#6-63-69

June 1963 Rocketdyne

Robert H. Boden
System Engineering

Propulsion, III-C-1
1

XI 14

INTRODUCTION

Propulsion is formally defined as the act of propelling a body. As an alternative definition for the systems oriented engineer or scientist, propulsion is the application of the basic sciences to generate, assess, and estimate the effectiveness of the methods to achieve controlled motion of a vehicle. There are many abstract implications to these definitions. One is that the motion is the result of human intervention, not free fall or drifting.

Propulsion, considering the alternative definition, has several points of view. The objective may be to establish the principles which underlie the design, application and operation of the devices, to organize information about the host of details which the practicing propulsion engineer must have at call, or to present those significant data which permit the organization of a feasible propulsion system for a vehicle. The objective here is to present system data.

A general analysis of the power plant system does not reduce to simple mathematical terms. An individual study needs to be made for each type of vehicle. The important vehicle performance parameters influenced by power plant performance are the range, X; the time duration of power plant operation, t_b; altitude, h; and the mission velocity or characteristic velocity of the vehicle, ΔV.

In general the vehicle performance increases with decrease in specific fuel consumption, drag coefficient, and vehicle velocity. The performance increases with increases in the propeller or transmission efficiency, η, and the ratio of the lift coefficient to drag coefficient, C_L/C_D. The wing loading $(\frac{W}{A_w})$ and specific power of the vehicle, P/W, may increase or decrease the performance depending upon what is wanted.

LAND VEHICLES.

For terrestrial vehicles the change in gross weights of the vehicle system are used to establish the factors which control the propulsion system performance (Ref. 1). The basic relationships are for the spatial rate of change of the gross weight,

$$\frac{dW}{dx} = \frac{w_f D}{375\eta} \text{ lb./mi.} \tag{1}$$

and the time rate of change of the gross weight,

$$\frac{dW}{dt} = \frac{w_f DV}{375} \text{ lb./hr.} \tag{2}$$

The parameters of the integrated equations are:

P = brake horsepower for a given speed V.

V = Speed, miles/hr.

T = Thrust = D = Drag, lb.

W = Gross weight = lift displacement

w_f = Specific fuel consumption, lbs. of fuel/bhp hr.

η = Propeller or transmission efficiency

X = Range, Miles

The drag of automotive and rail vehicles on level roads (Ref. 2,3) includes both rolling resistance, proportional to the vehicle gross weight, and air resistance, proportional to the (vehicle speed)^2. Grade resistance is a factor on hills but is not considered here as overall performance yardsticks for the power plant systems are desired. The total drag, D, is expressed in terms of the rolling resistance/unit weight, D_1, and the air resistance/unit frontal weight, D_2. The total resistance is $D = D_1 W + D_2 AV^2$, lb.

The rolling resistance per unit vehicle weight depends on road surface characteristics; material type and condition of tires, wheel bearing friction and the weight. For automotive vehicles the range of rolling friction is between .010 and .030 lb/lb. gross weight on smooth roads. The upper limit can approach 0.10-0.20 on rough roads.

The distance an automobile will travel under power is

$$x = \frac{375\eta}{D_1 W_f} \quad \frac{D_1 W_1 + D_2 AV^2}{D_1 W_2 + D_2 AV^2} \text{ miles} \tag{3}$$

When the weight of fuel carried is less than 5 percent of the total vehicle weight,

$$x = \frac{375\eta}{W_f} \quad \frac{W_f}{D_1 W_2 + D_2 AV^2} \text{ miles} \tag{4}$$

η is the efficiency of the power transmission system, W_f is the weight of fuel carried. W_1 is its gross weight including fuel, W_2 is the dry weight of the vehicle, D_1 is rolling resistance, lb/lb. gross weight, D_2 is air resistance, lb. in./ft.

The resistance factors for each axle of rail cars (Ref. 2,3) are of the form

$$D_i = 1.3 + \frac{29}{w_i} + B_i V + \frac{C_i A_i V^2}{w_i\, n_i}\text{, lb/ton} \qquad (5)$$

D_i = Resistance from each axle, lb/ton;

w_i = Average load per axle, ton;

n_i = No. of axles of i'th car;

B_i, C_i = Constants characteristic of equipment type.

Typical values of B_i and C_i are tabulated for standard types of equipment in Table I. Streamlined and light weight units require modification (Ref. 4).

TABLE I

Equipment	B_i	C_i
Locomotives	0.03	0.0024
Freight Cars	0.045	0.0005
Passenger Cars	0.03	0.0004
Motor Cars	0.09	0.0024

The drag, D, of a complete train is

$$D = \sum_{i}^{c} \left[(1.3 + B_i V)\, W_i n_i + 29\, n_i + V^2 A_i C_i\right] \qquad (6)$$

in which the summation is made over all of the cars of the train from the locomotive to the caboose.

The first two terms of the drag, D_1, are obtained empirically from coasting and dynamometer tests on standard equipment and are almost entirely from journal friction. These terms are a function of operating temperature. The values of the coefficients used in Table I are for normal temperatures. The third term is from flange friction and other factors proportional to speed. The last term is from air resistance. The coefficient, C_1, of V^2 in the last term should be modified to use the Totten coefficients when studying streamlined trains, for example, the diesel-electric streamliners such as the Santa Fe Super Chief.

The range, or distance the rail train will travel on tangent, level track under its own power is

$$x = \frac{375}{w_f\ (1.03 + 0.03V}\ \ln \frac{D_1}{D_2}\ \text{, miles,} \tag{7}$$

in which D_1 is the drag of the train fully loaded with fuel, and D_2 is the drag of the train after using the fuel to travel the distance, C miles. When the ratio of fuel used to gross weight of the train is less than 1:20, the range becomes

$$x = \frac{375}{w_f}\ \frac{w_f}{D_2}\ \text{, miles} \tag{8}$$

in which w_f is the weight in tons of fuel burned in traveling the distance X miles.

MARINE VEHICLES

A marine propulsion system is probably governed by more variables than any other vehicle propulsion system (Ref. 2, 5,6). No attempt is made to generalize here andif an accurate estimate of a ship's performance is required it should be made by methods given in the references. Only a few of the vehicle system variables are indicated here.

The performance of a completely submerged vessel is probably the most easily estimated. The drag is a function of the displacement, speed of the ship; a skin friction term and it's wave resistance. The range at constant speed thus being proportional to the $(\text{speed})^2$ and the logritham of the mass ratio, similar in many respects to the function describing the range of aircraft.

The performance of a surface ship is complicated by the wave resistance, trim, roll and pitch of the ship, and wind resistance. The power requirements of a displacement vessel can be roughly estimated from the Admiralty coefficient.

$$\text{Admiralty coefficient} = \frac{(\text{Displacement})^{2/3}(\text{Speed})^3}{\text{B H P}} \quad (9)$$

This coefficient is a function of the design speed, proportions of the ship, speed-length ratios, etc. It exhibits a consistency in value, ranging from 370-380 for single screw vessels, and from 360-390 for twin screw ships. Reliability is a prime factor in a marine system, as in aircraft. Reduction of weight per horsepower is accomplished by increased power plant speeds but is limited by lowered propeller efficiencies and increased power requirements, fuel consumption and dynamic effects which may affect other gains.

The performances of hydrofoil and vessels with planing hulls are specialized developments introducing more troublesome variables than displacement vessels and are not considered further.

In summarizing the marine systems, the important power plant system variables are the shaft, or brake, horsepower, specific power, $\left(\frac{B\ H\ P}{\text{displacement}}\right)$, overall weight, mass ratio and reliability.

AIRBORNE VEHICLES

Aircraft have three major performance criteria to satisfy (1) the rate of climb, R_c; (2) the absolute ceiling H and (3) the range, X. The major power plant parameters influencing these performance parameters are the net power, P; the specific power of the vehicle, $\frac{\eta P}{W}$, and the mass ratio, W_1/W_2. The propeller efficiency is a function of the parameter, $V/\eta D$, in which V is the flight speed, η is the propeller efficiency and D is propeller diameter (Ref. 1,7)

The rate of climb of an aircraft at sea level depends upon the reserve horsepower available.

$$R_c = \frac{\text{Reserve HP} \times 33{,}000}{W}\ \text{ft/min.} \qquad (10)$$

in which

Reserve HP = Thrust HP - Drag Power Losses

In terms of the equivalent monoplane span, b, the flat plate frontal area of the aircraft and the specific power P/W.

$$R_c = \frac{33,000\eta P}{W} - \frac{11,000W}{b^2 V} - \frac{0.288AV^3}{W} \text{ ft/min.} \qquad (11)$$

Specific power of the vehicle is of prime importance here. The effect of high speed is not great as the lift-drag ratio increases with speed and tends to reduce the effect of the last term.

The maximum altitude, or ceiling of an aircraft is estimated from the computed density ratio. Then the altitude is determined from a standard atmospheric density table (Ref. 8).

$$\frac{\rho_0}{\rho} = 358\, C_l \frac{A}{W} \left(\frac{L}{D}\right)^2 \frac{\eta' P'}{W} \qquad (12)$$

ρ_0 = Sea Level Density

ρ = Altitude Density

$\eta' P'$ = Thrust HP available at altitude

$$\frac{\rho_0}{\rho} = 358\, C_L \frac{A}{W} \left(\frac{L}{D}\right)^2 \left(\frac{\eta' P'}{W}\right) \qquad (13)$$

ρ_0 = Sea Level Density

ρ = Altitude Density

$\eta' P'$ = Thrust HP Available at Altitude

This ratio should be as large as possible for altitude performance and is achieved by supercharging internal combustion engines or high performance compressors in gas turbines.

The range of propeller driven aircraft can be estimated from Breguet's formula (Ref. 1,9).

$$X = 375 \frac{\eta}{w_f} \frac{(C_L)}{(C_D)} \ln \frac{W_1}{W_2} \text{ miles} \tag{14}$$

in which η is the efficiency of the propeller, C_L is the lift coefficient and C_D is the drag coefficient. The propeller efficiency, (Ref. 10), η is a function of the parameter, V/nD. V is the air speed, n the propeller speed of rotation and D the propeller diameter. For a given flight speed the product of the propeller efficiency and the lift drag ratio is inversely proportional to the specific power of the aircraft. High maneuverability requires high specific power while good range performance requires low specific power as long as the propeller efficiency remains high.

The helicopter obeys the same relationships for rate of climb, ceiling and range. However, the efficiency is no longer that of the propeller alone, but is the product of propeller efficiency and the transmission efficiency. Furthermore, typical lift-drag ratios of helicopters vary from 3-10, while the same ratio varies from 12-22 for contemporary winged aircraft. The net result is that the performance of the helicopter is substantially less that that of conventional aircraft and a high performance power plant system contributes toward maintaining the performance of this type aircraft.

The rate of climb of a jet aircraft depends upon the thrust-to-weight ratio of the aircraft, its air speed and the lift to drag ratio.

$$R_c = 1.467 \left(\frac{F}{W} - \frac{C_D}{C_L} \right) V \quad \text{ft/min.} \tag{15}$$

The density ratio from which the ceiling of jet aircraft can be obtained from standard atmosphere tables is

$$\frac{\rho_c}{\rho} = \frac{C_L}{393} \left(\frac{A}{W}\right)\left(\frac{C_L}{C_D}\right)^2 \left(\frac{FV}{W}\right)^2 \tag{16}$$

in which A is the wing area, and V is the maximum speed associated with the altitude lift coefficient and lift-drag ratio for the airplane.

The range of the jet airplane can be estimated from an approach similar to that used by Breguet. However, the specific fuel consumption is the ratio of the fuel rate per unit thrust.

$$X = \frac{V}{w_f'} \left(\frac{C_L}{C_D}\right) \ln \frac{W_1}{W_2} \quad \text{miles} \tag{17}$$

$$w_f' = \frac{\text{lb. fuel}}{\text{(lb. of thrust) hr.}}$$

This ratio should be as large as possible for altitude performance and is achieved by supercharging internal combustion engines or high performance compressors in gas turbines.

The performance parameters for range climb and ceiling of rocket powered, winged aircraft are precisely the same as those for jet powered vehicles (Equations 15, 16, 17). However, the specific fuel consumption, w_f', becomes the specific propellant consumption, the total propellant flow, lb. of fuel and oxidizer, per lb. thrust per hour. The specific propellant consumption expressed in the units of lb. and hours, is inversely proportional to the specific impulse, I_s, of the power plant. In the units of the mile, pound and hour, the specific propellant consumption is

$$w_f' = \frac{\dot{W}_o + \dot{W}_f}{F} = \frac{3600}{I_s} \text{ lb./lb.hr.} \qquad (18)$$

the specific impulse being expressed in units of pounds and seconds. Frequently the air specific impulse is used in assessing the performance of jet engines. This is directly related to the specific impulse based upon total propellant consumption by the relation

$$I_{sa} = I_s \left(1 + \frac{\dot{W}_f}{\dot{W}_o}\right) = \frac{3600}{w_f} \left(1 + \frac{W_f}{W_o}\right) \text{ sec.} \qquad (19)$$

in which the ratio $\dot{W}_f/\dot{W}_o$ is the fuel-air ratio of the jet engine. The air specific impulse can be expressed in a similar way.

Other performance parameters of aircraft which play a significant role are the cruising speed which is established by the most efficient operation of the power plant and propeller, the high speed which varies as the cube root of the specific power of the vehicle, the speed range which also varies as the cube root of the specific power and the endurance which is the ratio of the range, X, to the average flight speed, V.

SPACE VEHICLES

Rocket powered vehicles under steady motion obey the same basic laws of motion as do submersibles and aircraft (Ref. 11). The external forces acting are the rocket thrust, the aerodynamic and control forces, gravitational forces, centrifugal forces and Coriolis forces. These forces vary greatly in magnitude, depending upon the ambient environmental conditions. The rocket vehicle inherently has very high terminal and cruise speeds whether it is an aircraft, missile, projectile, space ship or satellite. During the period of operation, or burning time, of the power plant, the motion is dynamic. Since many variables including control forces, stability, lift coefficient, drag coefficient, altitude, flight angle, burning time, thrust level and the environment, all of which are usually different for each vehicle, no general solution can be given.

The velocity and altitude reached by a vertically ascending rocket powered vehicles are obtained from the basic equations of motion. These parameters indicate the significant performance factors to be looked for in rocket vehicles operating in the neighborhood of the Earth. The velocity of the vehicle at the end of burning, or cut-off velocity, V_c, is

$$V_c = g\, I_s \ln \frac{W_1}{W_2} - g\, t_c - \frac{B_1 C_o A}{W_1} + V_o \qquad (20)$$

g = acceleration of gravity, 32.2 ft./sec.^{2}

t_c = time of end of burning, sec.

A = equivalent flat plate cross section of vehicle, ft.^{2}

V_o = initial velocity, ft./sec.

$$B_1 = \int_0^{t_c} \frac{\frac{1}{2}\, r^2}{1 - \frac{W\, t}{W_1\, t_c}}\, dt$$

I_s = specific impulse, sec.

$W_p = W_o + W_f$ = total propellant burned, lb.

In free space where the gravitational losses are small and there is no atmosphere, the negative terms vanish, and an important performance parameter is generated,

$$\Delta V = V_c - V_o = g I_s \ln \frac{\dot{W}}{W_2} , \qquad (21)$$

which is defined as the characteristic velocity.

The height at end of burning is

$$h_c = g t_c I_s \left(1 - \frac{W_p}{W_2} \ln \frac{W_1}{W_2}\right) - \tfrac{1}{2} g t_c^2 \qquad (22)$$

$$+ V_o t_c + h_o - \frac{B_2 C_o A}{W_1}$$

$W_p = W_o + W_t$ = total wt. of propellants, lb.

The maximum altitude reached is

$$k = h_c + \frac{V_c^2}{2g} \text{, ft.} \qquad (23)$$

The kinetic energy of the vehicle at the end of burning is converted into potential energy, i.e., the vehicle coasts to its ceiling. The rocket power plant must generate sufficient thrust to make the initial thrust-to-weight ratio of the vehicle greater than unity. Operation of the power plant in which the time of burning is small in comparison to the time of flight to reach the desired range is defined as ballistic operation.

Space flight of a vehicle requires that the vehicle expend energy and undergo large changes in velocity to accomplish a mission. The energies and velocity changes are direct functions of the time of flight, thrust-to-weight ratio of the vehicle, drag losses, etc. The characteristic velocities required for various missions are summarized in Table II. These are estimated from mission analyses (Ref. 12,13,14) for ballistic rockets and low thrust rocket engines such as an ion rocket which can generate a thrust-to-weight ratio less than 10^{-4} from power supply systems which can be built within the limitations of current state of the art. The larger characteristic velocities required by the low thrust-to-weight-ratio vehicles result from gravity losses in flight. < TABLE II

The performance regimes in which different vehicles work is quite different. The specific jet power, specific impulse and the thrust-to-weight ratio of the vehicles are functionally related,

$$\frac{P}{W} = \frac{0.0292\ G\ I_s}{} \quad \text{hp./ lb.} \tag{24}$$

in which G is the thrust-to-weight ratio of the vehicle.

TABLE II. APPROXIMATE MISSION VELOCITY.

TYPICAL INTERPLANETARY MISSIONS.

MISSION	MISSION VELOCITY FT/SEC	
	BALLISTIC VEHICLE	LOW THRUST VEHICLE
EARTH ESCAPE	37,000	25,000 - 27,000 *
SATELLITE ORBIT AROUND EARTH	26,000 - 35,000	NOT APPLICABLE *
EARTH TO MOON (NO RETURN)	35,000 - 37,000	25,000 - 36,000 *
EARTH TO MARS		
NO RECOVERY	41,000 - 45,000	80,000 - 100,000 *
RETURN	53,000 - 65,000	
EARTH TO VENUS		
NO RECOVERY	43,000 - 44,000	80,000 - 100,000 *
RETURN	67,000 - 87,000	
EARTH TO JUPITER	80,000 - 90,000	

* FROM SATELLITE ORBIT AROUND THE EARTH

The range of variation for the chemical rocket, aircraft and the low thrust-to-weight ratio electrical propulsion systems are summarized in Fig. 1. The specific jet power for modern aircraft varies from 0.05 lb./hp. of propeller driven, internal combustion engined vehicles, to little more than 0.10 lb./hp. for high performance jet aircraft. The specific impulse of these engines seldom exceeds 100 sec.

Booster rocket engines operate in the thrust-to-weight ratio regime above unity in order to lift the vehicle from the ground. The terminal thrust-to-weight ratio near end of burning can exceed 10:1 in light, low payload vehicles.

Electrically powered vehicles are limited by current state of the art to maximum specific powers in the neighborhood of 10 jet hp./lb.

As the specific impulse increases the thrust-to-weight ratios must steadily decrease. This enforced low thrust-to-weight ratio results in an optimum specific impulse in order to obtain maximum payload for each mission. A typical estimate for a lunar vehicle using a low-thrust power plant is shown in Fig. 2.

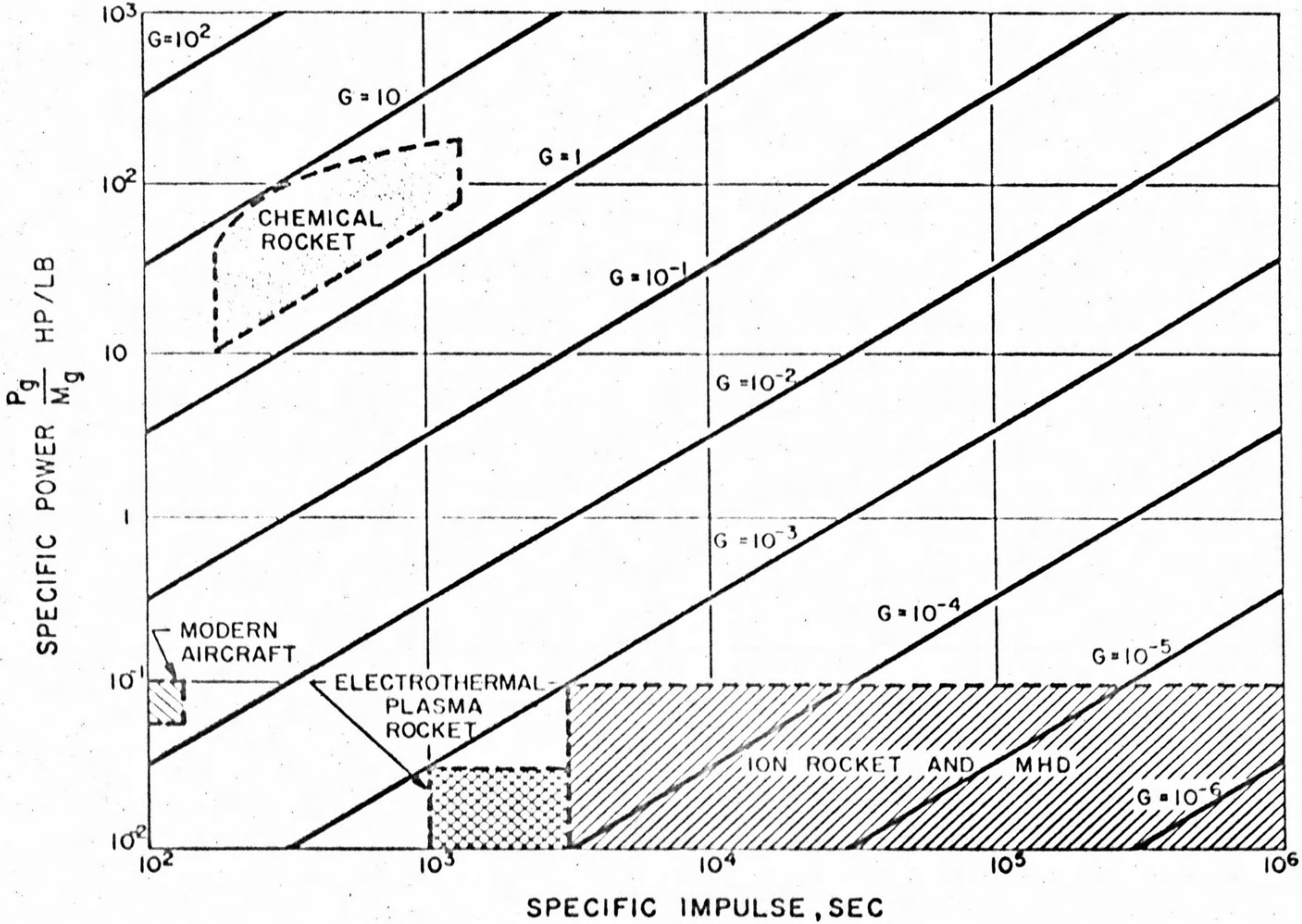


Figure 1. Specific Power Requirements of Vehicles

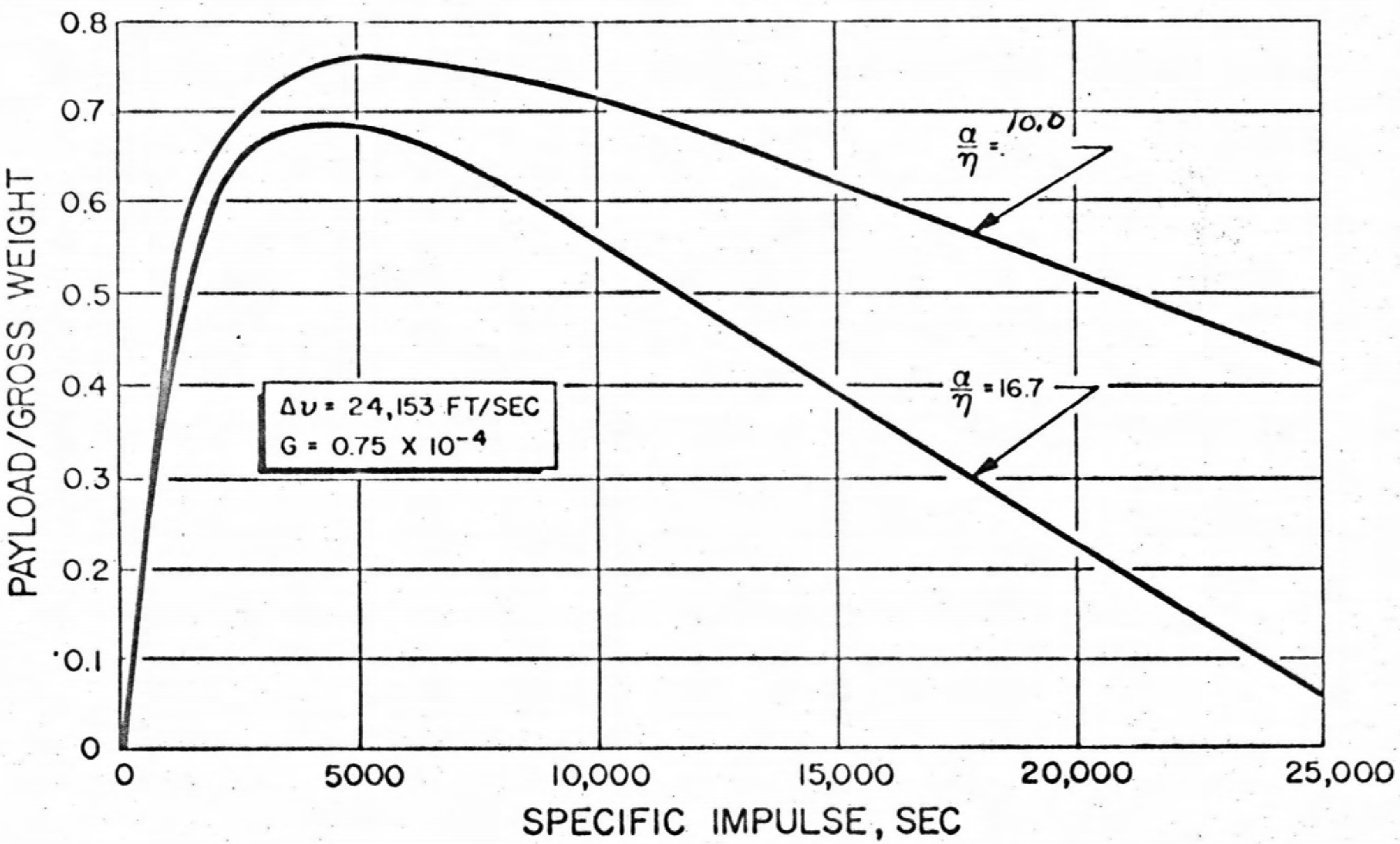


Figure 2. Typical Variation of Payload vs Specific Impulse, Ion Rocket

20b

The thrust and specific impulse of low-thrust rockets require programming to achieve the best payload and shortest flight time. (Ref. 14, 15). Because the propellant flow rate varies as I_s^{-2}, a mass flow variation of 100 is required when the specific impulse varies by a factor of 10. The estimated variations of specific impulse, I_s, and thrust-to-weight ratio, G, mass ratio, W_2/W_1, and trip times are summarized in Table II. (Ref. 14)

TABLE II

Performance Parameter Variations for

Round Trips to Mars

Trip Time Days	Specific Impulse Seconds	Thrust/Weight Ratio	Mass Ratio
500	2,500 - 37,000	$0 - 3.5 \times 10^{-4}$	1 - 0.23
800	5,000 - 38,000	$0 - 2.4 \times 10^{-4}$	1 - 0.45

Specific Power = 0.10 kw/lb

$\frac{\text{Power Plant Wt.}}{\text{Vehicle Wt.}} = 0.20$

Internal Combustion Engine Systems

The internal combustion engines are arbitrarily divided into two categories, the reciprocating systems (Ref. 7) and the gas turbines. (2,7) There are other specialized systems but they are little used because of low efficiency, complexity, etc. The details of the systems are different, but the principles are the same so that they are not considered further. The thermal efficiences of these power plants are governed by the principals of the Carnot cycle.

Reciprocating Engines

Four types of engines are available for application to different reciprocating engine systems.

(a) liquid-fuel (gasoline) - gaseous oxidizer (air), spark ignition

(b) gaseous fuel (H_2) - gaseous oxidizer (O_2), spark ignition

(c) liquid fuel (Diesel fuel) - gaseous oxidizer, (air), compression ignition

(d) liquid fuel - liquid oxidizer

The power output of a given engine cylinder is determined by six primary items:

1) the heating value of the fuel supplied to the engine, Q_f, B. T. U. per lb.

2) the weight of fuel supplied to the engine, W_f, lb./cycle

3) the mass of air (or of fuel-oxidizer mixture) which is introduced into the cylinder during a complete cycle of the engine, W_o, lb./cycle

4) the number of cycles per unit time, N per min.

5) the efficiency with which the energy entering and released is transformed to work at the piston face, η_i ; and

6) the efficiency with which the power delivered to the piston face is delivered to the engine drive shaft, η_m.

The brake horse power,

$$BHP = \frac{J\left[\left(Q_f \times \frac{W_f}{W_o}\right) W_o \times N \times \eta_i \times \eta_m\right]}{33,000}, \text{h.p.}, \qquad (25)$$

in which J = mechanical equivalent of heat

= 778 ft.lb./BTU.

The weight of air displaced per cycle is directly proportional to the valumetric efficiency, η_v ,

$$W_o = \rho L A \eta_v \text{, lb./cycle.}$$

ρ = air density, lb./ft.^{3}

L = piston stroke, ft.

A = piston area, lb./ft.^{3}

$$\eta_v = \text{valumetric efficiency} = \frac{\text{wt. of air displaced/cycle}}{\rho LA}$$

The net thermal efficiency of the internal combustion engine is the ratio of the brake horse power produced to the power release of the fuel consumed.

$$\text{Net thermal efficiency} = \frac{2.544 \times 10^5}{W_f\, q_f} \text{ per cent} \qquad (26)$$

The relative efficiency of the engine is the efficiency actually obtained to the air standard efficiency.

$$\text{Relative efficiency} = \frac{\text{net thermal efficiency}}{\text{air standard efficiency}} \times 100 \text{ percent} \qquad (27)$$

The air standard effieiency is the thermal efficiency in which the working medium is assumed to be air and it has a constant specific heat throughout the cycle.

For reciprocating engines the thermal efficiency is

$$\eta = 1 - \left(\frac{1}{r}\right)^{\gamma - 1} \qquad (28)$$

in which r is the compression ratio, the ratio of the maximum volume of the engine cylinder to the minimum. γ is the average specific heat of the combusted mixture in the engine cylinder. Maximum performance is achieved when there is a maximum flow of fuel and oxidizer through the system, i.e., a high volumetric efficiency, the ratio of the actual weight of air swept through the engine to the weight of air in the displacement volume. Because to achieve maximum power the fuel-oxidizer ratio has a well defined value for each fuel-oxidizer combination used, the volumetric-efficiency is a close measuree of the overall performance of the reciprocating engine when operating at the best fuel-air ratio.

A high compression ratio is advantageous for both spark and compression ignition engines because of high thermal efficiency. The tendency of fuels to detonate, and ultimately pre-ignite, places an upper limit upon the ratio which can be employed. The optimum ratio for a particular engine depends upon design details and the amount of supercharging.

In addition to the brake horse-power it has become customary to evaluate the performance of an engine upon the specific fuel consumption, indicated horse power, the indicated mean effective pressure and the brake mean effective pressure. The brake horse-power, BHP, can be measured directly on a torque dynamometer. The brake mean effective pressure is calculated from it.

$$B\,M\,E\,P = \frac{229\ (BHP)}{[illegible]}\ \text{p.s.i.} \qquad (29)$$

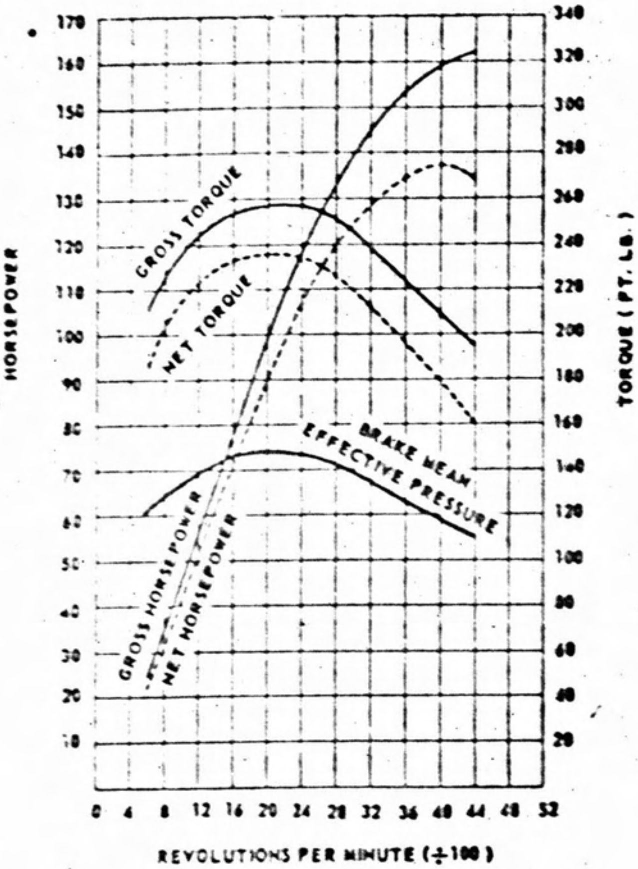
The indicated horse power is the sum of the brake horse power and the friction, or motoring, losses, or

$$I\,H\,P = \frac{B\,H\,P}{\eta_m} \qquad (30)$$

The indicated mean effective pressure which gives a closer measure of the maximum pressures in the cylinder is

$$I\,M\,E\,P = \frac{B\,M\,E\,P}{\eta_m} \qquad (31)$$

Comparison of engine cylinder pressure records and IMEP tests indicate that the maximum pressures reached in internal combustion engine cylinders are 4.5 - 6 times the IMEP, depending upon the compression ratio and operating conditions.



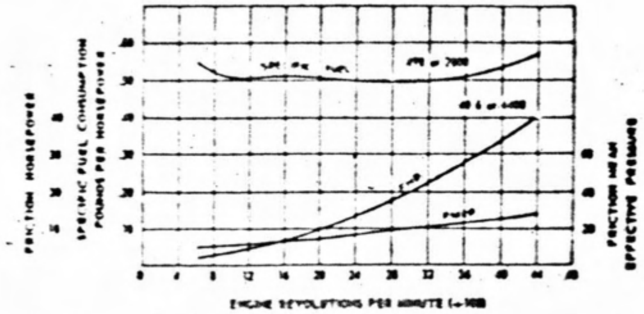


Fig. 3. Typical Performance Data of Reciprocating Engine. (Ref. 2)

The experimental performance of an internal combustion engine (IHP, IMEP, BHP, BMEP and SFC) are usually presented as a function of the engine speed. Fig. 3 is a typical presentation:

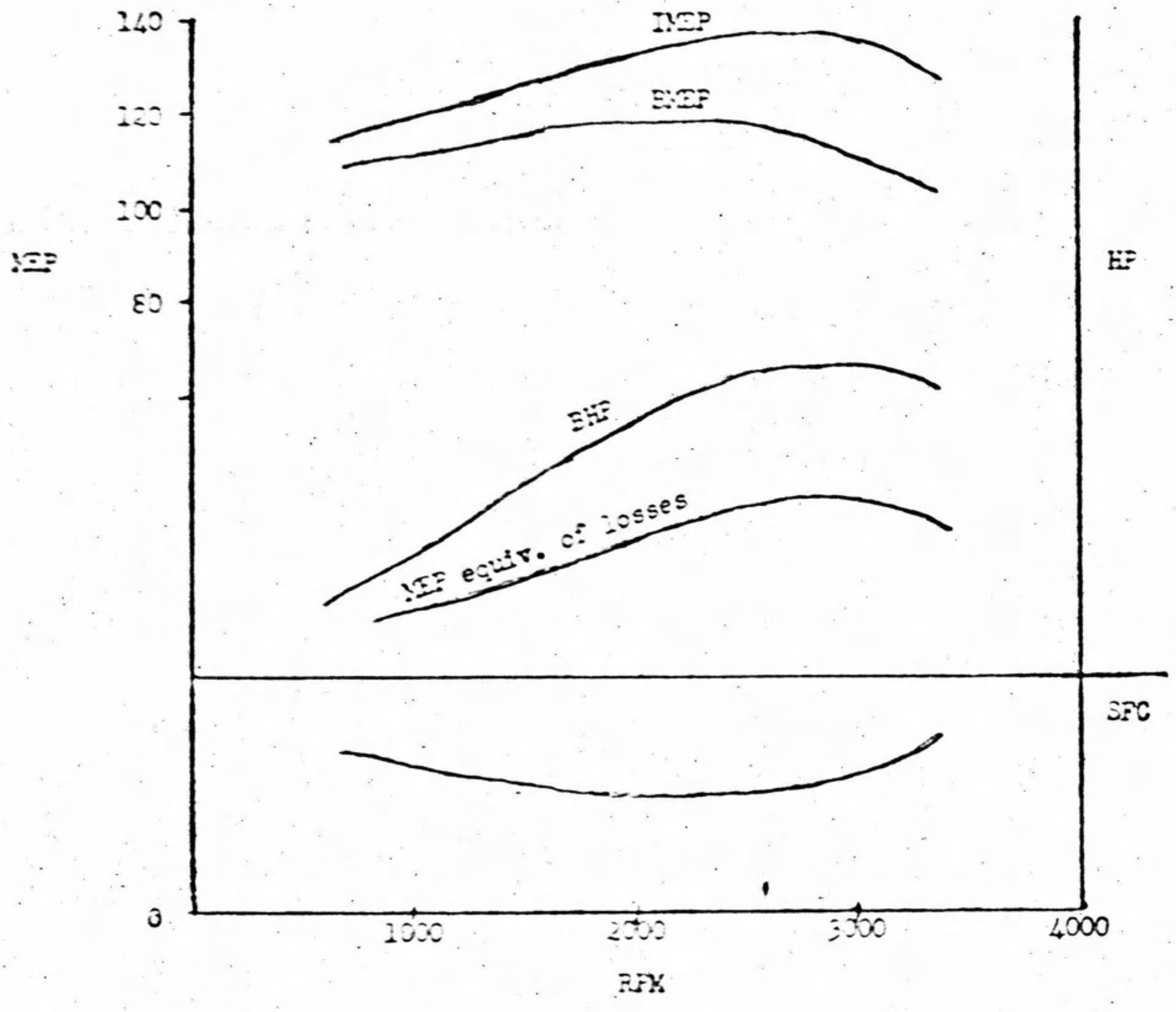


Fig. 3 Typical Performance Data
For Reciprocating Engines

The pressure records, or indicator cards, of the Otto cycle engines and the Diesel cycle engines basically do not differ. Because the Otto cycle engine is spark ignited, it can operate at lower compression ratios and pressures than the Diesel cycle which requires compression ignition of the fuel. The net result is that because of increased dynamic stresses the Diesel engine is substantially heavier than the gasoline engine and from a systems standpoint can be less desirable because of its lower specific power. In practice it is found that the ideal efficiency of the compression-ignition, or Diesel engines, are higher than the spark ignition engines but a maximum in the neighborhood of 75% of the fuel is burned in the Diesel cycle so that the full calorific value of the fuel is not achieved.

The power plant system for a flight vehicle includes the engine, gearing and propeller (Ref. 7,9) Both the engine and propeller performance are affected by the ambient operating conditions. Because of the wide variation in engine systems and ambient conditions, the performance must be considered for each specific case. In general the power output varies as the ratio of the air density and the square root of the absolute temperature.

Because the engine power required is constant if the air speed is constant, the following conclusions hold:

1) the fuel consumption per mile of a given vehicle depends only upon the indicated air speed and at a specific indicated airspeed is independent of altitude.

2) the minimum fuel consumption, which occurs at the indicated airspeed for which the power requirement is least, is independent of altitude except insofar as the specific fuel consumption, w_f, may vary with altitude.

3) the time taken to fly a given distance at any given fuel consumption per mile decreases with altitude in proportion to $\sqrt{\text{altitude density}, \sigma}$, because the flight speed varies inversely as the $\sqrt{\sigma}$.

Propellants

The propellants generally associated with internal combustion engines are gasoline for spark-ignition engine fuel, Diesel fuel for the compression ignition engines, and air as the oxidizer. Industrial use of gaseous fuels such as coal gas, blast furnace gas, natural gas and others is fairly common where economy and availability outweigh fuel storage space and high specific power. The behavior of hydrogen is unique in that it is possible to control the power from no load to full load by regulation of the hydrogen fuel supply alone up to the stochiometric mixture ratio. Auxiliary power units for space vehicles are currently proposed which use cryogenic oxygen and hydrogen (Ref. 16) and storable liquid rocket propellants, N_2O_4 and a 50:50 mixture of N_2H_4:UDMH.

Detailed descriptions, performance data and fuel requirements for reciprocating engines are summarized in Reference 2 and Table III from Ref. 17.

TABLE III

Reciprocating Engines

Manufacturer and Address	Designation	No. of cylinders	Cylinder arrangement	Propeller drive	Max takeoff power at S.L., hp	At rpm	Normal rated power, hp	At rpm	At altitude, ft	Fuel grade	Diameter or dimensions, in. (without cooling)	Blower ratio	Dry weight, lb
					Power Ratings								
Avco Corp., Lycoming Div. Williamsport, Pa.	O-235-C1B	4	He	Direct	115	2 800	108	2 600	S.L.	80	22 33x32		210
	O-290-D2C	4	He	Direct	140	2 800	135	2 600	S.L.	80/87	22 [illegible] 24		[illegible]
	O-320-A2C	4	He	Direct	150	2 700	150	2 700	S.L.	80/87	23 32x32 24		[illegible]
	O-320-B2C	4	He	Direct	160	2 700	160	2 700	S.L.	91/96	23 32x32 24		[illegible]
	IO-320-A1A	4	He	Direct	150	2 700	150	2 700	S.L.	80/87	19 38x32 24		[illegible]
	O-340-A1D	4	He	Direct	165	2 700	160	2 700	S.L.	91/96	24 72x33 37		[illegible]
	HIO-360-B1A	4	He	Direct	180	2 900	180	2 700	S.L.	91/96	19 [illegible]x33 37		[illegible]
	VO-360-A1B	4	Ve	Direct	180	2 900	180	2 900	S.L.	91/96	22 42x33 37		[illegible]
	IO-360-B1A	4	He	Direct	180	2 700	180	2 700	S.L.	91/96	22 42x33 37		291
	IMO-360-B1B	4	He	Direct	225	3 400	225	3 400	S.L.	100/130	17 47x33 37		274
	VO-435-A1E	6	Ve	Direct	260	3 400	230	3 200	S.L.	80/87	34 73x32 58		361
	VO-435-A1F	6	Ve	Direct	260	3 400	250	3 200	S.L.	80/87	24 13x33 58		394
	GO-435-C2B2-6	6	He	Gear	260	3 400	235	3 100	S.L.	80/87	29 80x33 12		438
	TVO-435-A1A	6	Ve	Direct	260	3 200	220	3 200	20 000	100/130	36 51x33 58		465
	GO-480-B1D	6	He	Gear	270	3 400	260	3 000	S.L.	80/87	29 82x33 12		432
	GO-480-C1F6	6	He	Gear	295	3 400	260	3 000	S.L.	100/130	22 49x33 12		437
	GSO-480-B1B6	6	He	Gear	340	3 400	320	3 200	6 300	100/130	23 29x33 12		500
	IGSO-480-A1B6	6	He	Gear	340	3 400	320	3 200	11 500	100/130	23 29x33 12		492
	O-540-E1A5	6	He	Direct	235	2 575	235	2 575	S.L.	80/87	24 56x33 37		395
	O-540-E1B5	6	He	Direct	260	2 800	235	2 800	S.L.	91/96	24 56x33 37		394
	O-540-A1D5	6	He	Direct	250	2 575	250	2 575	S.L.	91/96	24 56x38 12		397
	VO-540-B1B	6	Ve	Direct	305	3 200	305	3 200	S.L.	80/87	21 57x34 14		429
	VO-540-C1A	6	Ve	Direct	305	3 200	305	3 200	S.L.	80/87	21 58x34 70		[illegible]
	IO-540-A1A5	6	He	Direct	290	2 575	290	2 575	S.L.	100/130	19 42x34 25		411
	IO-540-B1A5	6	He	Direct	290	2 575	290	2 575	S.L.	100/130	19 60x34 25		417
	IO-540-C1A5	6	He	Direct	250	2 575	250	2 575	S.L.	91/96	22 [illegible] 37		[illegible]
	IGO-540-B1A	6	He	[illegible]	[illegible]	[illegible]	[illegible]	3 400	S.L.	100/130	23 66x34 25		501
	IGSO-540-B1A	6	He	Gear	[illegible]	[illegible]	[illegible]	3 200	[illegible] 500	100/130	30 29x34 25		512
	TVO-540-A1A	6	Ve	Direct	305	3 200	305	3 200	17 000	100/130	38 89x34 70		544
	IO-720-A1A	8	He	Direct	400	2 650	400	2 650	S.L.	100/130	46 [illegible]		600
Continental Motors Corp. Muskegon, Mich.	A-65-8F	4	He	Direct	65	2 300	65	2 300	S.L.	80/87	31 [illegible] 56		170
	C-85-12F	4	He	Direct	85	2 575	85	2 575	S.L.	80/87	31 29x31 56		184
	C-90-12F	4	He	Direct	90	2 475	90	2 475	S.L.	80/87	31 25x31 56		[illegible]
	E-185-9	6	He	Direct	205	2 600	185	2 300	S.L.	80/87	47 62x33 38		352
	O-200-A	4	He	Direct	100	2 750	100	2 750	S.L.	80/87	30 57x31 56		190
	E-225-4	6	He	Direct	225	2 650	185	2 300	S.L.	80/87	47 62x33 38		355
	O-300-A,-B,-C,-D	6	He	Direct	145	2 700	145	2 700	S.L.	80/87	39 [illegible]x31 50		268
	GO-300-A,-B,-C,-D	6	He	G .750:1	175	3 200	175	3 200	S.L.	80/87	39 42x31 50		[illegible]
	O-470-4	6	He	Direct	225	2 600			S.L.	91/96			415
	O-470-15	6	He	Direct	213	2 600			S.L.	80/87			405
	O-470-J	6	He	Direct	225	2 550	225	2 550	S.L.	80/87	36 03x33 32		374
	O-470-K,-L	6	He	Direct	230	2 600	230	2 600	S.L.	80/87	36 03x33 56		404
	O-470-G,-M,-R[1]	6	He	Direct	240	2 600	240	2 600	S.L.	91/96	37 56x33 56		410
	IO-470-J,-K	6	He	Direct	225	2 600	225	2 600	S.L.	80/87	37 57x33 56		404
	IO-470-C,-G,-P,-R	6	He	Direct	250	2 600	250	2 600	S.L.	91/96	37 57x33 56		432
	IO-470-D,-E,-F,-H,-L,-M,-N	6	He	Direct	260	2 625	260	2 625	S.L.	100/130	43 57x33 56		426
	TSIO-470-B	6	He	Direct	260	2 600	260	2 600	18 000	100/130	42 92x33 56		427
	FSO-526-A[2]	6	He	Direct	270	3 200	270	3 200	17 000	91/96	50 33x36 44	10.04	557
Franklin Engine Co., Inc. Syracuse, N. Y.	4A4-100-B3	4	He	Direct	100	2 550				80			200
	6A4-150-B3	6	He	Direct	150	2 600				80			274
	6A4-165-B3	6	He	Direct	165	2 800				80			274
	6AG4-185-B12	6	He	G .612:1	185	3 100				80			272
	6A4-200-C5	6	He	Direct	200	3 100				91			307
	6V4-200-C32, C33; O-335-5,-6	6	Ve	Direct	200	3 100				91			297
	6V-335-B[3]; O-425-1	6	Ve	Direct	215	3 275				80			272
	6V-335-A,-B	6	Ve	Direct	210	3 100				[illegible]			295
	6A-335	6	He	Direct									
Pratt & Whitney Aircraft Div. United Aircraft Corp. East Hartford, Conn.	R-2000-D5	14	Rad	G .500	1 450	2 700	1 200	2 550	[illegible]	100/130	49 10	7.15	1 565
	R-2000-2SD13-G	14	Rad	G .500	1 450	2 700	1 200	2 550	[illegible]	100/130	49 10	7.15+9.52	1 565
	R-2800-CB3	18	Rad	G .450:1	2 400	2 800	1 800	2 600	[illegible]	100/130	52 [illegible]	7.29	2 357
	R-2800-CB4	18	Rad	G .450:1	2 500	2 800	1 800	2 600	[illegible]	108/135	52 [illegible]	7.29	2 357
	R-2800-CB16	18	Rad	G .450:1	2 400	2 800	1 800	2 600	[illegible]	100/130	52 [illegible]	7.29+8.58	2 [illegible]
	R-2800-CB17	18	Rad	G .450:1	2 500	2 800	1 800	2 600	[illegible]	108/135	52 [illegible]	7.29+8.58	2 [illegible]
	R-2800-CH5,-54	18	Rad	Direct	2 100	2 700	1 [illegible]	2 [illegible]	7 [illegible]	115/145	52 [illegible]	7.29	2 [illegible]
	R-2800-B15,-46W	18	Rad	G .450:1	2 500	2 800	1 900	2 600	7 [illegible]	115/145	52 [illegible]	7.29+8.58	2 461
Wright Aeronautical Div., Curtiss-Wright Corp. Wood-Ridge, N. J.	[illegible]C9HE1-2 / R-1820-[illegible]	9	Rad	G .5625:1	1 525	2 800	1 275	2 500	3 500	115/145	55 74	7.21	1 [illegible]
	[illegible]TC18EA12 / R-3350-32W,-32WA	18	Rad	G .4375:1	3 700	2 900	2 800	2 600	4 100	115/145	56 6	6.46	3 560
	[illegible]C9HE3 / R-1820-84C,-84D	9	Rad	Direct	1 525	2 800	1 275	2 500	3 500	115/145	55 74	7.21	1 419
	[illegible]C9HE12	9	Rad	Direct	1 525	2 800	1 275	2 500	3 500	115/145	55 74	7.21	1 419
	[illegible]C7BA1	7	Rad	Direct	800	2 600	700	2 400	7 000	91/96	56 45	7.21	1 040
	[illegible]C9HE12	9	Rad	G .5625:1	1 525	2 800	1 275	2 500	3 500	115/145	54 98	7.21	1 479
	[illegible]C9HE12	9	Rad	Direct	1 525	2 800	1 275	2 500	3 500	115/145	55 75	7.21	1 422
	[illegible]C9HD[illegible]	9	Rad	Direct	1 425	2 700	1 275	2 500	3 300	100/130	54 53	7.21	1 363
	[illegible]TC18EA6	18	Rad	G .4375:1	3 400	2 900	2 800	2 600	4 700	115/145	56 59	6.46	3 [illegible]

G Geared
He Horizontally opposed
Rad Radial

S.L. — Sea level
Ve Vertically opposed
[3] For helicopter installation

[1] Extended propeller shaft
[2] Engine has a 92-lb. turbocharger that is mounted separate

Gas Turbine

A gas turbine is a heat engine in which the available energy of the working fluid is converted into kinetic energy by expansion through a nozzle from which the fluid issues as a jet. (Ref. 9, 18) The jet kinetic energy is partially converted into mechanical energy by directing the jet across contoured blades mounted in the periphery of the turbine wheel, or spool. The pressure on the blades, causing rotary motion, is developed from the change in momentum of the working fluid jets in passing through the array of these blades.

The performance of this simple, ideal system depends upon the gas flow, the pressure ratio in the working fluid across the wheel and the turbine inlet temperature. It has been confirmed, both analytically and experimentally, that at a specific gas flow

1) the power output increases with pressure ratio, for each turbine inlet temperature, to a maximum value and then decreases at a lesser rate with any further pressure ratio increase.

2) the power output increases markedly as the inlet temperature increases for a specific pressure ratio, and

3) the maximum power outputs, as the inlet temperatures are increased, occur at increasing pressure ratios.

The limiting thermal efficiency, η_a, of this ideal system is:

$$\eta_a = 1 - \left(\frac{1}{r}\right)^{\frac{\gamma - 1}{\gamma}} \tag{32}$$

The experimentally determined thermal efficiency is a function of the output brake horse power, the specific fuel consumption, w_f, and heat release of the fuel, q_f.

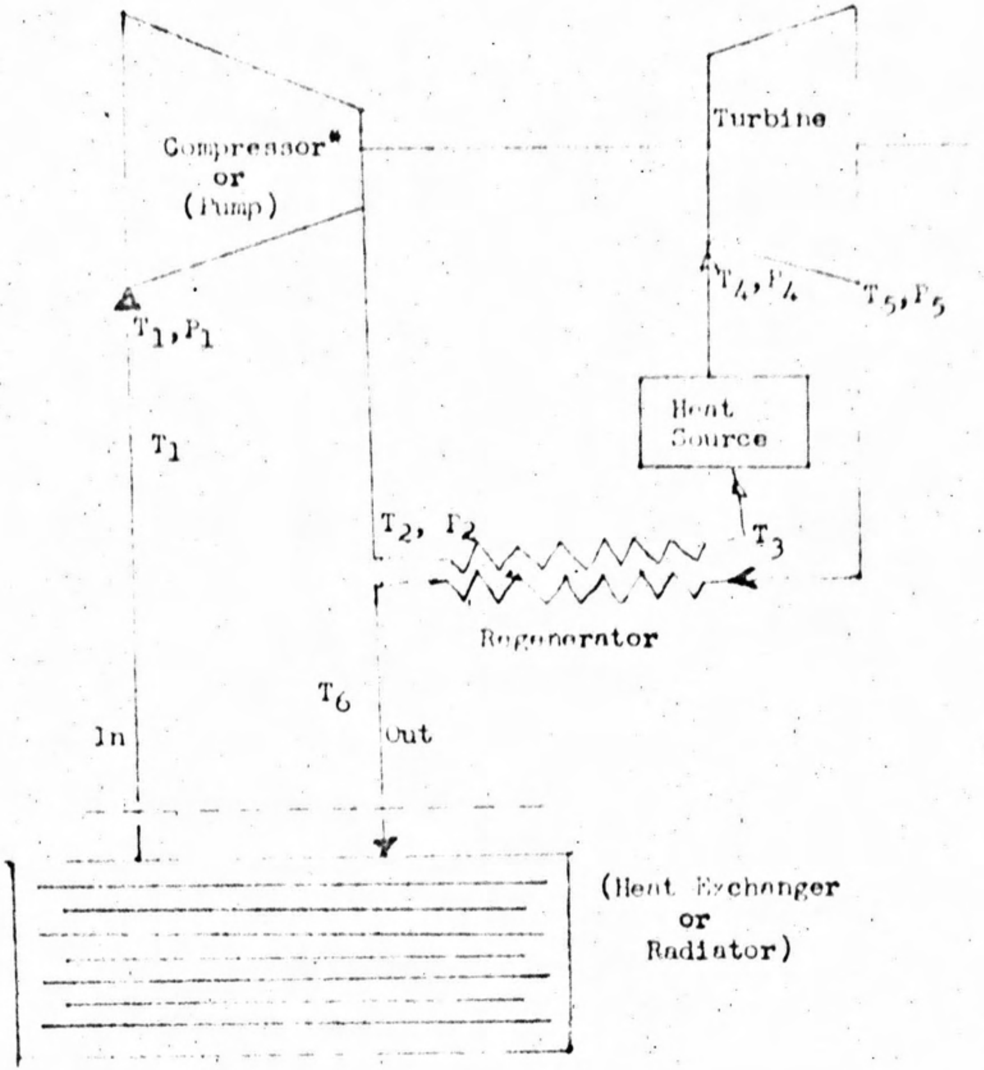
$$\text{Thermal efficiency (experimental)} = \frac{2.545 \times 10^5}{q_f \, w_f} \tag{33}$$

An analysis of the open cycle regenerative gas turbine power plant system, Fig. 4, is based on the assumptions that 1) the working fluid is a perfect gas, 2) there are no changes in the working fluid due to the combustion of fuel in the combustion chambers and 3) there are no pressure drops. (1) and (2) have compensating errors. An accurate analysis requires that the variation of the specific heat and the ratio of specific heats be considered. (3) is small because the gas turbine operates at air: fuel ratios of 60-70 in contrast to the internal combustion engine value of 13-14.

$$\eta = \eta_m \, \eta_b \, \eta_a \, \frac{\eta_t T_4 - \frac{T_1}{\eta_c} r^{\frac{\gamma-1}{\gamma}}}{T_4 - 2 T_1 \, r^{\frac{\gamma-1}{\gamma}}} \tag{34}$$

Figure 4

Elements of a Regenerative Gas Turbine Power Plant



( ) Used in closed cycle systems only

* Used in Brayton cycle

in which

$$y = 1 - \eta_R (1 - \eta_t \eta_a)$$

$$z = (1-\eta_R) \left[\left(\frac{1}{r}\right)^{\frac{\gamma - 1}{\gamma}} + \frac{\eta_a}{\eta_c} \right]$$

The overall efficiency of the engine system depends upon

1) the ideal cycle efficiency, η_a

2) the combustion chamber efficiency, η_b

3) the mechanical efficiency of the turbines, compressor drives, gear train and propeller, η_m

4) the internal efficiency of the turbine, η_t

5) the internal efficiency of the compressor, η_c

6) the pressure ratio, $r = \frac{P2}{P1} = \frac{P5}{4}$

7) the specific heat ratio, γ,

8) the inlet temperature to the turbine, T_4

9) the inlet temperature to the compressor, T_1

10) the effectiveness of the regenerator, η_R

$$\eta_R = \frac{\text{enthalpy change, cold side}}{\text{enthalpy change, heat side}}$$

Tables for computing the efficiency and specific examples are available in Reference 9.

The closed cycle system, which is characterized by the regenerative steam-turbine, can operate on the Brayton cycle in which the working fluids are gaseous in all parts of the engine system, or on the Rankine cycle in which the working fluid is condensed in the radiator or heat exchanger. The condensed liquid is pumped back through the system to be partially reheated in the regenerator which recovers part of the waste heat from the cycle, then the temperature is brought to its working level in the heat source.

Open cycle gas turbine systems are applicable in land and marine vehicles and preferable in aircraft applications because no radiator is required. The systems with no regeneration are characterized by simplicity, light weight, compactness, low cost but poor fuel consumption at full load and very poor under variable, or partial, load operation. The regenerative systems have specific fuel consumptions in the ranges from 0.40 to 0.80, and can have net thermal efficiences in the range from 20-35% over a wide range of loads. These systems are compact, light weight, of high reliability and are easily maintained.

The working fluid of these power plant systems is the air-fuel combustion product. The air-fuel ratio is high, 60-70:1, in comparison with that of the reciprocating engines, 13-14:1. Fuels are dictated by cost, availability and the application. They include pulverized coal, kerosine, heavy fuel oil such as Bunker C, or fuel gas. Aircraft gas turbines use the JP series of fuels. These are distillation cuts taken in the manufacturing process to satisfy the operating temperature requirements, but produce the highest fraction of gasoline.

The closed cycle systems show excellent fuel consumption under all conditions. This advantage is offset by the cooling requirements. The system weight includes the working fluid, the radiator or heat exchanger and the associated piping. A system operating on the all gas Brayton cycle tends to be heavier than one operating on the condensing Rankine cycle because of higher operating pressures.

The working fluids for the closed cycle systems are summarized in Table IV.

Detailed specifications of many gas turbine engines are summarized in Ref. 2 and 3 Table V of Ref. 17.

TABLE IV

WORKING FLUIDS FOR
CLOSED CYCLE GAS TURBINE SYSTEMS

BRAYTON CYCLE	RANKINE CYCLE
Air	Water
Steam	Alkali Metals (N_a, R_b, K, L_i)
Mercury Vapor	Mercury
Helium	Ammonia
Hydrogen	Diphenyl
Other Gases	Dowtherm A
	Sulfur
	Aluminum bromide
	Hydrocarbons

TABLE V

[illegible]. Gas Turbine Engines

Manufacturer and Address	Military designation	Type	No. compressor stages	No. turbine stages	No. combustors	Max. power at S.L.	Specific fuel consumption at max. power	Compression ratio at max. rpm	Max. envelope diameter, in.	Max. envelope length, in.	Dry weight, less tailpipe, lb.	Remarks
Avco Corp., Lycoming Div., Stratford, Conn.	T53-L-1A, -1B	ACFS	5 1	1	1	[illegible] shp	0.7[illegible]	6.0	2[illegible]	47.61	484	Bell UH-1A, Kaman HH-43B
	T53-L-3	AFP	5 1	[illegible]	1	[illegible] shp	0.6[illegible]	6.0	[illegible]	[illegible]	524	Grumman OV-1
	T53-L-5	ACFS	5 1	1	1	960 shp	0.6[illegible]	6.0	2[illegible]	4[illegible]	4[illegible]	Bell UH-1B
	T53-L-7	AFP	5 1	1	1	1,100 [illegible]	0.6[illegible]	6.0	2[illegible]	[illegible]	533	Advanced Grumman OV-1A
	T53-L-9, -9A, -9B	ACFS	5 1	1	1	1,100 shp	0.6[illegible]	6.0	2[illegible]	47.61	4[illegible]5	Bell 204, Bell UH-1D
	T53-L-11	ACFS	5 1	1	1	1,100 shp	0.6[illegible]	6.0	2[illegible]	47.61	505	Bell UH-1F
	T55-L-5	ACFS	7 1	2	1	2,2[illegible] shp	0.[illegible]	6.0	24.25	4[illegible].21	5[illegible]	Vertol CH-47A
	T55-L-7	ACFS	7 1	2	1	2,650 shp	0.[illegible]	6.0	24.25	[illegible]	5[illegible]	Vertol CH-47A (Advanced)
	LTC4[illegible]	AFP	7 1	2	1	2,445 shp	0.[illegible]	6.0	24.25	[illegible]	7[illegible]5	North American YAT-28E
The Boeing Company, Seattle, Wash.	T50-BO-8A	CFS	[illegible]	2	2	300 shp	0.9[illegible]	4.6	2[illegible]	[illegible]	3[illegible]	Navy DASH (anti-submarine destroyer carried drone helicopter program)
	T50-BO-10	ACFS	1 1	2	2	330 shp	0.85	4.9	22.5	25.5	2[illegible]8	As above
	T50-BO-4	CFS	[illegible]	2	2	550 shp	0.72	6.1	25.5	55	2[illegible]5	Turboshaft
	T60-520-10	CFS	[illegible]	2	2	6[illegible] shp	0.6[illegible]	6.2	25.5	55	240	Turboshaft
	T60-520-11	CFS	[illegible]	2	2	7[illegible] shp	0.5[illegible]	6.2	25.5	57.2	250	Turboshaft
	55[illegible]	ACFS	1 1	2	2	4[illegible] shp	0.7[illegible]	5.3	22.5	[illegible].5	1[illegible]	Turboshaft
	502-[illegible]B	CFP	2	2	2	205 ahp		4.3	2[illegible]	4[illegible]	4[illegible]	Compressed-air output
Continental Aviation & Engineering Corp., Detroit, Mich.	J69-T-25	CFJ	[illegible]	1	1	1,025 lb.t.	1.14		[illegible]	3[illegible]	3[illegible]	Cessna T-37B
	J69-T-29	ACFJ	1 1	1	1	1,700 lb.t.	1.10			4[illegible].3	3[illegible]5	Ryan Q-2C
	J69-T-[illegible]	ACFJ	[illegible]	2	1	2,550 lb.t.	0.9[illegible]	6.1	2[illegible]	4[illegible]	3[illegible]5	Target missile engine
	T[illegible]	ACFS	2	3	1	2[illegible] shp	0.71	5.4	1[illegible].2	3[illegible].7	130	
	[illegible]B	ACFS	1 1	[illegible]	1	600 shp	0.6[illegible]		[illegible].4	4[illegible].5	220	Turboshaft
	[illegible]B	ACFP	1 1	3	1	6[illegible] eshp	0.6[illegible]		[illegible].4	42.5	245	Turboprop
	14	CFP	[illegible]	1	1	191 ahp					1[illegible]	For pressure-jet helicopter
	[illegible]-23	ACFJ	3	2	1	2,400 lb.t.	0.94	6.1	22.3	47	410	
Curtiss-Wright Corp., Wright Aeronautical Div., Wood-Ridge, N. J.	J65-W-16A	AFJ	13	2	1	7,700 lb.t.			37.5	1[illegible]	2,7[illegible]	North American AF-1E
	J65-W-18	AFJ	13	2	1	10,500 lb.t.			37.5	1[illegible]	3,445	Grumman F-11A
Garrett Corp., AiResearch Mfg. Co., Phoenix, Ariz.	GTP30	CFS	[illegible]	1	1	50 shp	1.4[illegible]	[illegible]	17.[illegible]	29.[illegible]	[illegible].5	Portable generators
	GTCP30	CFSG	[illegible]	1	1	50 shp, 20 ahp	1.6[illegible]	[illegible]	17.[illegible]	2[illegible]	[illegible].5	On-board power, Jet Star, Gulfstream
	GTP70	CFS	2	[illegible]	[illegible]	[illegible] shp	[illegible]	[illegible]	[illegible]	[illegible].6	2[illegible]	Sergeant missile support
	GTC85	CFG	2	[illegible]	1	[illegible] ahp	[illegible]	4.0	[illegible]	33.5	[illegible]	Airport ramp support
	GTCP85	CFSG	2	1	1	345 shp, 215 ahp	1.[illegible]	4.0	23.[illegible]	32.4	2[illegible]	On-board auxiliary power for B-727, [illegible], C-141, Caravelle, KC-135
	GTCP105	CFSG	2	2	6	[illegible], 435 ahp	1.0[illegible]	5.5	25.9	36.5	[illegible]	B-52 ramp support
	TSE231	CFFS	2	[illegible]	1	5[illegible] shp	0.[illegible]2	6.5	2[illegible]	40.8	200	Helicopter engine
	GTP[illegible]	CFS	2	3	1	4[illegible] shp	0.[illegible]	6.5	3[illegible]	3[illegible]	[illegible]	Fixed ground-generator installations
General Electric Co., Flight Propulsion Div., Evendale, Ohio	J47-GE-25	AFJ	12	1	8	7,200 lb.t.	1.0[illegible]	5.4	3[illegible].3	1[illegible]5	2,5[illegible]	Boeing B-47E, water alcohol injection
	J47-GE-27	AFJ	12	1	8	5,970 lb.t.	1.0[illegible]	5.4	3[illegible].3	1[illegible]8	2,[illegible]	No afterburner
	J47-GE-33	AFJ	12	1	8	7,[illegible] lb.t.	1.15	5.5	[illegible].3	22[illegible]	3,[illegible]	Afterburner
	J73-GE-3	AFJ	12	2	8	8,[illegible] lb.t.	0.92	7.0	3[illegible].8	1[illegible]7	3,[illegible]	No afterburner
	J79-GE-2	AFJ	17	3	10	16,150 lb.t.	1.9[illegible]	12.9	3[illegible].3	20[illegible]	3,[illegible]	McDonnell F-4B, North American A-5A
	J79-GE-3B	AFJ	17	3	10	14,[illegible] lb.t.	2.04	12.0	3[illegible].3	2[illegible]	3,[illegible]25	Lockheed F-104A, B
	J79-GE-5B	AFJ	17	3	10	15,[illegible] lb.t.	2.0[illegible]	12.2	3[illegible].3	20[illegible]	3,[illegible]25	Convair B-58
	J79-GE-7	AFJ	17	3	10	15,[illegible] lb.t.	1.9[illegible]	12.2	3[illegible].3	20[illegible]	3,[illegible]75	CF-104, F-104C, D
	J79-GE-8	AFJ	17	3	10	17,[illegible] lb.t.	1.9[illegible]	12.9	3[illegible].3	20[illegible]	3,[illegible]	McDonnell F-4B, North American A-5A
	J79-GE-11A	AFJ	17	3	10	15,[illegible] lb.t.	1.9[illegible]	13.[illegible]	3[illegible].3	20[illegible]	3,[illegible]	Lockheed F-104G
	J79-GE-15	AFJ	17	3	10	17,[illegible] lb.t.	1.9[illegible]	12.9	3[illegible].3	20[illegible]	3,[illegible]	McDonnell F-4C
	J93-GE-3	AFJ				30,000 lb.t.						For NAA RS-70
	X353-5B	AFLF	8	2 1	1	7,4[illegible] lb.t.	0.3[illegible]	1.1	7[illegible].6	1[illegible].8	1,[illegible]45	Lift-fan for XV-5A
	CJ805-3A	AFJ	17	3	10	11,200 lb.t.	0.[illegible]	13.0	3[illegible]	1[illegible]	2,[illegible]	Convair 880, length includes reverser
	CJ805-3B	AFJ	17	3	10	11,[illegible] lb.t.	0.[illegible]	13.0	3[illegible]	1[illegible]	2,[illegible]	Convair 880M, length includes reverser
	CJ805-23	AFF	17	3	10	16,[illegible] lb.t.	0.5[illegible]	13.0	5[illegible]	1[illegible]	3,[illegible]	Aft-fan engine, Convair 990A
	CJ805-23C	AFF	17	3	10	16,[illegible] lb.t.	0.5[illegible]	13.0	5[illegible]	1[illegible]	3,[illegible]	Aft-fan with reverser, Caravelle Super A
Lynn, Mass.	T58-GE-8B	AFS	10	3	1	1,250 shp	0.[illegible]	8.3	1[illegible]	59	2[illegible]	[illegible], UH-2A, CH-46A, CH-3B, [illegible], VH-3A, CH-1[illegible], X-22[illegible]
	T64-GE-2	AFS	14	4	1	2,[illegible] shp	0.50	12.8	30	8[illegible]	[illegible]	
	T64-GE-4	AFS	14	4	1	2,[illegible] shp	[illegible]	12.8	3[illegible]	1[illegible]	1,13[illegible]	de Havilland Caribou testbed
	T64-GE-[illegible]	AFS	14	4	1	2,[illegible] shp	0.5[illegible]	12.8	3[illegible]	[illegible]	[illegible]	XC-142A, CH-53A
	T64-GE-[illegible]	AFS	14	4	1	2,[illegible] shp	0.[illegible]	12.8	[illegible]	1[illegible]	[illegible]	de Havilland CV-7A
	J85-GE-5A	AFJ	8	2	1	3,[illegible] lb.t.	2.[illegible]	6.6	2[illegible]	1[illegible]	5[illegible]	Northrop T-38A, Bell X-14A, Afterburner
	J85-GE-13	AFJ	8	2	1	4,0[illegible] lb.t.	2.[illegible]	7.0	2[illegible]	1[illegible]	5[illegible]	Northrop F-5A, Afterburner
	CT58-1[illegible]	AFS	10	3	1	1,250 shp	0.6[illegible]	8.3	18	5[illegible]	3[illegible]	Vertol [illegible], S-62, Commercial T58
	CJ610-1	AFJ	8	2	1	2,850 lb.t.	0.9[illegible]	6.8	1[illegible]	4[illegible]	3[illegible]	Aero Commander 1121, Lear Jet, [illegible], Piaggio-Douglas PD-808, Commercial J85
	CF700-2B	AFF	8	2	1	4,200 lb.t.	0.66	6.8	34	65	640	Aft-fan, J85 frame
General Motors Corp., Allison Div., Indianapolis, Ind.	T56-A-1	AFP	14	4	6	3,750 shp	0.54	9.2	3[illegible]	14[illegible]	1,645	Lockheed C-130A
	T56-A-7	AFP	14	4	6	4,050 eshp	0.5[illegible]	9.7	3[illegible]	14[illegible]	1,833	Lockheed C-130B
	T56-A-8	AFP	14	4	6	4,050 eshp	0.5[illegible]	9.5	4[illegible]	14[illegible]	1,872	Grumman E-2A
	T56-A-9	AFP	14	4	6	3,750 eshp	0.54	9.2	37	145	1,[illegible]76	Lockheed C-130A, GV-1
	T56-A-10W	AFP	14	4	6	4,5[illegible] eshp	0.5[illegible]	9.5	4[illegible]	14[illegible]	1,[illegible]	P-3A, water/alcohol injection
	T63-A-5	ACFS	6 1	4	1	[illegible]250 shp	0.7[illegible]		22.5	40.5	1[illegible]	For Army LOH
	501-D13	AFP	14	4	6	3,750 eshp	0.54	9.2	4[illegible]	14[illegible]	1,645	Lockheed Electra
Solar, San Diego, Calif.	T62-[illegible]	CFS	1	1	1	80 shp	1.10	3.4	15.6	24.[illegible]	61	Single-shaft, constant-speed, APU
	T[illegible] Mars [illegible]	CFS	1	1	1	370 shp	1.0	4.1	2[illegible]	[illegible]	1[illegible]	Single-shaft, air bleed, multipurpose APU

Note: Type of engine is designated by capital letters. First, or first two, designate compressor: A = axial; C = centrifugal. Next letter, F, stands for flow. Final letter or letters define output: J = jet; P = propeller; F = fan; S = shaft; G = gas; and LF = lift fan. For example: AFLF = Axial-flow lift fan.

ABBREVIATIONS:
ahp. — Air horsepower
eshp. — Equivalent shaft horsepower
lb.t. — Pounds of thrust
shp. — Shaft horsepower

Continued on page 188

TABLE I (CON'T)

Gas Turbine Engines continued

Manufacturer and Address	Military designation	Type	No. compressor stages	No. turbine stages	No. combustors	Max. power at S. L.	Specific fuel consumption at max. power	Compression ratio at max. rpm	Max. envelope diameter, in.	Max. envelope length, in.	Dry weight, less tailpipe, lb.	Remarks
United Aircraft Corp. Pratt & Whitney Aircraft Division E. Hartford, Conn.	TF33-P-3	AFF	15	4	8	17,000 lb. t.	0.52	13.0	53		3,900	JT3D-2, B-52H
	TF33-P-5	AFF	15	4	8	18,000 lb. t.	0.54	13.0	53		4,170	JT3D-3A, Boeing C-135B
	TF33-P-7	AFF	16	4	8	21,000 lb. t.	0.61	16.0	53		4,490	JT3D-8A, Lockheed C-141
	TF33-P-9	AFF	15	4	8	18,000 lb. t.	0.54	13.0	53		4,170	JT3D-3A, Boeing KC-135B
	J52-P-3	AFJ										JT8A, for Hound Dog missile
	J52-P-6A	AFJ	12	2	9	8,500 lb. t.	0.82	12.0	30	106	2,056	JT8B-1, Douglas A-4E, Grumman A-6A
	J52-P-8	AFJ	12	2	9	8,500 lb. t.	0.86	12.0	30	108	2,118	JT8B-3
	J57-P-20	AFJ	16	3	8	18,000 lb. t.	2.35	13.0	38.9	253	4,750	JT3C-26, F-8D
	J57-P-43 WB	AFJ	16	3	8	13,750 lb. t.	0.85	13.0	38.9	167	3,870	JT3C, B-52F & G
	J58	AFJ				30,000 lb. t.						
	J60-P-3	AFJ	9	2	8	3,000 lb. t.	0.96	6.5	21.9	78	436	JT12A-5, North American T-39, Lockheed C-140
	J60-P-4	AFJ	9	2	8	3,300 lb. t.	1.00	6.5	21.9	78	465	JT12A-7
	J75-P-19W	AFJ	15	3	8	26,500 lb. t.	2.20	12.0	43.0	258	5,960	Water inj. + afterburner, Republic F-105D
	JT3C-6	AFJ	16	3	8	13,500 lb. t.	0.78	13.0	38.9		4,234	Boeing 707-120, Douglas DC-8-10
	JT3C-7	AFJ	14	3	8	12,000 lb. t.	0.79	13.0	38.9		3,495	Boeing 720
	JT3C-12	AFJ	16	3	8	13,000 lb. t.	0.82	13.0	38.9		3,550	Boeing 720
	JT3D-1	AFF	15	4	8	17,000 lb. t.	0.52	13.0	53		4,060	Boeing 707-120B, -720B, Douglas DC-8-50
	JT3D-3	AFF	15	4	8	18,000 lb. t.	0.54	13.0	53		4,170	Boeing 707-320B, C, Douglas DC-8-50, DC-8F
	JT3D-8A	AFF	16	4	8	21,000 lb. t.	0.61	16.0	53		4,490	Lockheed L-300
	JT4A-9	AFJ	15	2	8	16,800 lb. t.	0.81	12.0	43	144	5,050	Douglas DC-8-20/-30, Boeing 707-320
	JT4A-11	AFJ	15	2	8	17,500 lb. t.	0.84	12.0	43		5,100	Douglas DC-8-20/-30, Boeing 707-320
	JT8D-1	AFF	13	4	9	14,000 lb. t.	0.80	15.0	44		2,994	Boeing 727, Caravelle Super B
	JT8D-5	AFF	13	4	9	12,000 lb. t.	0.57	14.0	44		2,994	Transport engine
	JFTD12A-1	AFF	9	4	8	4,050 shp.		6.5			970	Sikorsky S-64 Skycrane
	JFTD12A-2	AFF	9	4	8	4,050 shp.	0.66	6.5			970	Free turbine shaft drive
	JT12A-6	AFJ	9	2	8	3,000 lb. t.	0.96	6.5	21.9	78	436	Lockheed Jet Star, NAA Sabreliner
	JT12A-8	AFJ	9	2	8	3,300 lb. t.	1.00	6.5			465	Dassault Mystere 20
	JT12A-21	AF	9	2	8	4,025 lb. t.	2.42	6.5	21.9		651	Afterburner

U. S. VTOL Aircraft

	General Data		Dimensions				Weights			Powerplant					
Manufacturer and Address	Designation	Type	No. of seats	Span	Length	Height	Wing area sq. ft.	Empty weight lb.	Gross weight lb.	Number, make and model	Rotor-propeller diameter	No. of propellers	Service ceiling, ft.	Maximum speed, mph	Remarks
Bell Aerosystems Co. Buffalo, N.Y.	X-22A tri-service VTOL transport	Tilting ducted fans	6	39.2	36.3	15.2	409	10,500	15,300	4 GE T58-8B	8'	4		375	
Curtiss-Wright Corp. Wood-Ridge, N.J.	X-19	Tilting propellers	6	34.4	43.2	16.0		10,900	12,300	2 Lyc. T55-L-5	13' 0"	4	9,500	460	
Hawker Siddeley Aircraft Ltd. Great Britain	Hawker P.1127 (XV-6A)	British VTOL strike-reconnaissance fighter for U. S. Army use	1	24.3	41.2	10.3				1 Bristol Siddeley Pegasus lift thrust		0			
Lockheed-Georgia Co. Marietta, Ga.	XV-4A (VZ-10) (Hummingbird)		2	26.9	33.8	11.8	104	4,965	7,200	2 P&W JT12A-3		0		520	First flight July 1962
Piasecki Aircraft Corp. Philadelphia, Pa.	VZ-8PA (AirGeep 1)	Ducted fan	4	9.4	26.1	6.7		2,000	2,350	1 A. Research TNE 331-A	7.5	2			Tactical ground support
	VZ-8PB (AirGeep 2)	Ducted fan		9.3	24.6	5.8		2,500	3,670	2 Gas. Artouste IIC		2			VZ-8P with ground powered wheels
Ryan Aeronautical Co. San Diego, Calif.	VZ-3RY	Vectored slipstream	1	23.5	27.7	10.7	125	2,593	2,718	1 Lyc. T53	9' 2"	2			
	XV-5A (VZ-11)	Lift fan	2	29.7	44.6	14.8	256	6,931	9,200	GE X353-4	6' 0"		30,000		
Vanguard Air & Marine Corp. Bala-Cynwyd, Pa.	Model 2D	Fan-in-wing with ducted pusher prop	2	22.0	30.4	9.2			3,000	1 Lyc. T53	6.5'	2			Test bed. Wing-buried rotor-props.
Vertol Division The Boeing Co. Morton, Pa.	VZ-2A (Model 76)	Tilt wing	2	24.9	25.4	10.0	110	2,300	3,300	1 Lyc. T53	9.8'	2			

TABLE II (CON'T)

Gas Turbine Engines

Manufacturer and Address	Designation	Type	No. of compressor stages	No. of turbine stages	No. of combustors	Max. power @ S.L.	Specific fuel consumption at maximum power	Compression ratio at max. rpm	Maximum envelope diameter, in.	Maximum envelope length, in.	Dry weight, less tailpipe lb.	Remarks
CANADA Orenda Engines, Div., Toronto, Ontario	Orenda 10	AFJ	10	1	6	6,355 lb. t.	1.12	5.3	42	123	2,515	
	Orenda 11	AFJ	10	2	6	7,275 lb. t.	0.99	6.1	45	121	2,425	
	Orenda 14	AFJ	10	2	6	7,275 lb. t.	0.99	6.1	45	121	2,425	
	J79-OEL-7*	AFJ	17	3	10	15,800 lb. t.		12	35.31	204	3,290+	*CF 104—Built under license
	J79-11A*	AFJ	17	3	10	15,800 lb. t.		12	34.31	204	3,290+	
	J85-Can-40	AFJ	8	2	1	3,000 lb. t.	0.99	7.1	17.7	40.5	335	Canadair CT-114 Tutor
GREAT BRITAIN HAWKER SIDDELEY GROUP Bristol Siddeley Engines, Ltd. Filton	Artouste 510	CFJ	1	2	1	220 shp.	328+ lb/hr	75	19.1	37	235	+Actual fuel consumption
	Astazou 2	ACFP	1-1	3	1	523 shp.	.632	6.0	18.1	59	271	Short Skyvan
	Astazou 3	ACFP	1-1	3	1	562 shp.	.583		18.1	59	271	
	Aubisque	AFF	1-1	2	1	1,547 lb. t.	.60	7.0		51.5	498	
	Cumulus	CFG	1	2	1	230 shp.		3.95	17.7	34.4	210	
	Double Mamba 8	AFP	11	3	24	3,000 shp.	.66	5.6	51.29	103.35	2,500	Gannet
	Gnome H 1000	AFS	10	1	1	1,050 shp.	.64	8.30	16.0	55	296	GET 58-6 equiv. Whirlwind 10, Agusta 204B & 1010
	Gnome H 1200	AFS	10	1	1	1,250 shp.	.62	8.30	16.0	55	314	GET 58-8 equiv.
	Gnome P 1200	AFP	10	1	1	1,150 shp.	.65	8.30	31.6	85	615	Turboprop version of H 1200
	Coupled H 1200	AFS	10	2	1-1	2,500 shp.	.62	8.30	41.8	77.13	673	Wessex Mk. 2, 1,250 shp. is individual engine output at gearbox input.
	Gyron Jr. DGJ 2	AFJ	2	4	1	7,000 lb. t.			32.3	121.1	2,100	Buccaneer 1
	Gyron Jr. DGJ 10	AFJ	2	4	1	14,000 lb. t.			32.3	191	2,100	Bristol T 188
	Nimbus 150 (Series)	ACFJ	2	1	1	800 shp. (1 hr.)	.69	6.0		62	390	Westland Wasp & Scout
	Olympus 104	AFJ	7 & 6	1 & 1	10	13,500 lb. t.			42.4	132	3,615	Avro Vulcan Mk. 1
	Olympus 201	AFJ	5 & 7	1 & 1		17,000 lb. t.			41.7	129.4	3,650	Avro Vulcan Mk. 2
	Olympus 301	AFJ	5 & 7	1 & 1		20,000 lb. t.			42.0	135.6	3,900	
	Supersonic Olympus	TJ	CIVIL AND MILITARY VERSIONS UNDER DEVELOPMENT FOR SUPERSONIC APPLICATIONS									
	Olympus 2001	Ind	5 & 7	1 & 1	8	23,600 shp.	.52			312	35,000	15 Megawatt generator, industrial
	Olympus 2002	Ind	5 & 7	1 & 1	8	22,000 shp.	.52			255	10,000	Frigates, hydrofoils, GEMs etc.
	Orpheus 701	AFJ	7	1	7	4,700 lb. t.	1.057	4.36	32.4	73	790	Folland Gnat
	Orpheus 703	AFJ	7	1	7	4,850 lb. t.	1.063	4.36	32.4	75.5	825	HF-24
	Orpheus 803	AFJ	7	1	7	5,000 lb. t.	1.04	4.38	32.4	95.1	900	G91, Standard VI, Tam
	Orpheus 805	AFJ	7	1	7	4,000 lb. t.	1.04		32.4	75.5	825	T1A Trainer, FUJI
	Orpheus 101	AFJ	7	1	7	4,230 lb. t.	1.06	4.39	32.4	77.5	900	Gnat Mk. 1, Jet Trainer
	B.OR. 12	AFJ	8	2	7	6,810 lb. t.	.957	5.50	32.4		1,170	
	B.OR. 12 (SR)	AFJ	8	2	7	8,170 lb. t.	1.62	5.50	32.4	191.4	1,150	Afterburner, Basic Thrust 6810 lbs.
	Palas 600	AFJ	1	1	1	390 lb. t.	1.20	4.12	19.0	46	170	
	Palouste 502	CFJ	1	2	1		309 lb/hr		19.1	34	215	Canadair CL 66 Ground Starting Units
	Palouste 505	CFJ	1	2	1			3.91	19.1	34	214	
	Pegasus	AFL	UNDER DEVELOPMENT FOR V/STOL APPLICATIONS									
	Proteus 705	ACFP	1 & 12	2-2	8	3,290 lb. t.	.55	7.2		123	3,008	Britannia 102
	Proteus 761	ACFP	1-12	2-2	8	4,195 ehp.	.56			123	2,900	
	Proteus 755	ACFP	1-12	2-2	8	3,900 lb. t.	.56	7.3		123	2,990	Br. 250, 300, 310, 320
	Proteus 1277	Ind	1-12	2-2	8	2.64 MW				100	3,050	3 MW. Generator, industrial
	Proteus 1270	M	1-12	2-2	8	4,250 shp.	.575			112	2,985	"Brave Borderer" Class Marine
	Sapphire 100	AFJ	13	2	24	8,300 lb. t.	.882	7.46		109.8	2,670	Javelin
	Sapphire 200	AFJ	13	2	24	11,000 lb. t.	.86	7.46		107.7	3,050	Javelin, Victor
	Sapphire 209 (TR)	AFJ	13	2	24	12,270 lb. t.	1.2	7.49		107.7	3,190	Limited A/B Javelin
	Turmo 601	CFJ	2	1	1	400 shp.	4.62 lb/hr	5.6	19.25	47	233	V.A. 3 Hovercraft
	Turmo 602	CFJ	2	3	1	425 shp.		5.6	19.25	49	275	
	Turmo 603	CFJ	2	3	1	425 shp. (1 hr.)	1.10	5.6	19.25	49	313	
	Viper 8	AFJ	7	1	24	1,750 lb. t.	1.07	4.0	24.3	67	310	Provost Mk. 3
	Viper 9	AFJ	7	1	24	1,900 lb. t.	1.12	4.25	24.3	67	310	HP. 115
	Viper 11	AFJ	7	1	24	2,500 lb. t.	1.07	4.31	24.5	64	549	MB. 326 Macchi; Jet Provost Mk. 304
	Viper 12	AFJ	7	1	24	2,700 lb. t.	1.10	4.42	24.5	64	549	Uprated ASV. II.
	Viper 20	AFJ	8	1	24	3,000 lb. t.	.985	5.06	24.2	68.7	575	D.H. 125, PD. 808
	B.E. 100	UNDER DEVELOPMENT										
Rolls-Royce, Ltd. Derby	Avon RA. 24R	AFJ	NA	NA	8	14,420 lb. t.*	NA	NA	41.5	126	NA	*Nominal performance
	Avon RA. 29/1 Mks. 522, 526, 522A, 526A	AFJ	16	2	8	10,500 lb. t.	0.730	9.1	36	126	3,328	
	Avon RA. 29/1 Mk. 524B & 525B	AFJ	16	3	8	10,290 lb. t.	0.730	9.08	36	126	3,342	
	Avon RA. 29/3 Mk. 527	AFJ	16	3	8	11,300 lb. t.	0.749	9.35	36	126	3,347	
	Avon RA. 29/5 Mk. 531	AFJ	17	3	8	12,200 lb. t.	0.724	10.1	36	134	3,491	
	Avon RA. 29/6 Mk. 532R	AFJ	17	3	8	12,200 lb. t.	0.731	10.1	36	134	3,498	
	Avon RA. 29/6 Mk. 533R	AFJ	17	3	8	12,600 lb. t.	0.771	10.2	36	134	3,491	
	Avon RB. 146R	AFJ	NA	NA	8	16,500 lb. t.*	NA	NA	41.5	145	NA	*Nominal performance
	Conway R. Co. 11	AFF	NA	NA	10	17,250 lb. t.	NA	NA	42	135	NA	
	Conway R. Co. 12 Mks. 508 & 509	AFF	7x9	2x1	10	17,500 lb. t.	0.590	14.1	42	136	4,542	
	Conway R. Co. 17	AFF	NA	NA	10	20,500 lb. t.	NA	NA	42	138	NA	
	Conway R. Co. 42	AFF	7x9	2x1	10	20,370 lb. t.	0.603	14.8	50	154	5,001	
	Conway R. Co. 43	AFF	7x9	2x1	10	21,825 lb. t.	0.605	15.8	50	154	5,101	
	Dart R. Da. Mk. 506	CFP	2	2	7	1,545 shp.	0.746	5.4	38	95	1,036	
	Dart R. Da. Mk. 510	CFP	2	2	7	1,575 shp.	0.725	5.4	38	96	1,106	
	Dart R. Da. 6 Mks. 511 & 511-7E	CFP	2	2	7	1,525 shp.	0.690	5.4	38	96	1,098	
	Dart R. Da. 6 Mk. 514	CFP	2	2	7	1,710 shp.*	0.590	5.4	38	99	1,114	*Water methanol restoration
	Dart R. Da. 7 Mk. 520	CFP	2	3	7	1,630 shp.	0.710	5.6	38	98	1,207	
	Dart R. Da. 7/1 Mk. 525	CFP	2	3	7	1,720 shp.	0.665	5.6	38	98	1,277	
	Dart R. Da. 7/2 Mks. 526 & 527	CFP	2	3	7	1,835 shp.	0.665	5.6	38	98	1,227	

(Continued on p. 215)

TABLE II (cont'd)

Gas Turbines

Manufacturer and Address	Designation	Type	No. of compressor stages	No. of turbine stages	No. of combustors	Max. power at S. L.	Specific fuel consumption at maximum power	Compression ratio at max. rpm	Maximum envelope diameter, in.	Maximum envelope length, in.	Dry weight, less tailpipe, lb.	Remarks
Rolls-Royce Ltd. Derby (Cont.)	Dart R. Da. 7/2 Mk. 529-7 & 529-7F	CFP	2	3	7	1,835 shp.	0.683	5.6	38	98	1,223	
	Dart R. Da. 7/2 Mk. 529-7 & 529-7E	CFP	2	2	7	1,910 shp.	0.658	5.6	38	98	1,223	
	Dart R. Da. 7/2 Mk. 529-8 & 529-8F	CFP	2	3	7	1,910 shp.	0.658	5.6	38	98	1,235	
	Dart R. Da. 7/2 Mk. 530 & 531	CFP	2	3	7	1,835 shp.	0.683	5.6	38	98	1,227	
	Dart R. Da. 10/1 Mk. 542-10	CFP	2	3	7	2,750 shp.*	0.638	6.35	38	94	1,377	*Water methanol restoration
	Dart R. Da. 7/7 Mk. 21	CFP	2	3	7	1,670 shp.	0.650	5.5	38	94	1,268	
	Dart R. Da. 8 Mk. 101	CFP	2	3	7	2,470 shp.*	0.622	5.5	38	94	1,207	*Water methanol restoration
	RB 108	AFJ	8	2		2,[illegible] lb. t.*		5.25	20.8	48.5	269	*Nominal performance
	RB 145	AFJ	NA	NA		2,750*	NA	NA	22.0	59.6	NA	*Nominal performance
	RB 162	AFJ	NA	NA	NA	4,[illegible] lb. t.	NA	NA	31.6	25	275	
	RB 168	NA	NA	NA	NA	NA	NA	NA	NA	NA	NA	
	Spey R. Sp. 1 Mk. 505-5, 10 & Mk. 505-14	AFF	4×12	2×2	10	9,850 lb. t.	0.545	16.8	37	110	2,200	
	Spey R. Sp. 1 Mk. 505	AFF	4×12	2×2	10	9,850 lb. t.	0.545	16.8	38	110	2,200	
	Spey R. Sp. 2 Mk. 506	AFF	4×12	2×2	10	10,410 lb. t.		17.8	38	110	2,215	
	Spey R. Sp. 2W Mk. 506-2W & 506-14W	AFF	4×12	2×2	10	10,410 lb. t.		16.8	29	110	2,276	Water injection
	Spey R. Sp. 4 Mk. 510	AFF	5×12	2×2	10	11,000 lb. t.			36	115	2,312	
	Tyne R. Ty. 1 Mk. 506	AFP	6×9	3×1	10	4,[illegible] ehp.	0.485	13.5	43.2	109.7	2,275	
	Tyne R. Ty. 11 Mk. 512	AFP	6×9	3×1	10	4,[illegible] ehp.	0.471	13.5	43.2	109.7	2,275	
	Tyne R. Ty. 12 Mk. 512	AFP	6×9	3×1	10	5,[illegible] ehp.	0.455	13.5	43.2	109.7	2,177	
	Tyne R. Ty. 12 Mk. 101	AFP	6×9	3×1	10	5,0[illegible] ehp.	0.455	13.5	43.2	109.7	2,219	
	Tyne R. Ty. 20 Mk. 21	AFP	6×9	3×1	10	5,[illegible] ehp.	0.448	13.5	43.2	109.7	2,203	
	Tyne R. Ty. 20 Mk. 22	AFP	6×9	3×1	10	5,[illegible] ehp.	0.448	13.5	43.2	109.7	2,218	
FRANCE Hispano-Suiza Paris	H. S. Verdon	CFJ	1	1	9	7,[illegible] lb. t.	1.10	4.6	56	104	2,072	
	H. S. Tyne	AFP	6+6	1+3	10	6,[illegible] ehp.	0.439	13.6	43.2	108.7	2,167	
S.N.E.C.M.A. Paris	Atar E3	AFJ	8	1	A	6,160 lb. t.	1.03	4.8	35.4	184.8	2,136	
	Atar [illegible]	AFJ	8	1	A	8,[illegible] lb. t.	2.03	4.8	35.4	278	2,710	Afterburner
	Atar 09B	AFJ	9	2	A	9,700 lb. t.	0.98	5.5	34.2	153	2,293	
	Atar 09B	AFJ	9	2	A	13,230 lb. t.	2.20	5.5	40	247	2,940	Afterburner
	Atar 09C	AFJ	9	2	A	13,520 lb. t.	2.03	5.7	40	234	3,025	Afterburner
	Atar 09K	AFJ	9	2	A	15,435 lb. t.	2.18	6.0	40	234	3,153	Afterburner
	TF 106	AFF	16	4	8	18,845 lb. t.	2.25		30	237.4	3,330	Afterburner
Societe Turbomeca Paris	Artouste 2C	CFS	1	2	A	500 eshp.	87	3.88	1.8	4.72	253	Alouette 2-Sud Aviation helicopter
	Artouste 3B	ACFS	2	3	A	500 eshp.	65	5.20	1.8	4.90	287	Sud Aviation SE 3160
	Astazou 2	ACFS	2	3	A	5[illegible] eshp.	60	6.00	1.5	4.55	290	Sud Aviation Alouette-Agusta
	Astazou 3	ACFS	2	3	A	5[illegible] eshp.	60	6.00	1.5	5.75	271	Dornier DO 27-Breda SFERMA "Marquis"-Piper 440-Nord 1110-SIPA 251
	Aubisque	...	3	2	A	1,550 lb. t.	50	6.9	1.8	6.78		SAAB 105
	Astafan 2		4	3	A	3.5[illegible] lb./sec.	422 lb./hr.	4.2	2.0	4.33	287	
	Bastan 4	ACFP	2	3	A	1,000 eshp.	61	5.5	1.8	6.20	469	Hurel-Dubois-Bréguet 1500-Sud Aviation "Voltigeur"-"Diamant" SFERMA PD 175-Nord 2[illegible] Max Holste "Super Broussard" GAM Dassault 415, 418
	Bastan 6	...	2	3	A	1,100 eshp.	58	5.6	1.8	6.20	468	Nord Aviation 2[illegible] 262 "Super Broussard"
	Bi-Bastan		2	3	A	1,500 eshp.	64	5.5	2.4	6.50	996	Sud Aviation 3[illegible] helicopter
	Gabizo	ACFJ	2	1	A	2,420 lb. t.	1.06	5.2	2.2	6.93	585	
	Gourdon	ACFJ	2	1	A	1,[illegible] lb. t.	1.04	6.2	1.8	5.75	375	
	Marboré 2	CFJ	1	1	A	880 lb. t.	1.06	3.9	1.8	5.13	306	CM 170 & 175 Fouga Magister-MS 760 Morane Saulnier-CT20 Aerfer Cessna Beech Pilatus T-47A-Temco TT1 Cessna VT1-Rene Q2
	Marboré 6	CFJ	1	1	A	1,090 lb. t.	1.05	3.9	1.9	4.8	308	Can replace Marboré 2
	Piméné	CFJ	1	1	A	232 lb. t.	1.17	3.55	1.3	4.0	158	Fouga-Bréguet 90-SIPA S200 & S-300
	Palouste 4	...	1	2	A	2.32 lb./sec.	290 lb./hr.	3.80	1.8	4.0	206	Djinn-S0 1221 Helicopter ground starting "Bearn"
	Turmo 3C(T)	...	2	3	A		63	5.50	2.15	5.5	414	Sud Aviation SE 3200
	Turmo 3C3	...	2	2+2	A	1,500 eshp.	61	5.50	2.15	5.5	414	Sud Aviation "Super Frelon" helicopter
	Turmo 3D(T)		2	3	A	1,335 eshp.	61	5.50	2.97	6.0	881	Bréguet 941-942
	Turmo 3D(3)		2	2+2	A		63	5.50	2.97	6.0	661	G.A.M. Dassault "Spirale 3"
	Turmo 4		2	2+2	A	2,000 eshp.	55		2.97	6.0	1,102	Bréguet 941
ITALY Fiat-SMA Turin	4700 000	CFG	1	1+1	A	550 gshp.	1.38	4.0	24.1	54.2	335	Free turbine driving alternator and five-blade compressor
	4301 000	CFP	1	1+2	A	670 shp.	1.11	4.0	22.4	58.1	398	Free turbine driving the reduction gear
	4350 000	CFP	1	1+1	A	670 shp.	1.11	4.0	22.4	48.8	344	Free turbine driving the reduction gear
	4350 001	CFP	1	1+1	A	750 shp.	0.99	4.0	22.4	52.3	440	Free turbine driving the reduction gear
	6004 000	CFJ		1	A	594 lb. t.	1.18	4.0	22.4	36.2	209	
	6004 001	CFJ		1	A	594 lb. t.	1.18	4.0	22.4	34.78	193	

of pilots
s of run-
sy
ccepted
RANE
California

Jet Engines

The propeller, driven by a reciprocating engine or gas turbine, the turbo-jet engine, the ramjet and the pulse jet engine all obey the same physical laws. An idealized analysis (Ref. 9) suffices to establish the general characteristics of each type powerplant and indicate the situation in which each is best applied to a vehicle system. In the cases of the propeller and turbo-jet powered vehicles, the thrust forces generated are used for take-off and to maintain the rate of climb in the flight path, to maintain the cruising speed of the vehicle by overcoming its drag and to accelerate the vehicle to a higher velocity. Each action can occur in sequence or concurrently. The ramjet and pulse jet engines must be accelerated initially by external means to a sufficiently high velocity to develop sufficient pressures and air densities within the engine systems to achieve this objective.

Acceleration of an air mass flow, $\dot{W}_0$, from an initial velocity, V, the vehicle flight velocity, to the jet velocity, C, generates a reaction force equal to the product of the mass flow and the velocity increment. This follows from the momentum theorem. If fuel is added in the combustion chamber, Fig. 5a, it is accelerated from rest to the exit velocity, C, and contributes a small amount of thrust as does the pressure differential, $P_j - P_o$, that may exist across the engine system.

The total thrust from the accelerated air and fuel mixture is

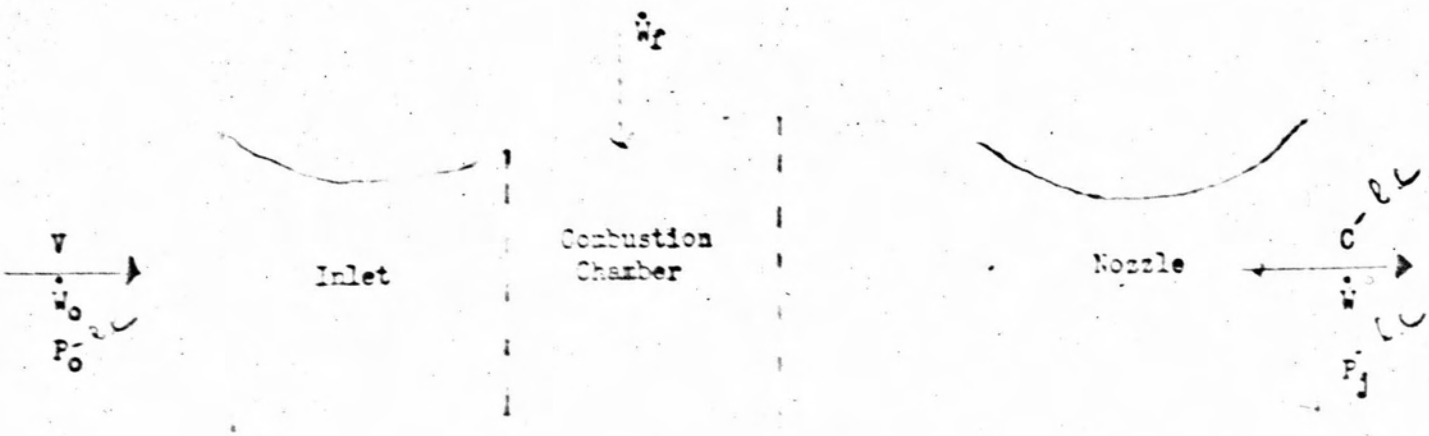
$$F = \frac{\dot{W}_a}{g}(C - V) + \frac{\dot{W}_f}{g}C + A_j(P_j - P_o) \qquad (35)$$

In practice the air-fuel ratio in jet engines is between 50:1 and 70:1, and P_j is little different from P_o. The first term describes the magnitude of the thrust with an accuracy of a few percent. Therefore, in terms of the air specific impulse $I_{s_a} = C/g$, the thrust becomes

$$F \cong \dot{W}\, I_{s_a}\left(1 - \frac{V}{C}\right) \qquad (36)$$

which is a satisfactory formula to use in estimating the net thrust of the propeller and jet engine systems under all operating conditions.

The internal efficiency, η_i, is a measure of the effectiveness with which the power, either mechanical in the case of the propeller drive or thermal in the case of the jet systems, supplied to the engine system is transformed into propulsion power.



a) Thermal Jet Engines

b) Propeller

FIG. 5 SCHEMATIC OF PROPELLER & THERMAL JET ENGINE SYSTEMS

$$\eta_i = \frac{P}{J\,\dot{w}_f\,Q_f} = \frac{c^2}{2g\,J\,\dot{w}_f\,Q_f}\left[(\dot{w}_o + \dot{w}_f)^2 - \dot{w}_f^{\,2}\left(\frac{V}{c}\right)^2\right] \qquad (37)$$

$$\cong \frac{c^2\,\dot{w}_o^{\,2}}{2g\,J\,\dot{w}_f\,Q_f} \;;$$

in which

$$P = \frac{FV}{375} + \frac{1}{750}\,\frac{(\dot{w}_o - \dot{w}_f)}{g}\,c^2 - \frac{1}{750}\,\frac{W_o\,V^2}{g}\,,\ \text{h. p.} \qquad (38)$$

The propulsive efficiency, η_p, of a jet powered vehicle is the ratio of the power required by the vehicle and the total power, the sum of the vehicle power and the residual jet power.

$$\eta_p = \frac{2\,\frac{V}{c}}{1 + \left(\frac{V}{c}\right)^2} = \frac{\frac{2.933\,V}{g\,I_s}}{1 + \left(\frac{[illegible]\,V}{g\,I_s}\right)^2} \qquad (39)$$

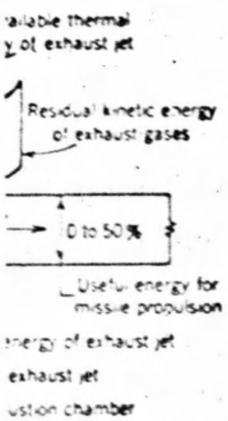
in which, V, the absolute vehicle velocity, and c, the effective jet velocity are in the same units, and I_s, sec., is the specific impulse. As shown in Fig. 6; the propulsive efficiency rapidly increases as the jet velocity approaches the vehicle velocity, reaches a maximum when the velocities are equal, then slowly DECREASES as the jet velocity exceeds the vehicle velocity.

Comparison of propeller driven and jet powered engine systems are made by averaging the flow velocity over the propeller disk. In this way an average value of the specific impulse can be assigned to a propeller driven vehicle.

chemical energy of the
It is defined as

(1–5)

e effective exhaust ve-
er unit of propellant at



m for rocket.

n factor, the so-called

ist gases is unavailable
s the nozzle as residual
ost in the high temper-
engines.
v much of the kinetic
elling a vehicle. It is

$$\eta_p = \frac{\text{Vehicle energy}}{\text{Vehicle energy} + \text{Residual kinetic jet energy}}$$

$$= \frac{Fv}{Fv + \frac{1}{2}(w/g)(c - v)^2}$$

$$= \frac{2v/c}{1 + (v/c)^2} \qquad (1\text{–}6)$$

where F is the thrust in pounds, v is the absolute vehicle velocity in feet per second, c is the effective rocket exhaust velocity with respect to the vehicle in feet per second, w is the weight flow rate in

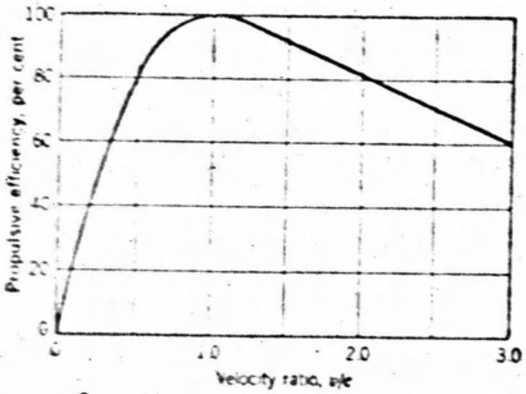


FIGURE . Propulsive efficiency at varying velocities.

pounds per second, and η_p is the propulsive efficiency. The propulsive efficiency is a maximum when the forward vehicle velocity is exactly equal to the exhaust velocity. Then the residual kinetic energy and the absolute velocity of the jet are zero and the exhaust gases stand still in space. Since the rocket exhaust velocities of present-day propellants are on the order of 5000 to 12,000 feet per second, the vehicle velocities would have to be 3400 to 6800 miles per hour for a maximum propulsive efficiency. If the vehicle velocity exceeds the magnitude of the exhaust velocity, the energy conversion into vehicle energy will be less efficient than at the condition of $v = c$.

The total efficiency, η, of an air jet propulsion system is the product of the internal and the propulsive efficiencies. It is a measure of the effectiveness with which the chemical energy of the fuel is converted into propulsive power.

$$\eta = \frac{c^2}{g\,J\,\dot{W}_f\,Q_f}\;\frac{\left[(\dot{W}_o + \dot{W}_f)^2 - \dot{W}_f^{\,2}\left(\frac{V}{c}\right)^2\right]}{\left[1 + \left(\frac{V}{c}\right)^2\right]}$$

$$\cong \frac{\dot{W}_o^{\,2}\,c^2}{g\,J\,\dot{W}_f\,Q_f}\left[1 - \left(\frac{V}{c}\right)^2\right] \qquad \begin{matrix} V < c \\ \dot{W}_o > \dot{W}_f \end{matrix} \qquad (40)$$

$$\cong \frac{F\,c\,r}{J\,\dot{W}_f\,Q_f} = \frac{3600\,C}{J\,W_f\,Q_f}$$

The propulsion efficiency, η_p, the propulsion power, P, the air speed, V, and the diameter of the jet, D, define the best type of power plant system for different flight regimes. The efficiency is determined in terms of the propulsion parameter, $V\,(D^2\rho/P)^{1/3}$. (Ref. 9

$$V\left(\frac{D^2\rho}{P}\right)^{1/3} = \frac{1.364\,\eta_p}{(8 - 12\,\eta_p + 4\eta_p^{\,2})^{1/3}} \qquad (41)$$

Fig. 7 shows the variation of the propulsion efficiency with the propulsion parameter.

Rearranging and taking the cube root of both sides of the equation gives

$$V\sqrt[3]{\frac{D^2\rho}{P}} = \frac{0.86\eta_p}{\sqrt[3]{1-\eta_p}} \tag{21}$$

In this equation the density ρ refers to the average density of the air in the plane of the propeller. Owing to the increase in pressure

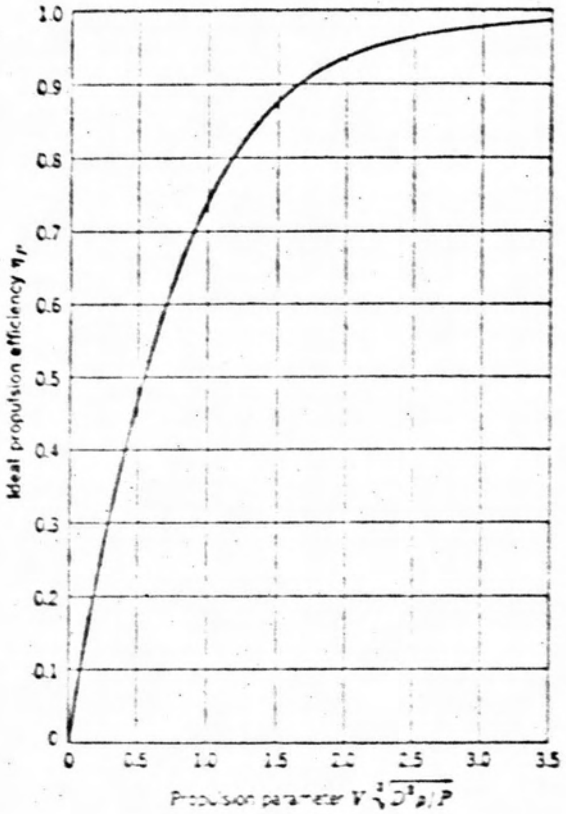


FIG. 1. Ideal propulsion efficiency vs. propulsion parameter for the ideal engine system.

in the plane of the actuator disk the air density is somewhat higher than the atmospheric pressure. This difference enters to the one-third power and is sufficiently small to be neglected for estimating purposes.

Figure 8 is a plot of the propulsion parameter $V\sqrt[3]{D^2\rho/P}$ as a function of the ideal propulsion efficiency. This curve gives the maximum attainable propulsion efficiency for the propeller system.

These data are applicable to the ideal propeller and the jet engine. For the propeller, D is its diameter, ρ is the atmospheric density, V is the airplane speed and P is the engine power. Figure 7 gives the maximum obtainable efficiency for the four different types of propulsion systems.

Power requirement for flight vehicles increase as the cube of the air speed. Furthermore, compressibility effects reduce the propeller efficiency seriously in the speed range of 400-500 mph. At low speed the propeller diameter can be made larger to offset the low speed. High vehicle speeds are limited to high altitude flight because of compressibility effects and aerodynamic heating. The following table IV summarizes roughly the flight regimes for jet-power vehicles. The overall efficiency of a turbo-prop engine and those jet engines, weighted to allow for the different power plant weights are compared in Fig. 8 (Ref. 19).

in that order. By adding a known weight to the rotor tip, a weight growth factor can be determined as can the additional engine weight required to support it. For a typical jet engine, 1 lb. added to the compressor rotor tip involves a total engine weight penalty to the extent of 12 lbs. The aircraft designer would have to multiply this by the aircraft growth factor. It is this knowledge that 1 lb. saving on blade weight can save 12 lbs. of engine weight that causes us to attach very great importance to detail design of rotor blades.

(5) *Form of Propulsion*

To obtain an idea of the efficiency of conversion of the energy of the fuel to useful work, we have to introduce a forward speed term with specific consumption. Fig. 12 shows a comparison of overall efficiency of a high pressure ratio turboprop and of the optimum jet engine over a range of flight Mach Number.

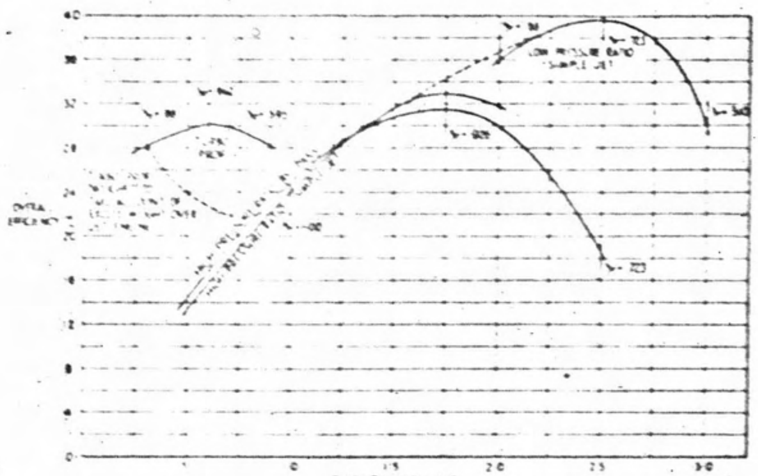


FIG. 8 Overall efficiency over a range of flight speed (11)

It is not generally realised that the jet engine operates at its maximum thermal efficiency at around $M = 2.5$, the exact value being largely dependent upon air intake efficiency. The high pressure ratio turboprop operates at its maximum overall efficiency at around Mach Number 0.6, this value being largely dependent upon the order of propeller efficiency that may be assumed at the higher speeds.

Between these two extremes of the low pressure ratio simple jet at 2.5 Mach Number and the propeller turbine at low subsonic speeds, there are simple jet engines of higher and higher compression ratio as the speed is reduced. An efficiency curve for one such engine is illustrated, and the dotted curve shows the envelope of such engines. For comparison the performance of a high pressure ratio by-pass engine is plotted. There are in general a family of such engines covering the gap between this and the propeller turbine to cover ducted fan types.

These efficiency curves should be "weighted" to allow for the different power-plant weights. Such a weighted curve is shown for the propeller turbine. It is

TABLE IV

FLIGHT REGIMES FOR VEHICLE POWER PLANTS

ENGINE TYPE	SPEED RANGE M.P.H.	ALTITUDE TESTING FT.
Reciprocating, unsupercharged	0 - 250	15,000
Reciprocating supercharged	150-450	30,000
Turbo-pump	150-600	45,000
Pulse-jet	300-600	10,000
Turbo-jet	350-Mach 3.5	60,000
Ram-jet	Mach 1-Mach 5	45,000-150,000
Supersonic Ramjet	Mach 5-Mach 25	50,000- -150,000

The ram-jet engine is the simplest of the thermal jet engines. It cannot produce static thrust and must rely on some other form of propulsion to bring it up to flight speed. At this point the action of the inlet duct and diffuser is to collect sufficient air and slow it down to produce a high enough pressure to permit sustained combustion of the fuel. To accomplish this the fuel is injected upstream of an array of flame holders which serve to mix the fuel and air and supply a sheltered region in which the combustion can be maintained under relatively low velocity conditions. The combustion products are then expanded through a suitably designed nozzle to generate thrust. Controlling the air-flow and fuel-air ratio to maintain combustion is difficult because of relatively low pressures and high velocities in the combustion chamber. The external drag of the engine must be carefully considered along with the specific impulse. The specific impulse tends to be less than 100 sec. and shows a maximum at a specific flight speed. The range of specific fuel consumption is from 2-3 lb/hr-lb. thrust for best operating conditions.

The nuclear ramjet supplies energy to the air flowing through the duct by heat exchange from a nuclear reactor. Because of the minimum critical mass of the reactor, heat exchanger and materials problems, this engine system is limited to large size vehicles. Its primary interest is generated from the potentially large supply of energy available from the nuclear fission reaction, one gram of fissioned Uranium-235 producing one megawatt-day of energy.

Recent work has been directed toward the super- and hyper-sonic ramjets in which combustion at supersonic velocities external to aerodynamically shaped bodies which slow down the air flow and then expand the combustion products without benefit of a conventional nozzle. Aero-thermal heating, supérsonic combustion processes and suitable fuels are primary problems in this type of engine system.

The pulse-jet engine depends upon the ram-effect to open flapper-type shutters and admit air to the combustion chamber. When fuel is injected and combustion takes place, the increased pressures from the combustion and the pressure surges in the duct close the shutters and the combustion products are expelled from the nozzle to generate thrust. When the combustion chamber pressure is reduced to a low value, the shutters re-open and the cycle is repeated. The pulsed thrust results in a low-efficiency system and the resonant frequency of the duct strongly influences the performance, (Ref. 20).

The turbo-jet engine includes a gas compressor system between the inlet and the combustor. Power is extracted from the combustion products by a turbine which drives the compressor. The residual power of the combustion products is used to generate thrust. The detailed design and arrangements of the components of this engine system are as varied, if not more so, than the reciprocating engines. References 2, 4, 9, 19, 20, present extensive descriptions and detailed design data.

Specifications of currently available power plants from Ref. 17 are summarized in Table V.

The specific impulse of these engine systems range from 50-155 sec. This is deliberately lowered to obtain higher efficiency and thrust at low flight speed by air augmentation, for example, the fan-jet engine in which part of the inlet air is ducted around the combustion system to mix with the effluent combustion products before being expanded through the nozzle, or by fluid injection, for example, alcohol, water or fuel, into the nozzle.

The effect of the specific weight of these powerplant systems on vehicle performance is illustrated by Fig. 9 from Ref. 19.

For long range aircraft, however, this fraction is very large and the engine characteristics have a decided effect on the range and pay load. Fig. 1 shows the all-up weight of an aircraft to carry a given pay load plotted against range. Two curves are shown for different engines both of which have the same fuel consumption, but one of which has a higher weight. It will be seen that at short range the effect on aircraft size is not large, but at high range the effect becomes catastrophic.

One of the objects of this paper is to stress the great importance of engine weight as well as fuel consumption in the development of turbine engines for long range aircraft.

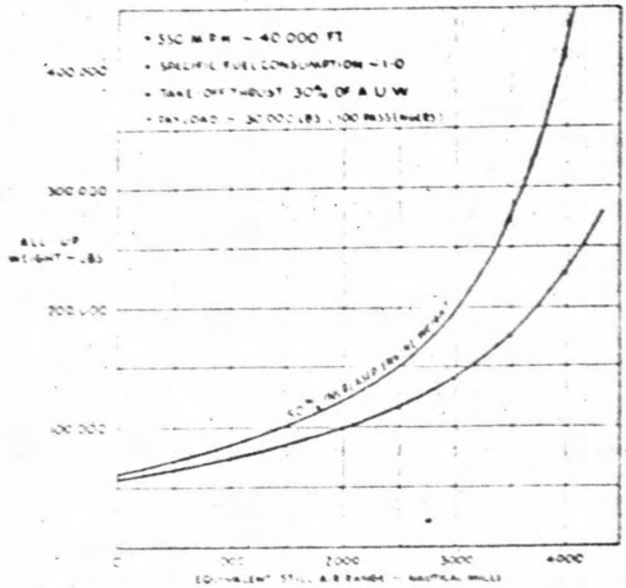


Fig. 1. Effect on aircraft weight of change in engine specific weight (Ref. 11).

Progress to Date

As a background to the subject let us look at the improvements in fuel consumption that have been made in the past 10 years (See Fig. 2.) Unfortunately figures for flight conditions are not generally available. This fact provides a suitable excuse to draw upon Rolls-Royce data to illustrate the progress made in reducing fuel consumption at representative flight conditions. Generally large improvements follow the introduction of a new type of engine, particularly in the type of compressor. Some progress of a lesser order has of course been made by the separate types, but this is not shown on the slide.

Progress made in reducing fuel consumption of Rolls-Royce propeller turbines at representative flight conditions is shown in Fig. 3; the dates relate to their first flight. The large improvement in performance between the Dart and axial types is to some extent due to the use of more efficient and higher pressure ratio compressors, but more because of the basic increase in size which makes such efficiencies possible.

Rocket Engine Systems

The rocket engine systems generate thrust by ejecting a high energy jet from a closed thrust chamber in which a suitable oxidizer and fuel are reacted to release energy in the form of heat (6,12,9,11). Basically, no different in concept from the air-breathing jet engines, for many applications they gain in flexibility from the use of many different propellant combinations, and the ability to operate in FREE space because they carry their own oxidizer. Propulsion is dependent upon internal conditions alone and not the effect of air flow through the system.

The thrust of the rocket engines depend upon the total propellant flow rate, $\dot{W}$, the exhaust velocity, C, the area of the exit plane of the nozzle, A_e, the pressure at the exit plane of the nozzle, p_e, and the ambient atmospheric pressure, p_o.

$$F = \frac{\dot{W}}{g} C + (p_e - p_o)A_e \qquad (42)$$

The exhaust velocity C, depends upon the effectiveness of the combustion process described by the characteristic velocity, C^*, and the behaviors of the nozzle under differing ambient conditions, described by thrust coefficient, C_f.

$$C = C^* C_f = \frac{I_s}{g}, \text{ ft/sec.} \qquad (43)$$

The characteristic velocity of the thrust chamber is established from the thrust chamber pressure, P_c, the throat area, A_t, and the propellant flow rate, $\dot{W}$.

$$C^* = \frac{P_c A_t g}{\dot{W}} = \frac{g I_s}{C_f}, \text{ ft/sec.}$$

$$= \frac{\left(g \gamma R \frac{T_c}{M}\right)^{\frac{1}{2}}}{\gamma \left(\frac{2}{\gamma + 1}\right)^{\frac{\gamma+1}{2(\gamma-1)}}} \qquad (44)$$

γ = specific heat ratio of combustion products, C_p/C_v.

T_c = absolute temperature of combustion products, $^{\circ}R$.

R = gas constant

M = average molecular weight of combustion products, lb/mol. volume

A_c = area of nozzle thrust, ft^2

The thrust coefficient depends upon the pressure ratio across the nozzle, nozzle dimensions and combustion product physical properties.

$$C_f = \frac{F}{P_c A_t}$$

$$= \left[\frac{2\gamma^2}{\gamma - 1}\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}}\left(1 - \left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right)\right]^{\frac{1}{2}} + \frac{P_e - P_o}{P_c}\,\frac{A_e}{A_t} \qquad (45)$$

A_e = area of ixit section of nozzle, ft^2

The variation of the thrust coefficient with pressure ratio, area ratio and specific heat ratio is summarized in Fig. 10.

These formulae are based upon the behaviour of ideal gases and the assumption that the chemical equilibrium does not change at a specific operating mixture ratio. Good design practice permits experimentally determined C^* values within 90-95 percent pf these computed values, and experimental values of C_F approach with 95-98 percent of theoretical values.

The thrust coefficient, C_F, is a function of altitude. As a result the area ratio, Ae/At, for booster engines is less than for upper-stage engines operating at high altitudes or in FREE space.

Both the characteristic velocity of the thrust chamber, C^*, and the thrust coefficient, C_F, are functions of the mixture ratio of the propellants, maximum performance occurring at slightly fuel rich operation conditions. Therefore the performance of thepropellants is specified not only by the combination, but the weight ratio of oxidizer to fuel.

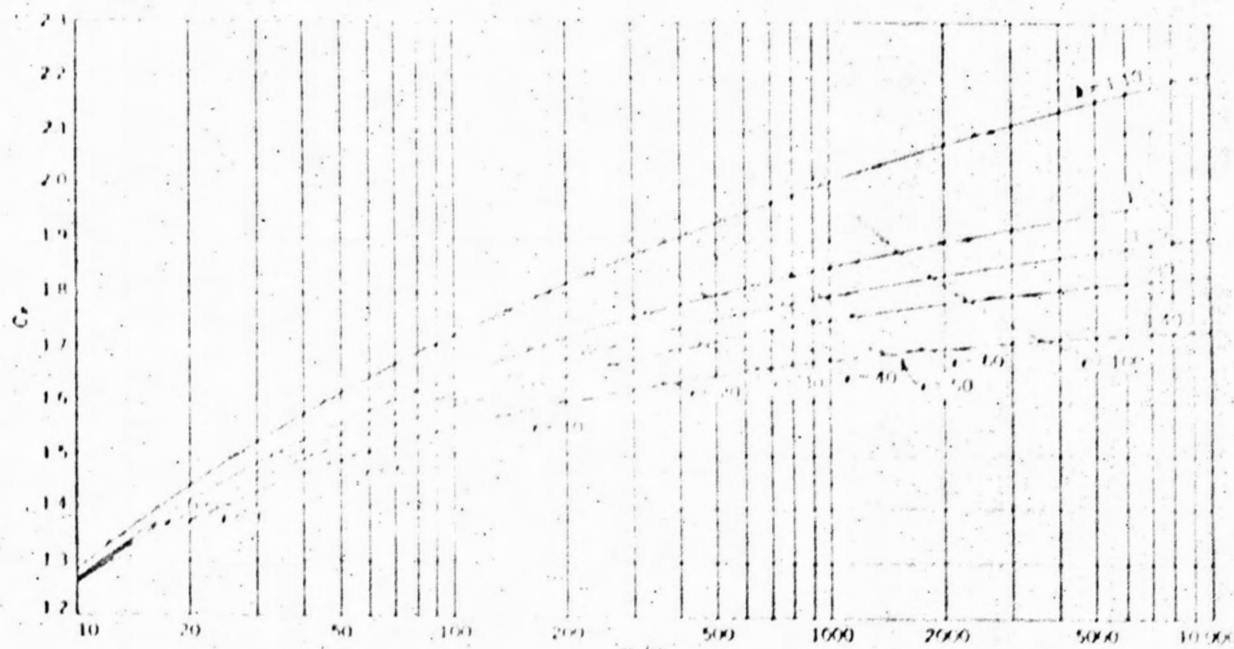
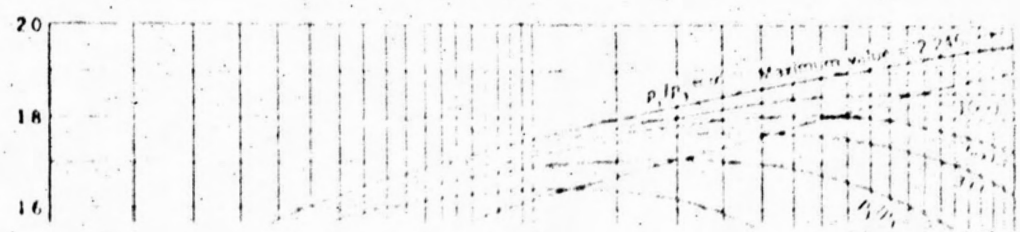


Figure 10 Thrust coefficient C_F as a function of pressure ratio, area ratio, and specific heat ratio for optimum expansion conditions



A high performance propellant combination does not always result in the most compact system which includes the propellant and their tanks, the propellant feed system and the thrust chamber. A measure of the compactness of the engine system is the density impulse parameter, I_{sd}, which is defined by the relation

$$I_{sd} = I_s \delta_p \qquad (46)$$

in which the propellant bulk specific gravity is defined by the mixture ratio, r, and the oxidizer density, δ_o, and the fuel density, δ_f.

$$\delta_p = \frac{(1 + r)\,\delta_f\,\delta_o}{\delta_f\,\delta_o} \qquad (47)$$

The chamber pressure, P_c, also influences the weight and overall dimensions of the rocket engine system. As the chamber pressure exceeds a few hundred p.s.i., the overall size and weight of a given thrust level engine system diminishes to a minimum then increases because of heavier structures required to withstand the higher stresses induced by increasing chamber pressures. The maximum size and weight and the optimum operating chamber pressure must be determined by a specific analysis of each engine system.

The total impulse, I_t, defined by the relation,

$$I_t = \int_0^{t_b} F\,dt = Ft_b = I_s W_p \qquad (48)$$

is a parameter associated particularly with solid rocket engine systems. The total impulse can be accurately determined by test from thrust, F, burning time, tr, and burned propellant weight, W_p, from which the effective specific impulse can be computed.

The equivalent power in the jet is

$$HP_{jet} = \frac{FC}{550} \text{ H.P.} \qquad (49)$$

and the propulsive efficiency is the same as that of the air breathing thermal jet and propeller driven systems.

$$\eta_p = \frac{2\,V/C}{1 + (\frac{V}{C})^2} = \frac{\frac{2\,g\,V}{I_s}}{1 + (\frac{g\,V}{I_s})^2} \qquad (50)$$

The liquid propellant engine system includes the tanks to store the propellants, propellant feed and control system, a power source to drive the feed system and the thrust chamber.

Major differences are in the propellant feed system. Smaller engines are often pressure fed andcontrol is obtainable by balancing thepressure drops in the feed lines. As the thrust level increases the pressure fed system rapidly increases in weight and the pump fed system, turbine driven from a gas generator or bleed from the thrust chamber, is the favored type. (Fig. 11). Bi-propellant engine systems are by far the most widely used because of the relstive ease of handling and controlling the propellants. While the monopropellant systems are simpler having only a single feed system and no balancing is needed, the propellants, for example hydrazine, hydrogen peroxide and nitro[illegible], are sensitive to contaminants, heat and impact.

The theoretical performance of a number of bi-propellants are summarized in Table VI and of several monopropellants in Table VII. A well designed engine system can achieve a performance in the neighborhood of 95% of the theoretical values and this depends upon the specific design. The theoretical performance of several monopropellants are summarized in Table VII. Other propellants are summarized in Ref. 14.

Currently, some attention is being given to hybrid engines which use a liquid oxidizer and solid fuel, or vice-versa. No applications to vehicle systems have been made.

The advantages of the liquid propellant system includes light weight, capability of throttling, relative ease of handling the propellants for large systems, and the availability and low cost of a wide variety of propellant combinations.

Fig 11 LIQUID PROPELLANT ENGINE

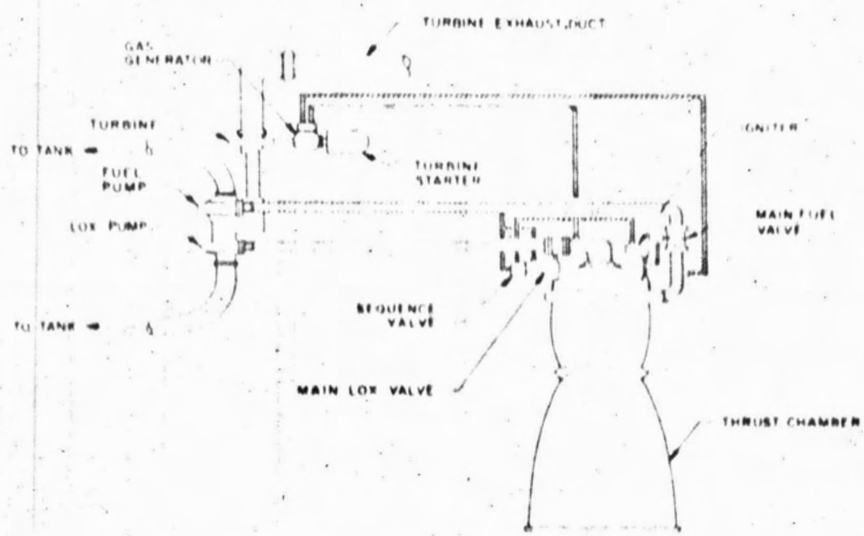


Fig 1X ION ENGINE

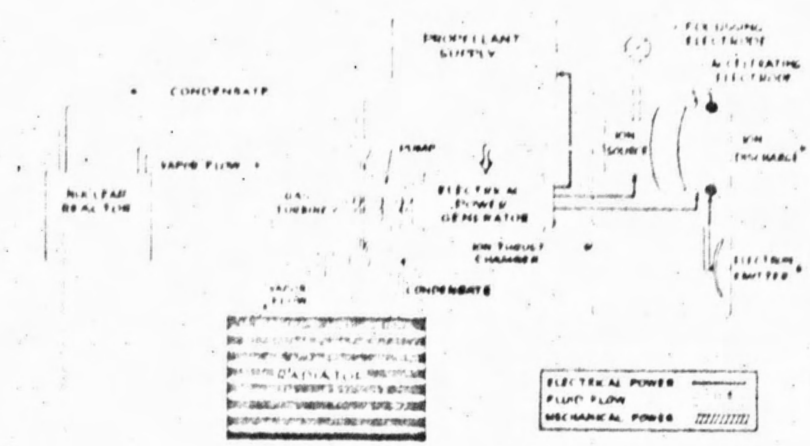
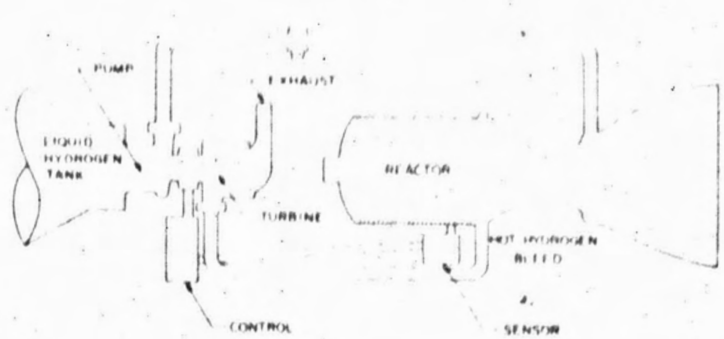


Fig 12. NUCLEAR ENGINE



(62)

TABLE VII

MONOPROPELLANTS

Name	Formula	P_c=Isp @300 psi	Specific Gravity	Isp.d.
Ethylene Oxide	C_2H_4O	166	0.887	147
Hydrazine	N_2H_4	174	1.0045	175
Nitromethane	CH_3NO_2	218	1.13	246
N-propyl nitrate	$CH_3(CH_2)_2NO_3$	179	0.935	167
90% Hydrogen Peroxide	H_2O_2	133	1.387	184

Currently available liquid propellant rocket engines range from a few pounds thrust up to 1,500,000 lb. thrust, for example the Rocketdyne F-1 engines which burn liquid oxygen and kerosine (JP-4). The Saturn launch vehicle uses an engine system consisting of a cluster of eight Rocketdyne H-1 engines of 188,000 lbs. each.

The nuclear rocket engines (Fig. 12) are liquid monopropellant systems, the energy of the nuclear reactor utilizing heat exchange processes to transfer the heat energy to the monopropellant, then expanding it through the nozzle to generate jet power. The propellants used are preferably of low molecular weight to obtain high specific impulse at low operating temperatures. TYPICAL propellants are hydrogen, H_2, or ammonia, NH_3. Currently in the early development phases, specific impulses of 600-900 sec. are anticipated. The engines will be relatively large because of critical mass requirements of the reactor. The specific weight of the engine system is expected to be about a magnitude larger than that of the best chemical rocket engines, partially offsetting the advantage of high specific impulse.

A solid propellant rocket engine system is an integrated assembly including an igniter, the engine case, the solid propellant grain, and

(Ref 11).

the nozzle. There is no separate combustion chamber as in the liquid propellant system, The propellant storage tank and the combustion chamber are combined in the engine case. After ignition, control of the propellant mass flow rate is accomplished by precise control of the exposed surface area of the solid propellant. The re-acting grain generates combustion products at the exposed surface which regresses normal to itself in parallel layers. The rate of regression is defined as the burning rate of the propellant. The burning rate of the grain must be matched to the flow rate of the nozzle to maintain a chamber pressure high enough to give stable reaction yet produce the combustion product flow at a rate high enough to produce the required thrust level.

A solid propellant is a homogenous material, or a mixture of materials, which when ignited, reacts to evolves combustion products continuously at elevated temperatures without dependence on the atmosphere. (Ref 21) A grain, encased in the rocket chamber, must hold its shape over an extended temperature range and must withstand the mechanical stresses resulting from handling, igniting and firing the rocket. To meet these requirements a special class of plastic materials have been developed which are in themselves combinations of oxidizing and reducing agents or in the case of mixtures the fuel has the necessary plastic qualities. ~~have been developed.~~

The principal homogeneous propellant is nitro-cellulose which is plasticized with nitroglycerin and other organic materials (24,25). The composite propellants include the oxidizers, potassium perchlorate, ammonium nitrate, ammonium picrate, etc. The fuels include the asphalts, rubber, plastic materials such as the poly iso butylenes, vinyl polyesters, polybutadienes, polysulfides, and the metals, aluminum and beryllium. The particular combination used depends upon the application and its requirements.

The size of solid propellant engine systems range from small units such as used in the Bazooka weapon to large units which are of 154 in. and 160 in. diameter, currently in development. Maximum thrusts currently being achieved are in the million pound class.

ELECTRICAL SPACE PROPULSION

Concepts of electrical powerplant systems are varied, both from the system standpoint, and in the use of components. The electrical rocket engine has three important subsystems: the energy source, the power convertor, and the electrical thrust devices (Fig. 3). The engine system parameters for overall performance of the electrical propulsion systems are summarized in Table VIII.

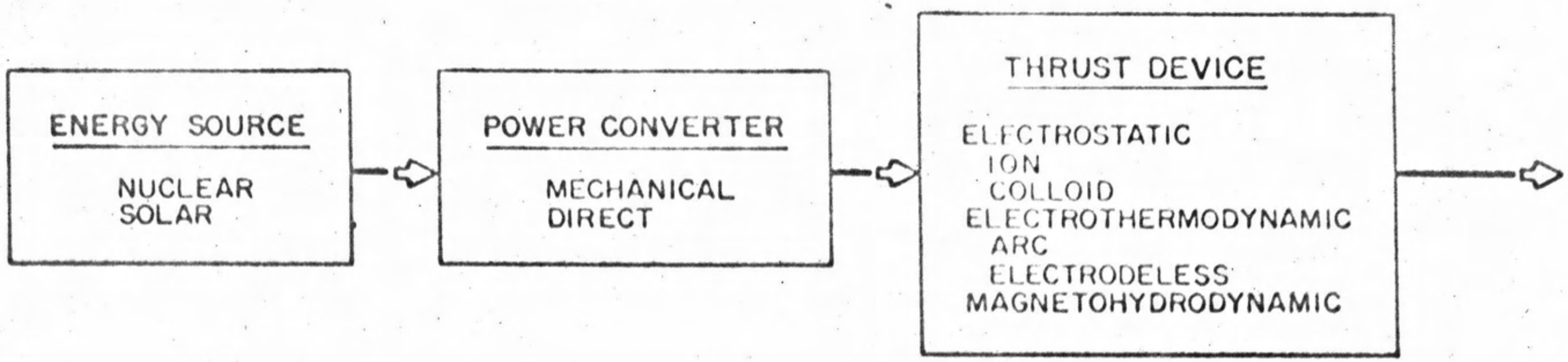


Figure 13. Basic Subsystems of Electrical Rocket Engine

TABLE VIII

SUMMARY OF PERFORMANCE PARAMETERS,

ELECTRICAL ROCKET ENGINES

Parameter	English System	CGS System
Specific Power	$\frac{P}{W} = \frac{21.8\ G I_s}{\eta}$, watts/lb	$\frac{P_E}{W} = \frac{4.91 \times 10^{-5}\ A\ I_s}{\eta}$, watts/gm
	$\frac{P}{W} = \frac{0.0293 G I_s}{\eta}$, hp/lb	
Power per Unit Thrust	$\frac{P}{F} = \frac{21.8\ I_s}{\eta}$, watts/lb	$\frac{P}{F} = \frac{4.91 \times 10^{-5}\ I_s}{\eta}$, watts/dyne
	$\frac{P}{F} = \frac{0.0293\ I_s}{\eta}$, hp/lb	
Exhaust Velocity	C ft/sec	C cm/sec
Specific Fuel Consumption	$w_f = \frac{1}{3600\ I_s}$ hr^{-1}	$w_f = \frac{1}{3.53 \times 10^6\ I_s}$, hr^{-1}
Specific Impulse	$I_s = \frac{C}{32.2}$ sec	$I_s = \frac{C}{980}$ sec
Engine Operating Time	$t = \frac{1-\exp\left(\frac{-\Delta v}{32.2 I_s}\right)}{G} \cdot I_s$ sec	$t = \frac{1-\exp\left(\frac{-\Delta v}{980\ I_s}\right)}{G} \cdot I_s$ sec

The key factor in the successful development of an electrical rocket engine is the energy source. Chemical sources known today produce specific energy too low to be used effectively. Solar energy, while it is relatively limitless in amount, is limited in rate, and the rate varies with distance from the Sun. Nuclear energy, however, can produce the large amounts of energy at the rate required for extended space missions. It can be applied using present techniques, and has future potential which will be further augmented by improvements in power conversion systems.

Power conversion can be accomplished mechanically, thermoelectrically, thermionically, and by the application of magnetohydrodynamic principles. Mechanical methods appear promising for early application in electrical power plant systems. Other methods mentioned have not reached the technical perfection of mechanical systems, but steady progress is being made in many areas. As a result, direct conversion devices combined with an electrical thrust chamber may make early contributions to the accomplishment of continuously powered and controlled space flight.

Electrical thrust chambers have a variety of forms. The electro-thermodynamic thrust chamber, variously described as the arc or plasma jet, uses electrical energy for heating the propellant gas, which in turn is expanded through a de Laval nozzle to generate thrust in the same manner as a chemical rocket. Other devices generate a plasma which may be accelerated by external, or induced, electromagnetic fields. Ions can be extracted from the plasma, or formed directly by surface contact phenomena and then accelerated electrostatically. This is the ion rocket engine. Substitution of charged colloid particles for ions offers the potential advantages of low specific impulse to the electrostatic ion rocket. The concepts embodied in the electrical thrust chambers are as varied as those in the power conversion systems.

ENERGY SOURCES

The energy sources for electrical propulsion include conventional chemical, radio-isotope, the nuclear fission reactor, the fusion reactor, and the Sun. If the demands of electrical space propulsion are to be met, the specific power of the vehicle must be in the neighborhood of one-tenth horsepower per pound to achieve a thrust-to-weight ratio of 10^{-4} (Fig. 1) if the specific impulse is 10,000 sec. Since a Moon mission requires a characteristic velocity of approximately 24,000 sec, this implies an operating time of 1700-1800 hr. Similarly a Mars mission which requires a characteristic velocity approaching 100,000 sec demands nearly 9,000 hr of engine operation. These data serve for an over-all comparison to determine the capabilities of the different power supply systems.

Table IX is a summary of energy sources and power conversion systems considered for electrical powerplant systems. Comparison of these data with the requirements presented above eliminate all systems except the nuclear and solar power supplies.

Among the chemical systems, batteries are reliable and inexpensive. These advantages are offset by low power, heavy weight and short life. The turbines driven by a gas generator give a good power yield for short times, but the systems are heavy, have too short a life, and require too much fuel. The fuel cells apparently combine many of the disadvantages of the chemical systems in addition to having low efficiency.

The radio-isotope sources have the advantage of being direct convertors, and the thermoelectric type is [illegible] reliable. However, the specific power is low, ranging from 0.0003 to 0.0001 kw/lb. The lifetime is limited by the half-life of the radio-isotope.

TABLE - IX

SUMMARY OF ELECTRICAL POWER SUPPLY SYSTEMS

Type	Specific Weight	Operating Life	Status
Chemical			
Batteries	20 lb/kwh	50-200 days	Production
Gas Turbine	7-10 lb/kwh	0.5 - 5 hr	Production
Fuel Cell	12 lb/kwh	30 days	Development
Radio-Isotope	1000-3000 lb/kw	280 days	Development
Solar			
Photovoltaic	50-200 lb/kw	Indefinite	Development
Gas Turbine	10-20 lb/kw	Unknown	Research
Thermoelectric	100 lb/kw	Unknown	Research
Nuclear			
Turbine	8-100 lb/kw	365 days	Development
Thermoelectric	100 lb/kw	Unknown	Research
Thermionic	Unknown	Unknown	Research
Fusion	Unknown	Unknown	Research

Approximate.. 218 kw of electrical power are quired for a 1-lb- thrust electrical engine operating at a specific impulse of 10,000 sec. and 100 percent efficiency. From the data of Fig. 14, a nuclear turbo-electric powerplant

... lb., approximately

8 lb. per kw. In contrast, currently available photovoltaic cells weigh about 72 lb. per kw. These rough data indicate that, assuming collector structural weights to be the same, a 1-lb photovoltaic engine has a weight penalty of 14,000 lb. This factor is offset by a 700 sq. ft. radiator area, increased mechanical complexity and possibly lesser reliability of the nuclear turbo-electric system.

The estimated performance of mechanical power conversion systems using different working fluids and operating cycles are summarized in Table X. p. 23

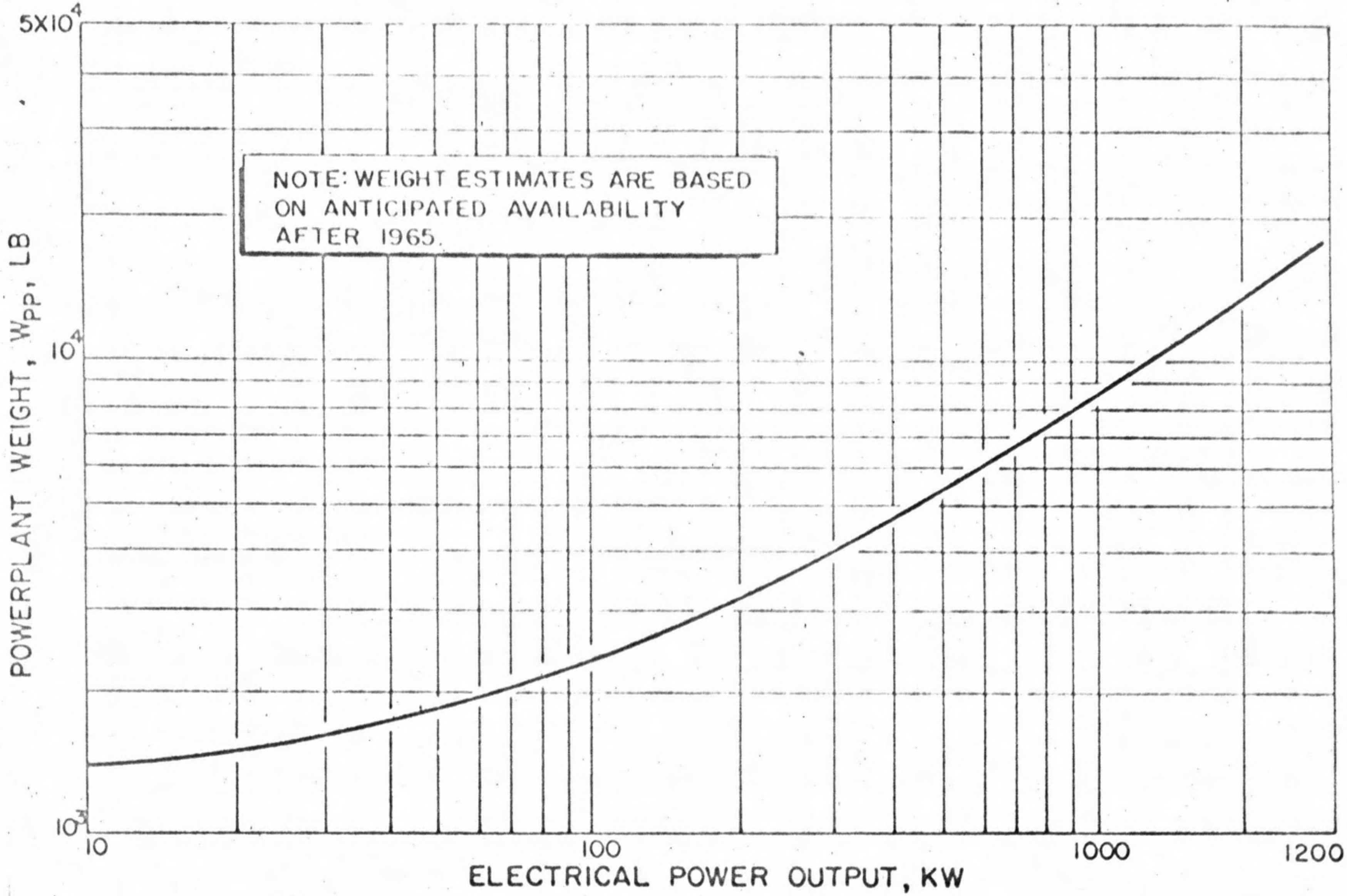


Figure 14. Nuclear Turboelectric Powerplant Weight Including 1000 lb Shield (1, 24).

WORKING FLUID	CYCLE	TURBINE INLET PRESS (PSIA)	REACTOR OUTLET TEMP (°F)	RADIATOR* AREA SQ FT	REACTOR* POWER MW
HELIUM (WORKING FLUID & COOLANT)	BRAYTON	1500	1900	8800	4.00
		1500	1500	12300	5.26
HELIUM (SODIUM COOLANT)	BRAYTON	1500	1460	21400	6.25
MERCURY (SODIUM COOLANT)	RANKINE	5000	2000	1100	4.00
		2500	1800	1700	4.16
		1200	1600	2000	5.26
		500	1400	3200	5.26
ALKALI METAL (WORKING FLUID & COOLANT)	RANKINE	460	2000	1000	3.57
		217	1800	1300	3.57
		89	1600	1500	4.16
		31	1400	3400	7.14

* FOR ONE MW MECHANICAL POWER OUTPUT

OPERATING REQUIREMENTS FOR

MECHANICAL POWER CONVERSION SYSTEMS

Table X.1

The choice between a solar-electric or a nuclear turbo-electric system will probably be based upon the economics of placing a large area, heavy system of possibly higher reliability in orbit in contrast to the costs of the nuclear turbo-electric system. ~~Such a selection is beyond the scope of this discussion.~~

The over-all picture of a low-powered system may be clarified by considering the Snap II reactor, developed by Atomics International Division of North American Aviation. This reactor, recently described in the press, is quoted to have a power output of 3 kw and a weight of 220 lb. An electrical rocket using this power source would be capable of producing not more than 0.014 lb of thrust at a specific impulse of 10,000 sec. Arbitrarily allowing 60 lb. for propellant and thrust device, the maximum thrust-to-weight ratio is 5×10^{-5}. A 3 kw photovoltaic solar rocket would weigh approximately 360 lb. and have a minimum area exposed to the Sun of 25 sq ft, if operated in cislunar space. The thrust-to-weight ratio would be approximately 2.5×10^{-5}. Because of the possibility of reducing the photovoltaic cell weights, the solar electric and the nuclear electric systems are competitive at this power level. At lower power levels, the solar electric rocket may have an advantage.

application in an electrical rocket engine system. Because of the anticipated power levels and the energy levels of the particles produced, direct use of the particles to generate reactive thrust appears to be a more logical application than converting energy to electrical power and then accelerating the same, or other particles. There are so many unknown factors in this new development that the most that can be recommended at this time is to watch the progress of the basic research programs and, if possible, devise a sound approach to a propulsion system (Ref. ~~23~~ 22).

~~The use of solar energy is attractive for small electrical propulsion systems and for applications in vehicles where size and a small thrust-to-weight ratio are not serious limitations. Recent performance of solar cells in supplying satellite electrical power for extended periods enhances the solar cell's attractiveness for propulsion application. On the basis of currently available photovoltaic cells with an efficiency approaching 11 percent, and the generalized analysis presented so far, the collector area for a 1-lb-thrust engine having a specific impulse of 10,000 sec is approximately 16,000 sq. ft. within the earth's orbit. If constant thrust is maintained, this area increases to 36,000 ft. in the Mars orbit.~~

ELECTRICAL THRUST DEVICES

The electrical thrust devices are dependent upon the formation of a plasma, which is an electrically neutral gaseous mixture of electrons, ions, and neutral particles.

ELECTROTHERMODYNAMIC

The electrothermodynamic thrust device, or plasma jet, is characterized by high-temperature gas, high-temperature electrons and high pressures (Fig 25). Several techniques are used to inject propellant into the thrust device, Fig 17. The propellant is heated by a low voltage, high current arc, then expelled through the nozzle.

). Table X summarized gas temperatures measured experimentally within high-pressure arcs, and indicates operating temperatures for a number of promising propellants. At high specific impulses, the existence of these high temperatures, the relatively high pressures in the arc chamber, and the low propellant flowrates induce severe convective heat transfer; therefore, major material problems can be

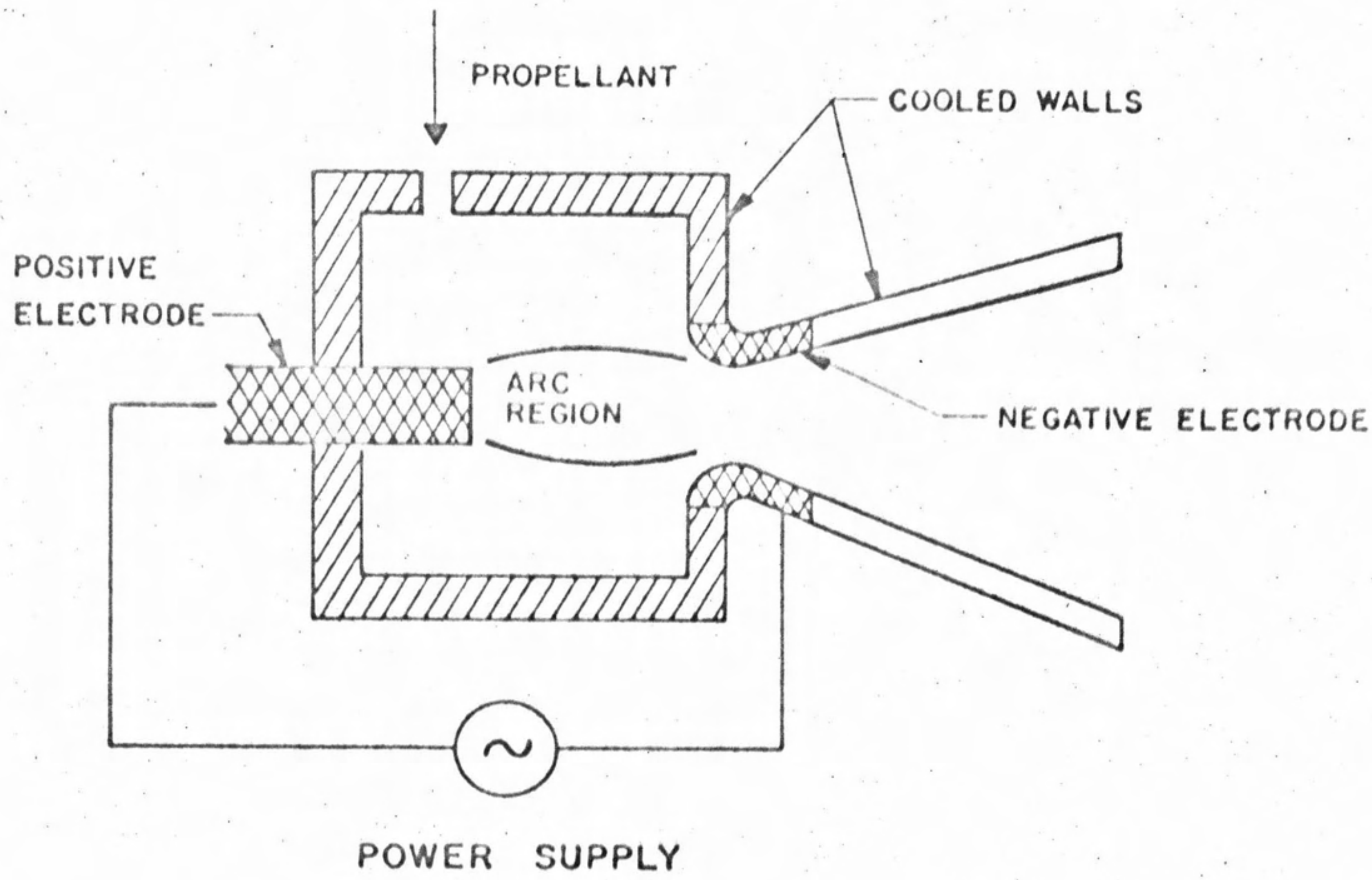


Figure 15. Electrothermodynamic Thrust Device

TABLE XI

MEASURED TEMPERATURES AT THE AXIS OF POSITIVE

COLUMN OF ARCS OPERATING AT

ONE ATMOSPHERE PRESSURE

Gas	Electrode	Arc Length, cm	Current, Amp	Temperature, deg K
Air	Carbon	0.8	2	5,900
Air	Carbon	0.3 - 1.8	1 -12	6,200 - 7,800
Air	C - Ca	Few	180	12,000
H_2	Tungsten		4	4,900
H_2	Tungsten		8 - 9	5,300 - 6,300
H_2	Carbon		Few	5,000
N_2	Copper	15	2	5,300
Air	Copper	15	2	4,700
Air	Copper		14	5,470
Air	Copper		10	6,100
H_2O		(Jet)	500	35,000
H_2O		(Jet)	1,500	50,000

expected with this device which will limit the maximum specific impulse.

The lower limit for efficient use of this device is established by the power required to ionize and heat the large flow of propellants, which increases inversely with the specific impulse. For specific impulses less than 1000 the reported results are consistently less than 20 percent power efficiency. Because of the relatively slow change of payload with increasing specific impulse (see Fig. 2), and the improved efficiency of the electrothermodynamic device, operation below specific impulses of 1000 to 1200 sec does not appear warranted.

ELECTROMAGNETIC THRUST DEVICES

The acceleration of a plasma by electromagnetic fields, has the objective of generating particle motion in an ordered linear flow with the minimum expenditure of energy. This objective has not been achieved to date and is reflected in low power efficiencies of the engine systems.

Two forces act on a charged particle of mass M and charge e when it moves through a region in which an electric field E and a magnetic field H exist. (Ref 26) If e is the charge on the particle, the product eE is the electrostatic force parallel to the field. The magnetic force is at right angles to the particle velocity, v, and to the magnetic field, H, and is the vectorial cross product of v and H.

$$M \frac{dv}{dt} = e\,(E + v \times H).$$

When H vanishes, the charged particles move with constant acceleration, $\frac{e}{m}$ E, as in the electrostatic ion engine. When there is no electric field, the acceleration, $\frac{e}{m}(v \times H)$, is perpendicular to the particle velocity, curving its trajectory but not changing the scalar velocity, v.

The motion of a discrete particle is complicated in a macroscopic situation such as exists in the discharge of a high-intensity arc. When theelectric and magnetic field distribution, the characteristics of the plasma, and transient discharges are considered, the magnitude of fully understanding the phenomena emphasizes why the magnetohyirodynamic systems are still in an early stage of development.

Button Plasma Guns

In button-type plasma guns (Fig. 1, Ref. 25, p. 105) the discharge of a capacitor bank across the exposed ends of the conductors produces a plasma of metallic ions and electrons. The high discharge currents induce magnetic fields which project small plasma clouds at speeds up to 6.7×10^5 ft/sec. Figure 16c is a variation in which plasma can be injected into the gun from an external source.

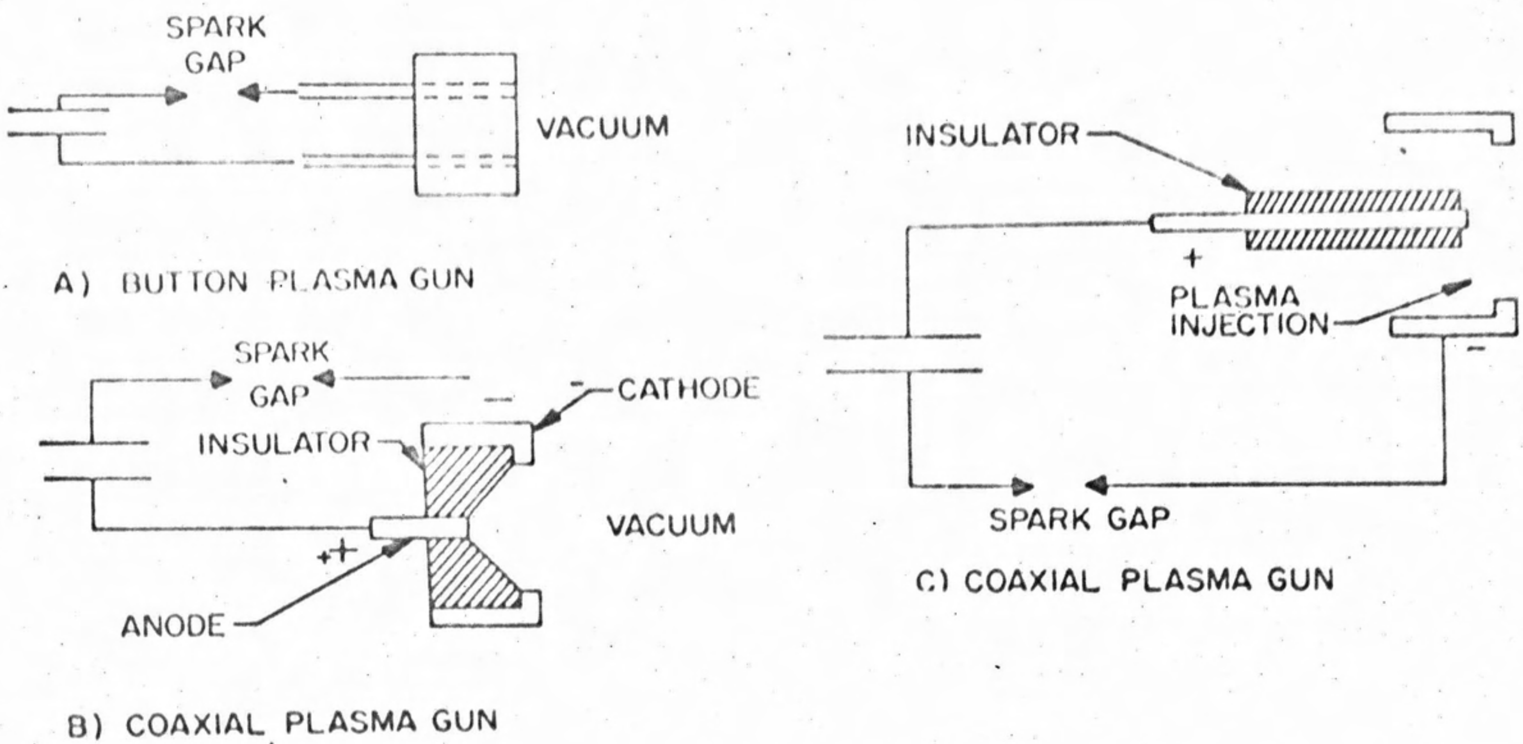


Figure 16. Button Plasma Guns

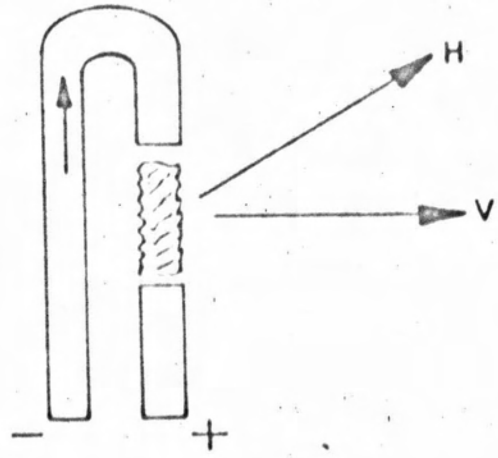
Major problems associated with these units are small size, intermittent operation, coupling of the energy-storage system to obtain efficient energy transfer and a wide-angle plasma pattern, which results in a loss of thrust.

Crossed-Field Motors

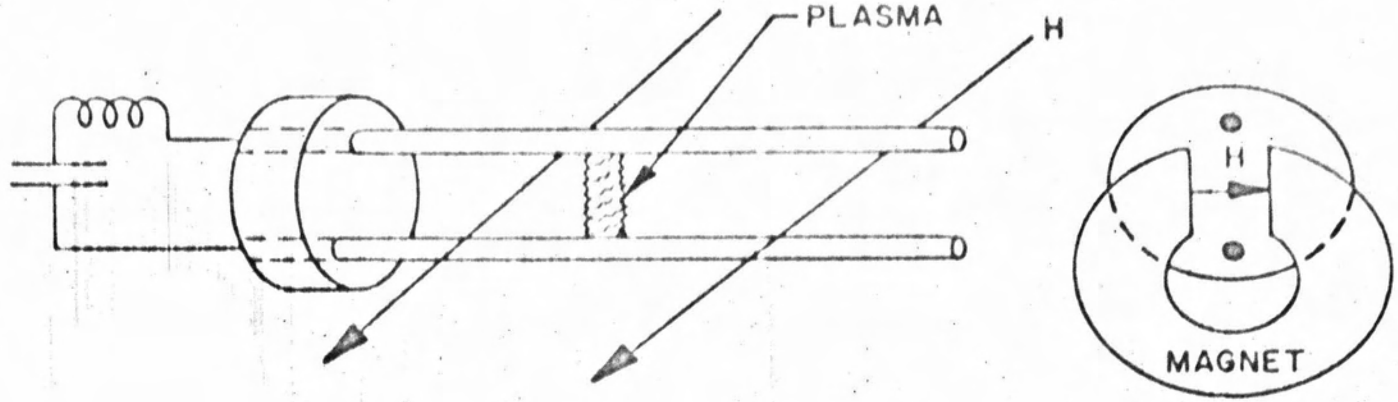
The rail-type motor, Fig. 17a, depends upon the initiation of a discharge between the two rails. Once the discharge is started, the current in the system induces a magnetic field to drive the plasma along the rails to the ends. When no external magnetic field is applied perpendicular to the plane of the rails, the rail motor is analagous to a series-wound motor. With an external magnetic field, the system is analogous to a series shunt-wound motor. Substitution of flat plates for cylindrical rails provides better focussing of the plasma (Ref. 25). Plasma velocities up to 3.3×10^6 ft/sec have been obtained.

The rail-type motor presents less energy-source coupling difficulties than the button guns. Operation is necessarily intermittent, the flow pattern is wide-angled, and the thrust limited.

86.



B) T-STRAP MOTOR



A) RAIL-TYPE MOTOR

Figure 17a. Crossed-Field Motors

The Kolb T-strap, motor (Fig. 17b) also applies the principles of crossed-field acceleration. The electric field across the arc gap and the induced magnetic field of the cathode lead, which is brought close to the arc gap, are at right angles. The plasma velocity is away from the cathode lead. The plasma generated includes metallic ions from the electrodes and electrons.

In both the rail-type and Kolb guns, plasmas have been successfully produced by exploding wires across the arc gaps. Using 1 mil tungsten wire, specific impulses of 2200 sec at efficiences of 39 percent have been obtained (Ref. 27).

Traveling-Field Accelerators

Thrust can be generated by the interaction of a plasma ring and a moving magnetic field. The ring corresponds to the rotor bar of an induction motor and is forced to move by the moving field. A schematic of such a system is shown in Fig. 18. The ionized gas leaves the plasma source and enters a cylindrical coil whose windings are arranged to produce an alternating radial field moving in the direction of flow. The coil arrangement moves the field with increasing velocity corresponding to a

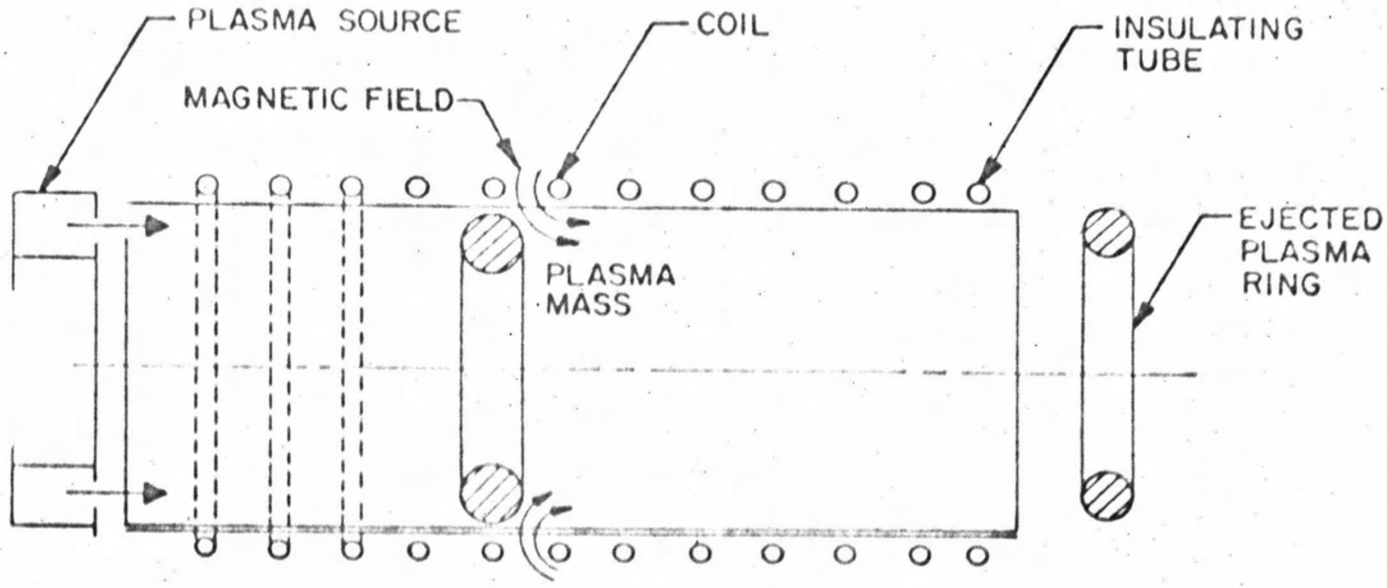


Figure 15 Traveling Magnetic Field Accelerator

constant rate of acceleration. The initial field velocity is somewhat higher than the entering plasma. The final velocity is determined by the number of coils.

The force to produce the constant acceleration of the plasma is from the interaction of the magnetic field and the induced current circulating around the ring through the conducting plasma. Because the ring is moving with a velocity slightly less than the field, the current is maintained. The current tends to maintain a high degree of ionization in the plasma and a stable configuration from pinch effect. These effects are sustained only if the acceleration is high. Thus a high specific impulse is expected to result.

Current estimates of the obtainable specific impulse are from 3,000 sec to 15,000 sec (see Ref. 28). Experimental values up to 25,000 sec have also been obtained (Ref. 29). The estimated efficiences of this type of device is 85 percent. This figure does not include the power required to generate the plasma which is accelerated.

ION ROCKET ENGINE

The ion rocket engine is a system including three major subsystems; the energy source, the power convertor, and the ion thrust chamber, Fig. 11. The thrust chamber lies within the rectangle enclosed by the dashed lines. This thrust chamber (Ref. 21) includes an ion source, an array of electrodes to focus and accelerate them, and an electron emitter.

Vaporized propellants are fed to the ion source, where they are ionized either by contact with heated metals of high work function, or by being subjected to electron bombardment within electromagnetic fields. The latter method of ionization is commonly described as the arc source.

The ionized propellant is directed from the ion source through focusing electrodes and accelerating electrodes, which impart kinetic energy to the ions. This results in the required changes of momentum of the particles to develop thrust for the vehicle. The major performance parameters of the ion engine (Ref. 28) depend upon the rates of the accelerating potential, V, volts, and the ion mass, A, the atomic or molecular weight in grams. These parameters (Ref. 28) are summarized in Table 12.

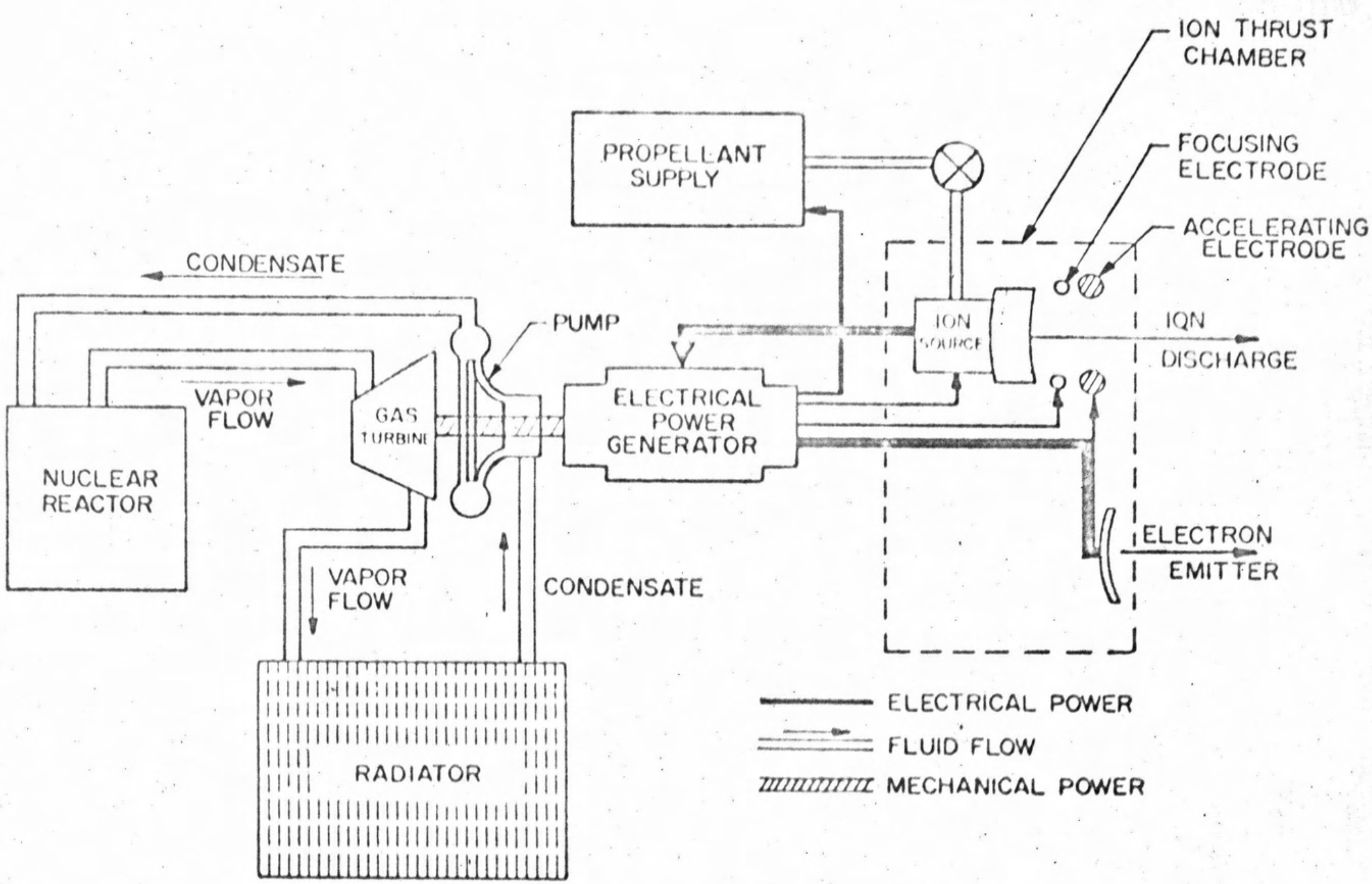


Figure 19. Typical Ion Propulsion System

912

TABLE 12

SUMMARY OF PERFORMANCE PARAMETERS, ELECTRICAL ROCKET ENGINES

	Definition	
	English System	Metric System
Specific Power	$\frac{P}{M} = 3.089 \times 10^4 G \left(\frac{V}{A}\right)^{1/2}$ watts/lb	$\frac{P}{M} = 0.0695\ a \left(\frac{V}{A}\right)^{1/2}$ watts/gm
	$\frac{P}{M_g} = 41.42\ G \left(\frac{V}{A}\right)^{1/2}$ hp/lb	
Power per Unit Thrust	$\frac{P}{F} = 3.089 \times 10^4 \left(\frac{V}{A}\right)^{1/2}$ watts/lb	$\frac{P}{F} = 0.0695 \left(\frac{V}{A}\right)^{1/2}$ watts/dyne
	$\frac{P}{F} = 41.42 \left(\frac{V}{A}\right)^{1/2}$ hp/lb	
Average Exhaust Velocity	$C = 4.557 \times 10^4 \left(\frac{V}{A}\right)^{1/2}$ ft/sec	$C = 13890 \left(\frac{V}{A}\right)^{1/2}$ meters/sec
Exhaust Current per Unit Thrust	$\frac{I}{F} = \frac{3.089 \times 10^4}{A \left(\frac{V}{A}\right)^{1/2}}$ amp	$\frac{I}{F} = \frac{0.0695}{A \left(\frac{V}{A}\right)^{1/2}}$ amp
Propellant Flowrate per Unit Thrust	$\frac{\dot{M}}{F} = 7.060 \times 10^{-4} \left(\frac{V}{A}\right)^{-1/2}$ sec $^{-1}$	$\frac{\dot{M}}{F} = 7.199 \times 10^{-7} \left(\frac{V}{A}\right)^{-1/2}$ sec $^{-1}$

TABLE 12

(Continued)

	Definition: English System	Definition: Metric System
Specific Impulse	$I_s = 1417 \left(\frac{V}{A}\right)^{1/2}$ sec	$I_s = 1417 \left(\frac{V}{A}\right)^{1/2}$ sec
Minimum Throat Area per Unit Thrust (Ion Rocket Only)	$\frac{A_t}{F} = 5.651 \times 10^{11} \left(\frac{V}{d}\right)^{-2}$ sq ft	$\frac{A_t}{F} = 127.1 \left(\frac{V}{d}\right)^{-2}$ (meters) ^{2}/DYNE
Engine Operating Time	$t = \left[\frac{1-e^{\frac{-\Delta V}{4.557 \times 10^{4} \left(\frac{V}{A}\right)}}}{7.060 \times 10^{-4}\ G}\right]^{1/2} \left(\frac{V}{A}\right)^{1/2}$ sec	$t = \left[\frac{1-e^{\frac{-\Delta V}{1.386 \times 10^{6} \left(\frac{V}{A}\right)^{1/2}}}}{7.199 \times 10^{-7}\ a}\right] \frac{V}{A}^{1/2}$ sec
Equivalent Particle Weight	$A = \left[\sum \frac{I_i}{I} \left(\frac{A_i}{h_i}\right)^{1/2}\right]^2$ gm	$A = \left[\sum \frac{I_i}{I} \left(\frac{A_i}{h_i}\right)^{1/2}\right]^2$ gm

914

The ion discharge is balanced by an equal flow of oppositely charged particles; otherwise the surface of the vehicle develops a high induced charge, causing the ion flow to slow down and stop. An electron emitter is shown in Fig. 19. The electron discharge can be achieved by thermionic emission from a heated tungsten plate or by field emission of electrons from discharge points incorporated in the accelerating electrodes. An alternative method of obtaining ~~xxx~~ a neutral beam is by charge exchange techniques.

Power to drive the ion thrust chamber can be obtained from high-speed electrical power generators. Three types which may be applied to the ion rocket engine are a permanent magnet generator, an a-c generator, or an electrostatic generator. The distribution of power to the various components of the system is shown by the solid lines in Fig. 19. Most of the power is absorbed in pumping the electrons, which are collected at the ion source, up to the potential of the accelerating electrode. Power required to generate ions in the ion source is a significant factor in determining the power efficiency of the thrust device.

ION SOURCES

Several types of ion sources, including radio frequency, thermionicarc type, and surface contact, have been investigated for application in ion rocket engines.

What little information is available on radio frequency ion sources indicates that the efficiency is low and the associated operating equipment is heavy. These factors eliminate these sources for applications in the ion engine. Thermionic sources are based on the emission of ions from heated mixture of chemical compounds. These also have proved heavy and inefficient.

The adoption of an arc-type ion source or a surface contact source depends upon the power demands of the source. A brief preliminary analysis of the efficiency is carried out for both sources.

The power required for a surface-contact source includes that radiated from the heated ionizer surface, the heat required to vaporize thepropellant from ambient conditions, and radiation from structural parts. The last is difficult to predict except by test.

The efficiency of the surface-contact engine is:

$$\eta = \frac{P}{P + P_R + P_R'}$$

$$= \frac{1}{1 + \dfrac{1.04 \times 10^{-4} \; \epsilon T^4}{I^2 \left(\frac{V}{I}\right)^{\frac{1}{2}}} + \dfrac{P_R'}{P}}$$

P_R' represents the power required to vaporize the propellant, beam loss from interception, and other external losses. Preliminary computations indicate that these power losses are small in comparison to the radiation loss if the ion engine is well designed.

The efficiency of the arc system becomes:

$$\eta = \frac{P}{P + P_{arc} + P_R'}$$

$$= \frac{1}{1 + \dfrac{V_a}{V} \sqrt{1836A} + \dfrac{P_R'}{P}}$$

in which V_a is the operating voltage of the arc. Again P_R' represents the power for vaporization of propellant, loss from beam interception and other external losses.

after taking into account the mechanical and electrical units which the investigator wishes to use. In the cgs system of units, the thrust becomes:

$$F = 14.39\, \eta\, IV \left(\frac{V}{A}\right)^{1/2} \text{ dynes,} \tag{41}$$

and in the foot-pound-second system,

$$F = 3.235 \times 10^{-5}\, \eta\, IV \left(\frac{V}{A}\right)^{1/2} \text{ lb} \tag{42}$$

The parameter V/A, expressed in volts per gram, occurs in all of the engineering design parameters:

Specific Power

Power per Unit Thrust

Average Exhaust Velocity

Exhaust Current per Unit Thrust

Propellant Flowrate per Unit Thrust

Specific Impulse

Minimum Throat Area per Unit Thrust

Engine Operating Time

The detailed derivations of these parameters are developed in Ref. 1. They are summarized in Table 4, for the English and metric unit systems.

The data of Fig. 20 summarize and compare the variation of the power efficiency, η, of an arc-type ion engine and a surface contact engine. These preliminary data indicate how this parameter can be expected to vary, furnish additional information for selection of a propellant, and present areas of application for the two types of engines. Curve 1 is the efficiency variation with accelerating voltage of an arc-type engine with the arc operating at 20 volts and using cesium propellant. Curve 2 is similar data for an engine incorporating a 10-volt arc, again with cesium metal as the propellant. Many possible propellant materials are most efficiently ionized for arc voltages between 100 and 150 volts. Curve 5 indicates the efficiency of a 100-volt arc, using a propellant of the same ion weight as cesium.

Curve 3 is the efficiency variation of a surface-contact engine using cesium which is ionized by a 2200 deg K tungsten plate. The strength of the accelerating field, X, is maintained at 10,000 volts/cm. As the field strength exceeds 100,000 volts/cm, the efficiency approaches 100 percent, as shown in curve 4. In all computations the total emissivity of the tungsten ionizer was assumed to be 0.28.

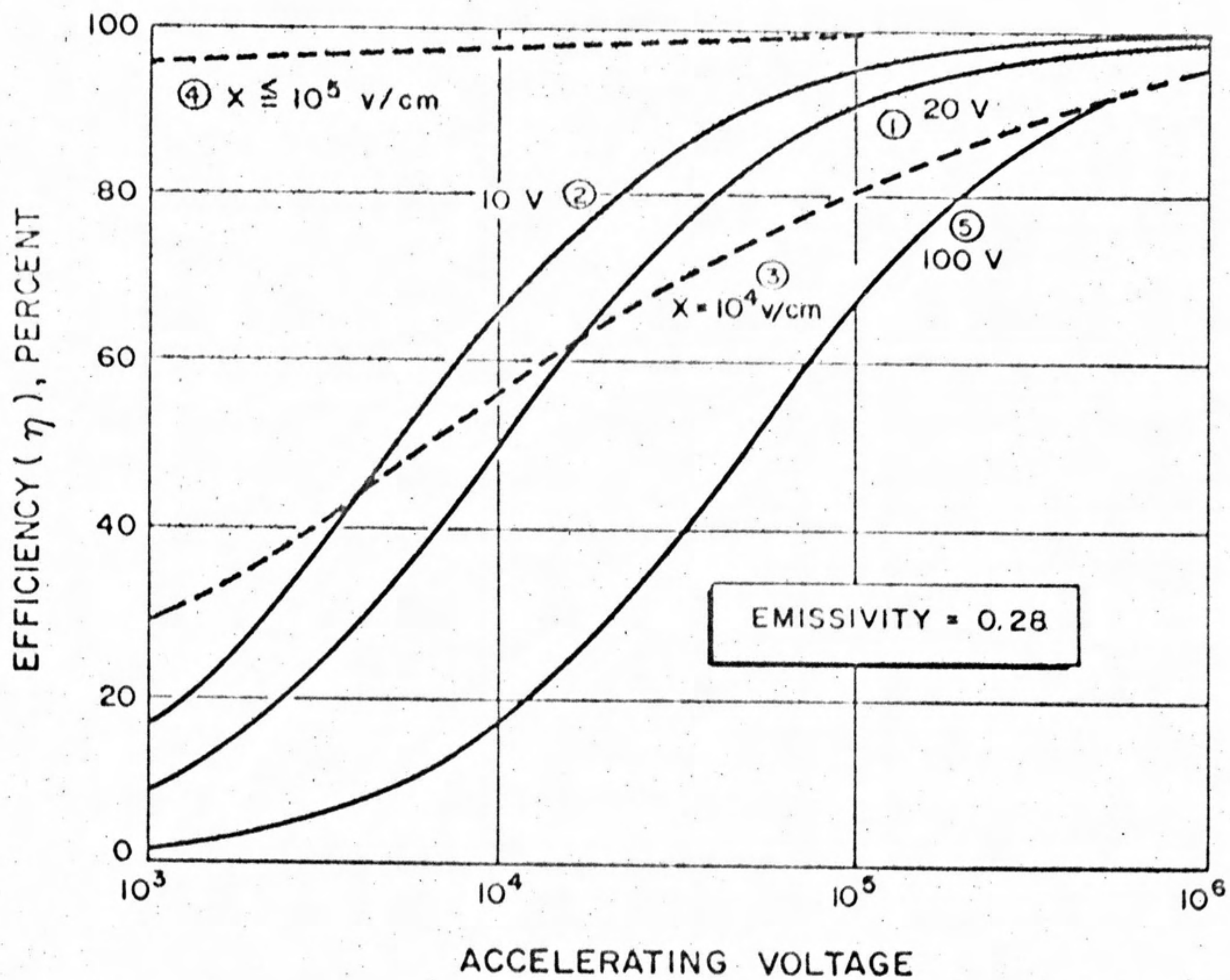


Figure 20. Comparative Efficiencies of Ion Rocket Engine

The power output per unit thrust that the vehicle must develop for the arc-type ion engine has been estimated from curve 1, Fig. 20 and the surface-contact ion engine from curve 3, Fig. 20. In both cases, sumarized in Fig. 21, cesium metal is the propellant. The arc engine assumes an arc operating voltage of 20 volts, which is close to the value for maximum operating efficiency . The surface-contact source assumes an operating temperature of 2200 deg K, an emissivity of 0.28, and accelerating field strength of 10,000 volts per centimeter.

These curves are not generally applicable, being computed for specific operating conditions which were selected to establish the minimum efficiences and maximum power per unit thrust of the two types of engines. The singificant factors are that in the specific impulse range covered by this summation, 3900 to 124,000 sec, the total power demand of the arc-type engine can be expected to have a minimum for a specific impulse between 11,000 and 13,000 sec.

These qualitative data indicate that, within the limits of the analysis, cesium ionized on tungsten is superior for low accelerating voltages and low specific impulse. In the intermediate range of specific impulse, a cesium arc is competitive with a cesium surface-contact engine,

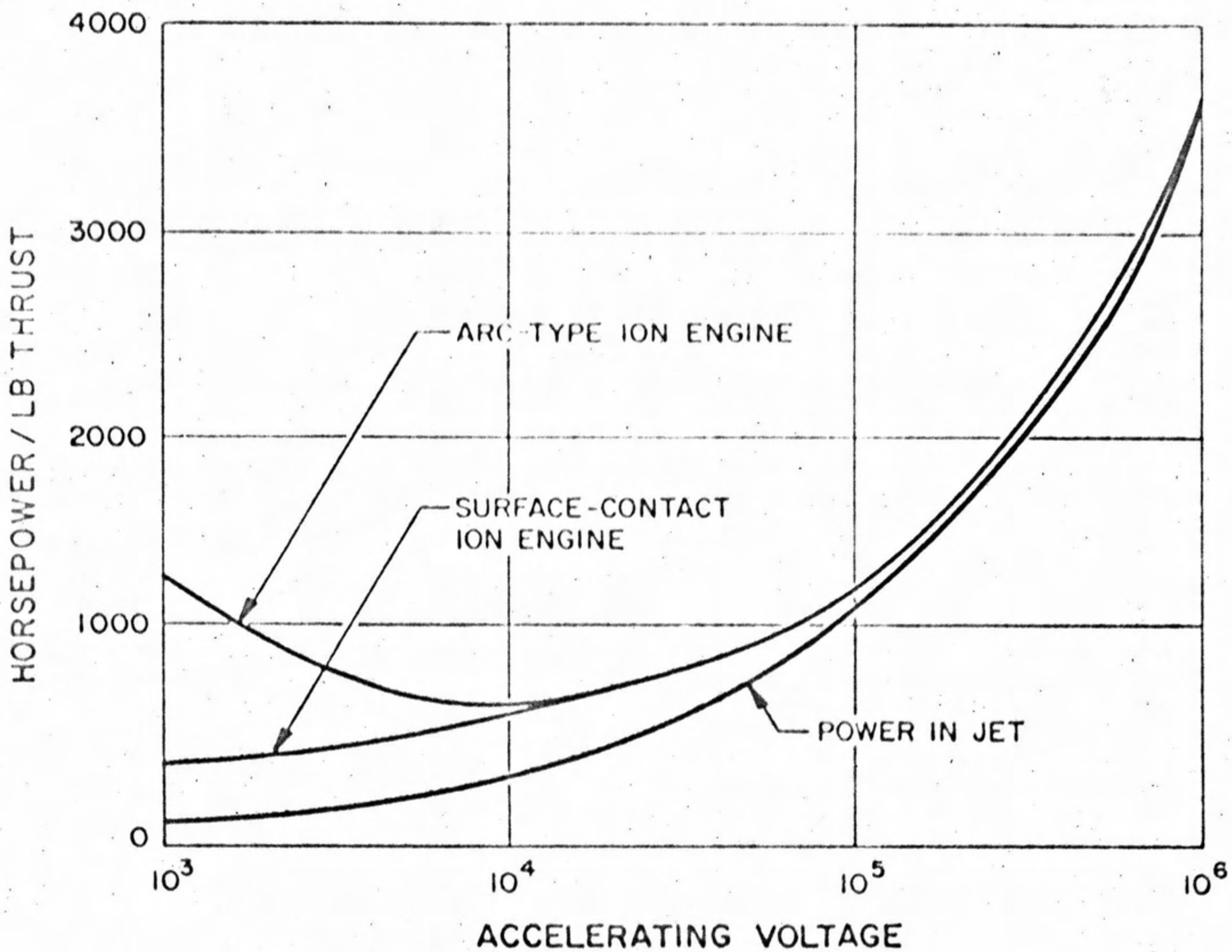


Figure 21. Power Requirements of Ion Engine

and is superior to an arc engine using a propellant requiring a high operating voltage. For high accelerating voltages and specific impulse the two types of sources are competitive.

The difference between the power requirements at lower specific impulse indicates a significant superiority for the surface-contact engine if the accelerating field strength is high. An accurate evaluation can only be made from the experimental results from tests of the two different engine types.

Since the plasma jet and magnetohydrodynamic engines all depend upon generation of plasma, and the designs reviewed postulated arc-type sources, consideration of the power effieiency of the plasma sources is important.

CHARGED COLLOID SOURCES

The generation of charged colloid particles is a two-stage process. First, the propellant must be dispersed as colloidal-sized particles in a near vacuum. Then, the colloidal particles are charged to give the desired charge-to-mass ratio and ejected through an accelerating electrode system to generate thrust.

Colloid propulsion is in early stages of research. It is still early to predict what combinations of techniques and equipment can be devised to produce a useful thrust device.

PROPULSION SYSTEM APPLICATIONS

The propulsion system weights and the operating times of electrical rocket engines are also convenient variables with which to study the operation of different low-thrust powerplants. The results of a study performed at Rocketdyne to determine the competitive operating regimes for rocket engines of thrust levels between 0.1 and 5.0 lb. (Ref. 24) are summarized in Fig. 23. The minimum propulsion system weight is shown as a function of the operating time of the rocket engines between the two thrust levels: 0.1 lb and 5.0 lb. The specific impulse for a minimum weight powerplant establishes the operating conditions of the electrical engines. These preliminary data indicate that chemical, electrothermal plasma, and ion powerplants for an upper-stage vehicle will be the lightest in the areas so labeled (Fig. 22).

Intermediate thrust levels and the specific impulse of the electrical thrust devices are shown. The chemical rocket system is arbitarily held at the fixed specific impulse of 330 sec. substantially all of the

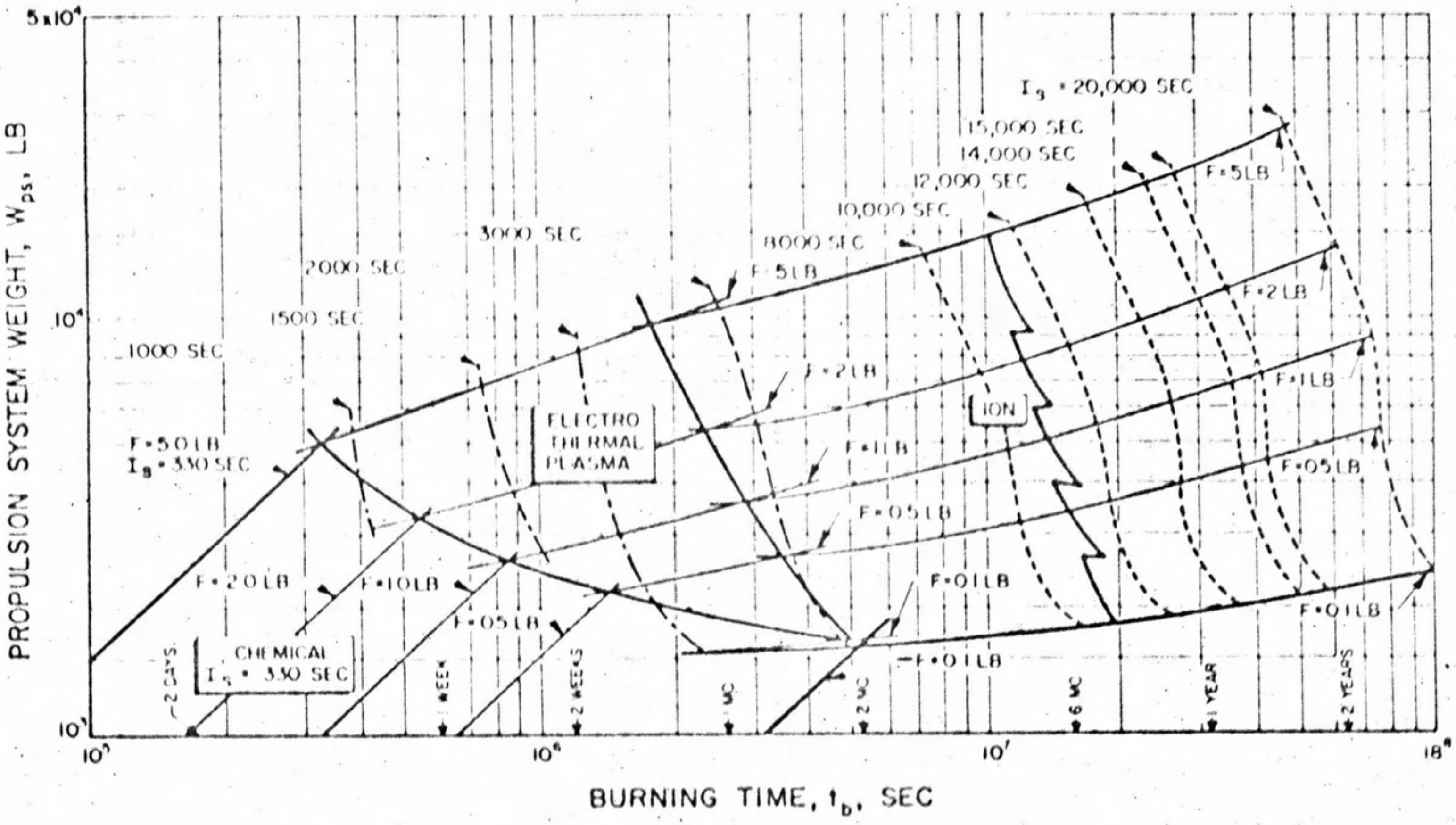


Figure 22. Propulsion System Weight vs Burning Time for Optimum Specific Impulse

propulsion system weight being in the propellant. The operating regime of the colloid engine overlaps the electrothermal plasma engine and the ion engine. How much overlap exists is now difficult to determine because the colloid engine is in the early research stage.

The method of analysis to obtain the data of Fig. 23 neglects the thrust chamber, feed system, and propellant tank weights, a working assumption for electrical propulsion systems which do not use heavy magnets or large capacitors as in the magnetohydrodynamic engines.

The weight of the powerplant system considered includes the powerplant section weight and the propellant

If the thrust of the second-stage vehicle is less than 0.1 lb, an ion engine is superior to both the chemical and colloid rockets provided the operating time for the vehicle mission continued for longer than six weeks. For shorter duration the low-thrust chemical engine will be the lightest system.

REFERENCES

1. Durand, W. F.: Aerodynamic Theory, Volume V, 1943.
2. Salisbury, J. K.: Kent's Mechanical Engineers Handbook, Power Volume, John Wiley and Sons, New York.
3. Marks, L. S.: Mechanical Engineers Handbook, McGraw-Hill Book Co., Inc., New York, New York.
4. Totten, A. I.: Trans. American Society of Mechanical Engineers, Paper #3559-3, May 1937.
5. Chapman: The Marine Power Plant, McGraw-Hill Book Co., Inc., New York, New York.
6. Taylor, Speed and Power of Ships, U.S. Government Printing Office, 1943.
7. Pye, D. R.: The Internal Combustion Engine, Two Volumes, Oxford University Press, 1931.
8. Livingston, S. P.; Gracey, W.: Tables of Airspeed, Altitude and Mach Number Based on Latest International Values for Atmospheric Properties and Physical Constants. National Aeronautics and Space Administration, Langley Research Center, Langley Field, Virginia, TN D-822, August 1961.
9. Zucrow, M. J.: Jet Propulsion and Gas Turbines, John Wiley & Sons, 1947.
10. Weick, F. E.: Working Charts for the Selection of Aluminum Alloy Propellers of Standard Form to Operate with Various Aircraft Engines and Bodies, NACA Technical Report 350.

11. Sutton, G. P.: Rocket Propulsion Elements, John Wiley & Sons, 1956.

12. Koelle, H. H.: Handbook of Astronautical Engineering, First Edition, McGraw-Hill Book Co., Inc., New York, New York, 1961.

13. Clarke, H. C.: Interplanetary Flight, Harper Brothers, New York City, 1951.

14. Moeckel, W. A.: Fast Interplanetary Missions with Low-Thrust Propulsion Systems, NASA TR R-79, Lewis Research Center, Cleveland, Ohio, 1961.

15. Irving, J. H.; Blum, E. K.: Comparative Performance of Ballistic and Low-Thrust Vehicles for Flight to Mars, Vistas in Astronautics, Volume II, Pergamon Press, 1959.

16. Morgan, N. E.; Marsth, W. D.; Thomas, R. C.: Piston Engines Offer Advantages for Space Power Applications, Aerospace Fluid Power Conference, Detroit, Michigan, October 29-30, 1962.

17. Thirtieth Inventory of Aerospace Power, Aviation Week and Space Technology, Vol. 78, No. 10, 11 March 1963, McGraw-Hill Book Co., New York.

18. Judge, A. W.: Gas Turbines for Aircraft, Chapman & Hall, LTD, London W.C.2., 1958.

19. Lombard, A. A.: Low Consumption Turbine Engines, Fifth International Aeronautical Congress, Los Angeles, California, June 20-25, 1955, Institute of the Aeronautical Sciences, Inc., New York 21, New York.

20. Giles, C.: The V-1 Robot Bomb, Astronautics, September 1944.

21. Warren, F. A.: Rocket Propellants, Reinhold Publishing Corporation, New York, 1958.

22. Post, Richard F.: Controlled Fusion Research - an Application of the Physics of High Temperature Plasmas, Rev. of Modern Physics, Vol. 28, No. 3, July 1956.

23. Huebner, A. L.; Boden, R. H.: Critical Power Supply Problems in Ion Propulsion, Space Exploration Regional Meeting, Institute of the Aeronautical Sciences, American Rocket Society, 5-6 August 1958, San Diego, California.

24. Littman, T. M.: A Critical Evaluation of the Ion Rocket, Second Symposium on Advanced Propulsion Concepts, AFOSR, Boston, Massachusetts, 1 October 1959.

25. Alperin, M.; Sutton, G. P.: Advanced Propulsion Systems, Pergamon Press, 1959.

26. Spitzer, Jr. L.: The Physics of Fully Ionized Gases, Interscience Publishers, Inc., New York, 1956.

27. Kash, S. W.; Starr, W. L.: Experimental Results with a Collinear Electrode Plasma Accelerator and a Comparison with Ion Accelerators, Reprint 1008 - 1959, American Rocket Society Annual Meeting, Washington, D. C., November 16-20, 1959.

28. Fonda-Bonardi, G.: Final Report - Research Study on Plasma Acceleration, AFOSR TR-59-170, Litton Industries, Beverly Hills, California, September 30, 1959.

29. Patrick, R. M.: Description of a Propulsive Device Employing a Magnetic Field, Research Report No. 28, ASTIA No. AD 159614, AVCO Research Lab., Everett, Mass., May 1958, Contract AF49(638)-61.

30. Boden, R. H.: The Ion Rocket Engine, AFCSR TN 57-573 (R-645), Rocketdyne, A Division of North American Aviation, Inc., Canoga Park, California 3 July 1958.

31. Sanders, R. F.: The New Chevrolet V-8 Engine, Transactions, Society of Automotive Engineers, Vol. 63, 1955, Pages 401-420.