is somewhat greater. It is also seen that greater sensitivity of the photoconducting crystals is increased by cooling. For cooling purposes, air-borne cryostats of low weight and size are available employing pressurized nitrogen, which cools upon expansion and is continually recirculated to compression from expansion.

Optical techniques are also available for collecting and focusing the radiation and are similar to those employed in the visible portion of the spectrum. However, the optical materials cannot be opaque to infrared radiation. Characteristic materials are tabulated below.

TRANSMISSION PROPERTIES OF MATERIALS TO BE USED AS PRISMS OR FILTERS

Material	Approximate Transmission In Percent	Wave Length (μ)	Thickness
Arsenious sulfide (As_2S_3)	80	0.5 to 10	2 mm
Silver Chloride (AgCl)	80	2.0 to 20	10 mm
Potassium Bromide (KBr)	90	0.5 to 27	10 mm
Sodium Chloride (NaCl)	90	0.4 to 15	10 mm
Linde Sapphire (Al_2O_3)	90	0.3 to 5.5	2.5 mm
Lithium fluoride (LiF)	90	0.3 to 5	10 mm
Calcium Fluoride (CaF_2)	90	0.3 to 9	10 mm
Thallium Bromide Iodide (KRS-5)	70	0.6 to 38	2 mm
Fused Silica (SiO_2)	90	0.2 to 2.0	10 mm
Optical Silicon (Si)	55	1.5 to 6.5	0.1 inch
Germanium (Ge)	50	2.0 to 16	4 mm

Some problems of detection arise in background radiation, especially that of the sun. Appropriate use of filters can block out undesirable radiation, allowing the CO_2 spectra to pass. The materials used for filters are similar to those employed for optical radiation concentration. The feasibility of employment of filters is immediately evident from the consideration of transmitting and absorption wave lengths of the materials tabulated.

The fact that the missile plume radiation may be considered as a point source against background radiation, which is diffuse and widespread, lends itself to design characteristic that uses this phenomenon to advantage. Essentially, alternate transmitting and opaque bars may be used in a scanning device. It is apparent that the point source will be modulated, whereas the diffuseness of the background radiation over considerable steradians will not be modulated.

The problem of detection of the theoretically possible radiation available from an altitude of 100 miles can next be considered on the basis of improvement of amplification devices and processing of the signal subsequent to crystal photoconduction. There are many infrared frequency power measurement devices based on theoretical equivalence of work and heat.

IV. EXPERIMENTAL ROCKET ENGINE STATIC FIRINGS

It was the intent in this phase of the program to demonstrate certain capabilities of liquid bipropellant rocket engines pertaining to the subject feasibility questions. The demonstrations were to include:

(1) Long burning duration, accumulation of 50 minutes firing time

(2) Start-stop-restart engine operation

(3) Performance of RFNA-N_2H_4 and RFNA-UDMH

(4) Comparative emissivity of exhaust flames of RFNA-N_2H_4 (carbon-free) and RFNA-UDMH (carbonaceous).

The rocket engine planned for use in these demonstrations was the Hughes Tool Company--Aircraft Division Model 315 water-cooled propellant evaluation engine, Figure 4. This engine design had originally been developed for evaluation of high energy monofuels in an Air Force program [Contract AF33(600)-36797], but since its ignition system uses a hypergolic hydrazine slug the hardware was readily adaptable to bipropellant operation. Figures 5 and 6 are schematics of the engine thrust stand and feed system used in the static firing test setup.

The emissivity measurements were obtained with a recording IR spectrometer (Beckman Model IR-2S). This instrument measures the distortion effect produced by the radiant energy source of interest on the energy reception profile of a standard light source. In the static firing test setup the standard light source was mounted near the radiometer detector and its beam directed by means of reflecting mirrors to traverse the exhaust flame laterally before entering the detector. Figure 7 is a photograph of this test setup.

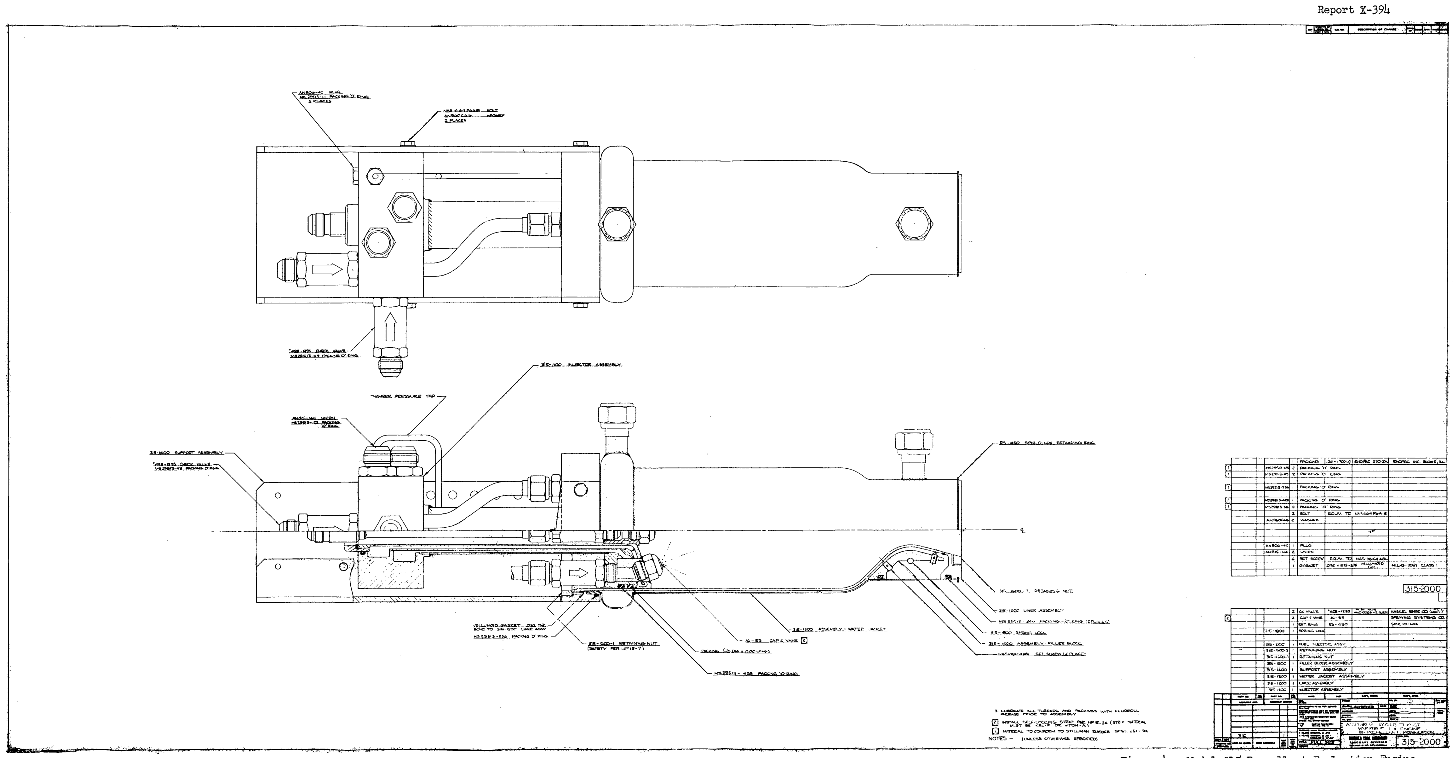


Figure 4. Model 315 Propellant Evaluation Engine

Sanitized Copy Approved for Release 2011/01/11 : CIA-RDP89B00709R000400840007-6

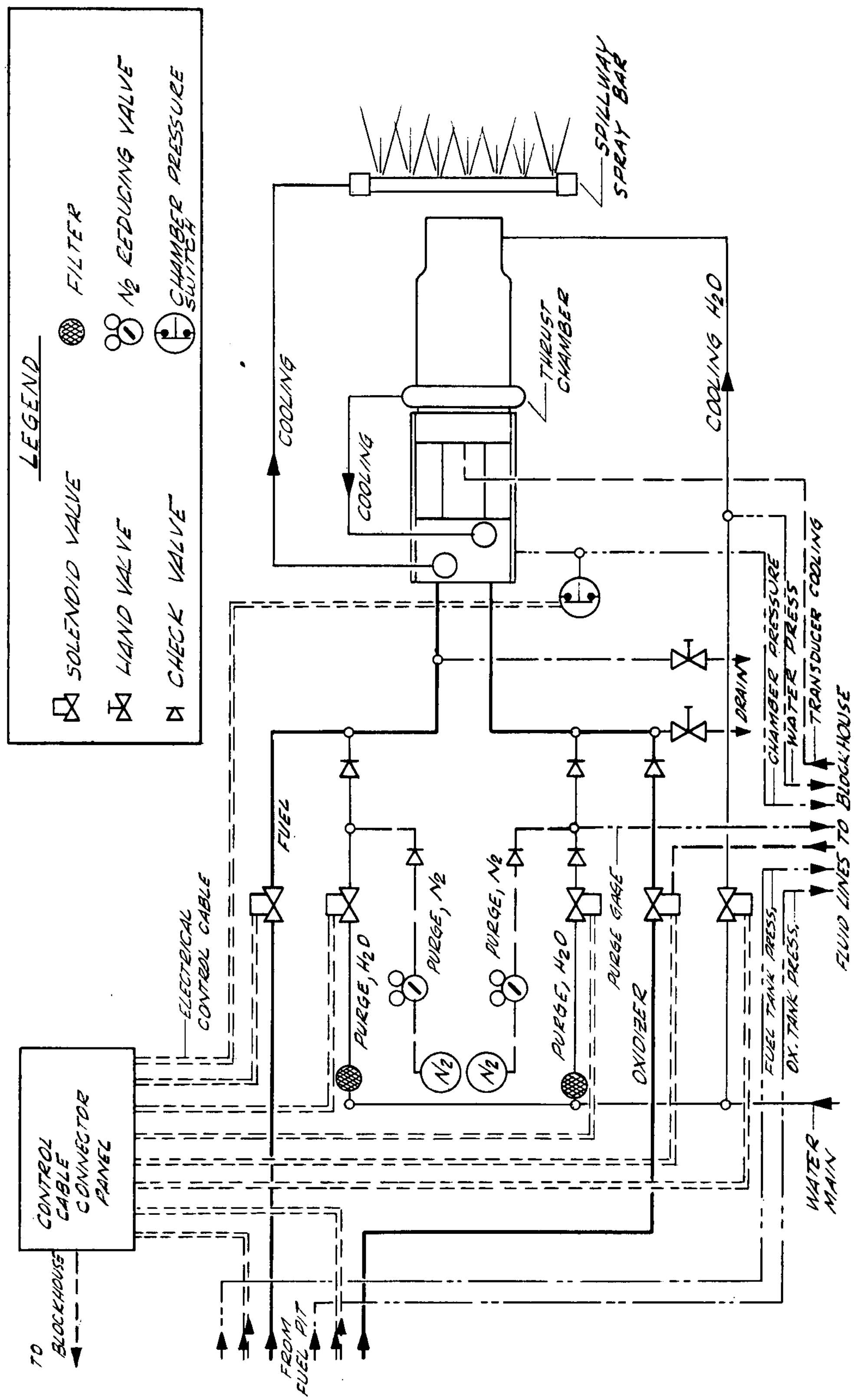


Figure 5. Model 315 Thrust Stand

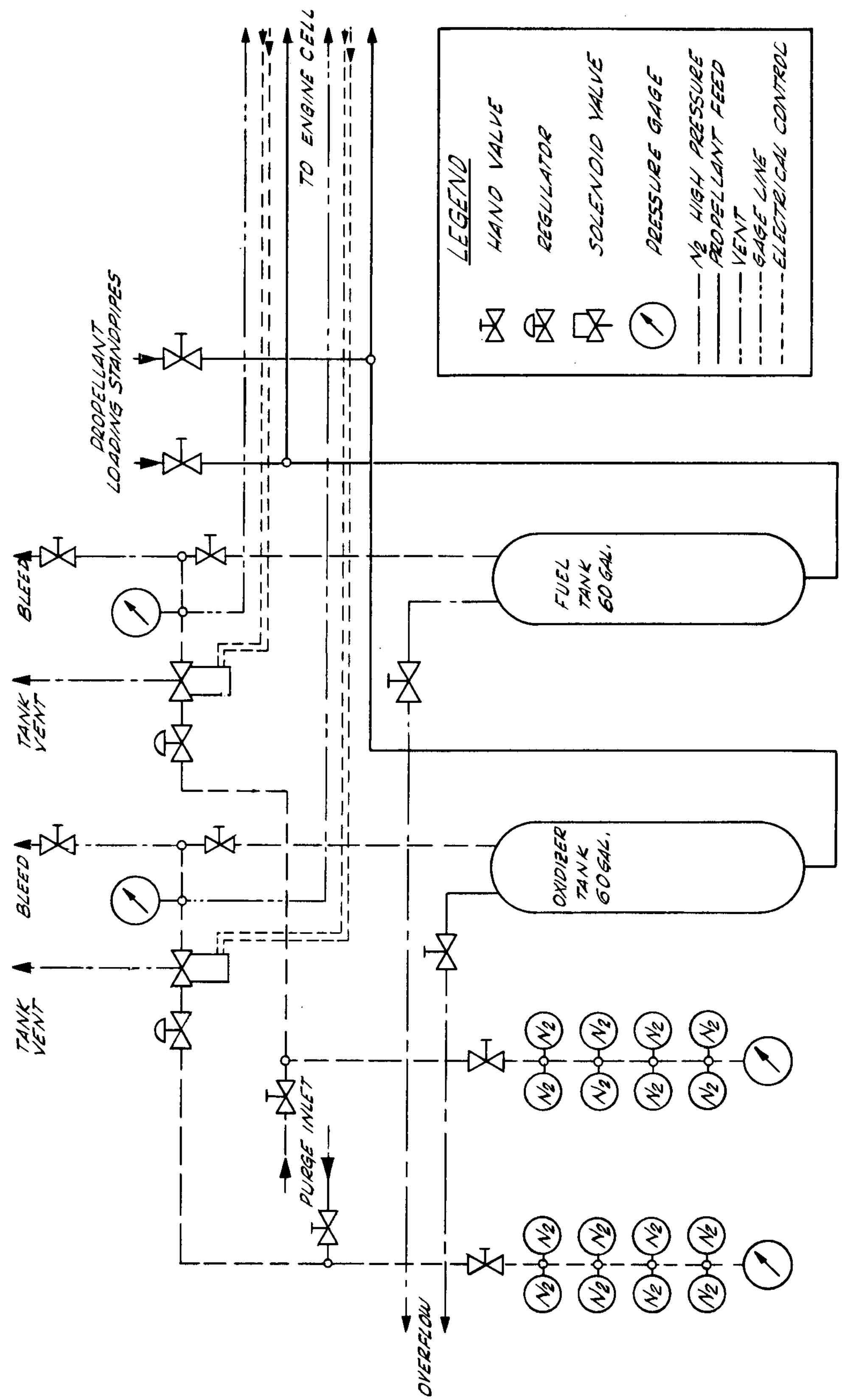


Figure 6. Model 315 Feed System

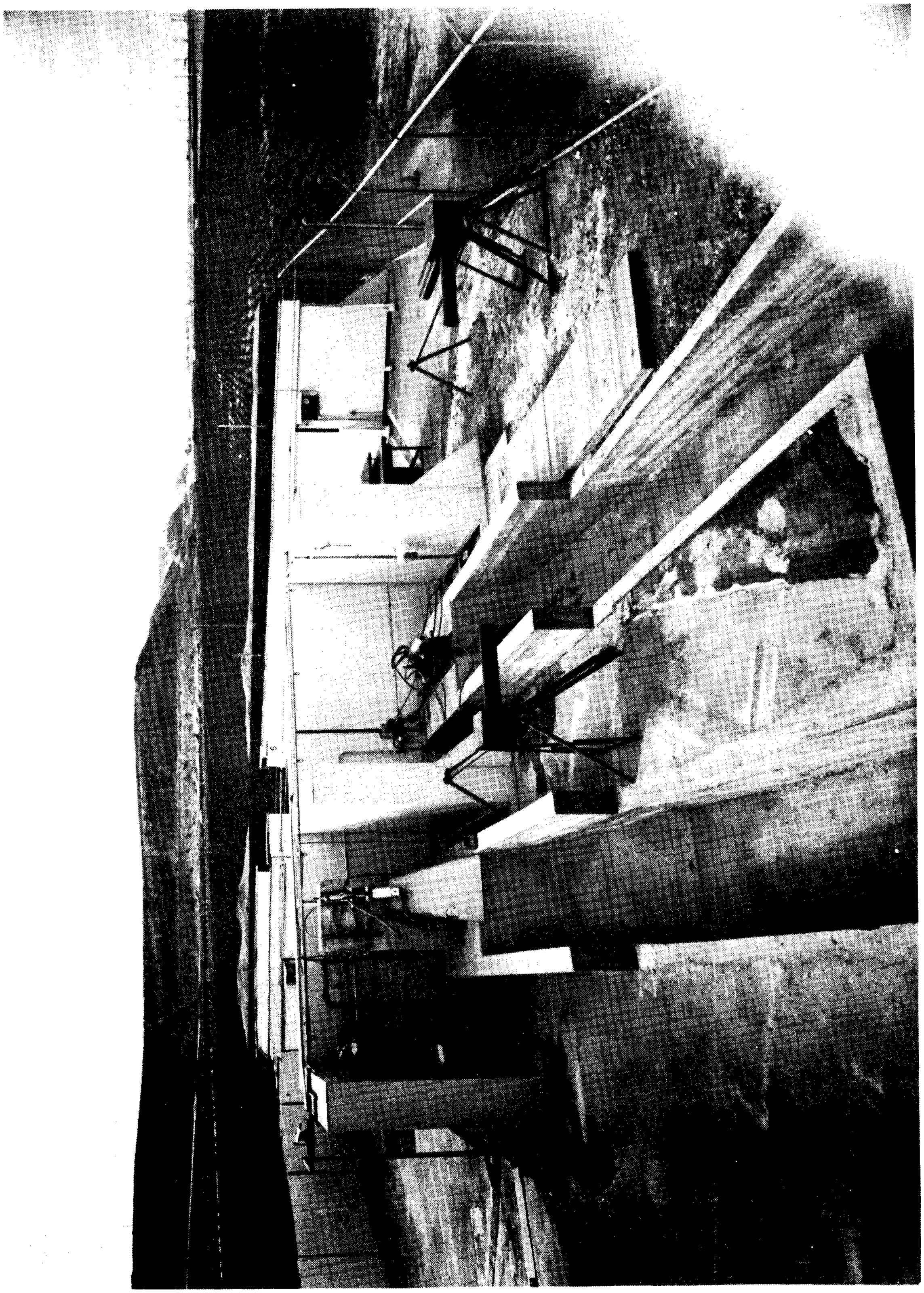


Figure 7. Rocket Exhaust Emissivity Test Setup

The static test firings, so far as they went, were quite successful in demonstrating smooth reproducible engine operation using RFNA-N_2H . The longest runs were of two minutes duration, sufficient to obtain a complete scan of the IR spectrum. A total of approximately five minutes operating time was accumulated on the engine using this propellant combination. The results of these runs are summarized in Figure 8.

Unfortunately, however, program limitations on time and funds necessitated a termination of the static firing tests prior to operation with the RFNA-UDMH combination, so that no comparison of the relative emissivities of the two exhausts is reportable at this time.

Item	Units	Run Number				
		315-5	315-6	315-7	315-8	315-9
Injector Manifold Pressure, P_i	PSIA	343	363	328	329	331
Combustion Chamber Pressure, P_c	PSIA	285	286	260	273	272
Mass Flow Rate, Oxidizer, $\dot{\omega}_O$	Lb/Sec	1.35	1.36	1.32	1.29	1.29
Mass Flow Rate, Fuel, $\dot{\omega}_f$	Lb/Sec	1.13	1.16	1.12	1.12	1.11
Mass Flow Rate, Bulk Propellant, $\dot{\omega}_t$	Lb/Sec	2.48	2.52	2.44	2.41	2.40
Mixture Ratio, O/F	--	1.19	1.17	1.18	1.15	1.16
Mass Flow Rate, Coolant, $\dot{\omega}_c$	Lb/Sec	5.63	5.33	5.57	6.23	6.25
Coolant Temperature Rise, ΔT_c	° F	22.5	24.5	25.2	30.0	32.4
Total Heat Transfer, $\dot{H}$	BTU/Sec	127	130	141	187	202
Duration, t	Sec	3.4	4.3	28.0	120.0	120.0
Thrust, F	Lb	507	507	493	522	515
Thrust Coefficient, C_F	--	1.24	1.24	1.32	1.33	1.32
Specific Impulse, I_{sp}	Sec	204	201	204	216	214
Characteristic Velocity, C*	Ft/Sec	5340	5270	5040	5260	5260

Figure 8. Summary of RFNA-N_2H_4 Static Test Firings in Model 315 Rocket Engine

Figure 9 shows qualitatively the effect of the interjection of the RFNA-N H exhaust flame into the view of the spectrometer. This figure is a superimposition of two scannings of the instrument, the dotted line representing the reception of the standard beam through the atmosphere, and the heavy line the reception of the standard beam through the atmosphere and the exhaust flame.

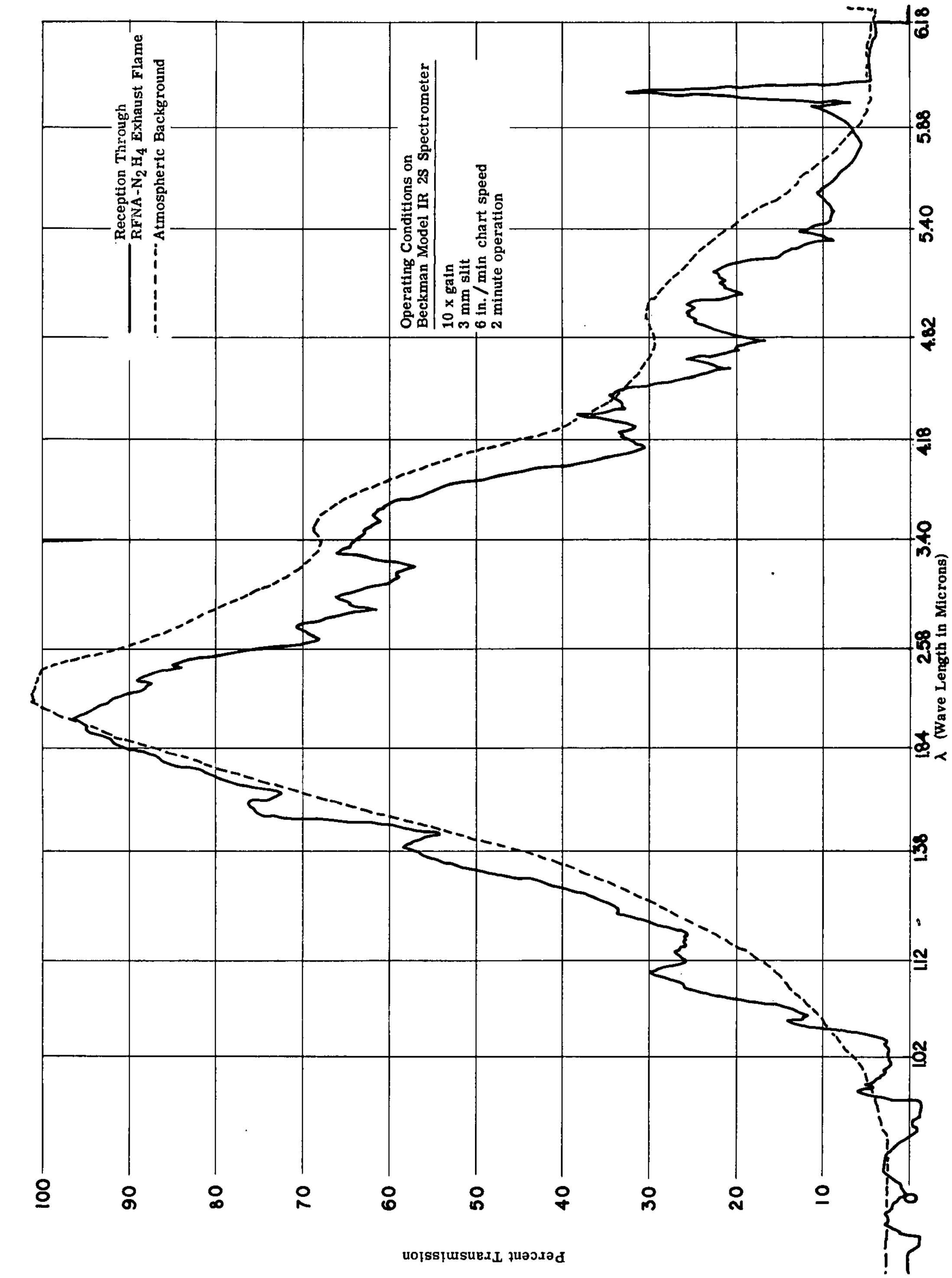


Figure 9. Effect of RFNA-N_2H_4 Exhaust Flame on Standard Source Adsorption Profile

BIBLIOGRAPHY

1. Lester Lees, Space Technology, Lecture 6A (February 1958), UCLA
2. S. S. Penner, Approximate Emissivity Calculation for Polyatomic Molecules I. CO_2, Jour. Applied Phys. 25, 660, May 1954
3. H. Aroeste and Wm. C. Benton, Emissivity of Hydrogen Atoms at High Temperatures, Jour. Applied Phys. 27, 117, February 1956
4. Instrument News, Perkin Elmer Corporation, 9 , 1 (Fall 1957)
5. E. A. Brande and F. C. Machod, Determination of Organic Structures by Physical Methods, Academic Press, Inc., New York, N. Y. (1955)
6. W. Brode, Chemical Spectroscopy, John Wiley and Sons, New York, N. Y. (1949)
7. G. P. Harrison, R. C. Lord, J. R. Loofbourow, Practical Spectroscopy, Prentice-Hall, Inc., Englewood Cliffs, N. J. (1955)
8. E. H. Kennard, Kinetic Theory of Gases, McGraw-Hill Book Co., New York, N. Y. (1938)
9. G. Herberg, Molecular Spectra and Molecular Structure, I Spectra of Diatomic Molecules, D. Van Nostrand Co., Inc., Princeton, N. J. (1957)
10. G. Herzberg, II Ingrared and Raman Spectra of Polyatomic Molecules (1956)
11. L. B. Loeb, Basic Processes of Gaseous Electronics, UCLA Press (1955)
12. P. Debye, The Dipole Moment and Chemical Structure, Blackie and Son, Ltd., London and Glasgow (1931)
13. W. Finkleburg, Conditions for Black Body Radiation of Gases, J. Optical Soc. of Am., February 1949, p. 185-186
14. S. S. Penner, Emissivity Calculations for Diatomic Gases, Jour. of Applied Mechanics, March 1951, p. 53-58
15. S. Mrozowski, On the $2\pi_{\mu} \rightarrow 2\pi_{g}$ Bands of CO_2, Part II, Phys. Reviews, September 1942, p. 270-279

16. H. J. Ramey, Project Squid, Heat Transfer Coefficients for Gasses--Effect of Temperature Level and Radiation, Purdue University, 1953, T.R.Pur-24-P

17. U. S. National Bureau of Standards, Energy Transfer in Hot Gases, NBS Circular 523 (November 1957)

18. D. V. Cogate and D. S. Kothari, Flow of Energy in Thermal Transpiration for a Bose-Einstein and a Fermi-Dirac Gas, Phys. Reviews, 349-358 (March 1942)

19. R. Gordon, Emissivity of Transparent Materials, J. Am. Cer. Soc., 39, August 1956, p. 278-287

20. Infrared Emission from High Frequency Discharge in CO_2, J. Ap. Physics, 28 , June 1957, p. 737-741

21. R. E. Peck, Fagan, Merlein, Heat Transfer Through Gases at Low Pressures, Transactions of the ASME, April 1951, p. 281-287

22. H. Latzko, Heat Transfer in a Turbulent Liquid or Gas Stream, TM 1068, October 1944

23. Jackson, Statler, Jinkoff, Heat Transfer to Bodies Traveling at High Speed In Upper Atmosphere

24. Robert MacFee, Aerojet Corporation, personal communication (1958)

25. Robert S. Neiswander, HAC, personal communication (1958)

26. Robert S. Neiswander, HAC, TM 481 (Secret)