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GENERAL SUMMARY

This report is a consideration of power plant requirements as applied to a satellite rocket study and is separated into two parts for convenience and clarity. PART A deals with the main thrusting and guiding rocket motors, whereas PART B is devoted to the auxiliary power supply which energizes the electrical payload.

Each of the two parts has its own separate, detailed summary: see pages 1 and 31.

SATELLITE ROCKET POWER PLANT

PART A. MAIN ROCKET POWER PLANT

I. SUMMARY

Each stage of the proposed three-stage satellite rocket will have its own complete power plant. The results of a study for a power plant for each stage which will meet the requirements of the trajectory[1], structural weight[2], and fuel[3] studies are outlined below. The motors were designed for a constant propellant flow rate. A power plant which would produce constant acceleration in the second and third stages of the satellite rocket would reduce the overall gross weight of the vehicle about 9%[1]. This power plant study was completed before it became apparent that such a constant acceleration motor might be feasible for these stages without undue complication or increase in motor weight (see Appendix XIII). The propellants used throughout this study are hydrazine and liquid oxygen in a mole ratio of 1.5 : 1.

STAGE I

A pump-fed rocket power plant is used with vaporized liquid oxygen to maintain a given minimum pressure in the oxygen tank, and helium from a high pressure supply to maintain that pressure in the hydrazine tank.

In addition to the main rocket motor, four smaller rocket motors, the axes of thrust of which can be controlled, are used for satellite attitude control. Jet vanes are not used.

All motors are supplied with propellants by the same turbine-centrifugal pump system. All motors are regeneratively cooled.

STAGE II

A pump-fed rocket power plant is used with helium from a high pressure supply maintaining a given minimum pressure in both the oxygen and hydrazine tanks. Four control motors, equally spaced about the main motor, are again used.

All motors are supplied with propellants by the same pumping system. The centrifugal pumping system will be started in sufficient time to allow full propellant pressure to be available to the rocket motors at the end of stage separation. As at the time of stage separation the propellants are in a relatively acceleration-free condition, a small gas bag is incorporated in each of the propellant tanks to take up any excess volume not occupied by the propellants. This insures that the propellants will be in the lines leading to the pumps. All motors are regeneratively cooled.

For references see page 109.

STAGE III

In this stage there is a period of coasting during which the main rocket motor is shut off and after which it is again started and operated for a few seconds. During the coasting period it is desirable to operate the control rocket motors to maintain attitude control. For this reason two separate propellant supply systems are used in this stage, one for the main rocket motor and the other for the control motors. These are described separately:

Main Rocket Motor

A pump-fed rocket power plant is used, similar to that of stage II, with the added provision for complete shut-down and restarting.

Control Rocket Motors

A pressurized rocket power plant is used throughout the stage III trajectory. The propellant tanks are pressurized to a value which will produce the required combustion chamber pressure. Small gas bags in the propellant tanks are again needed to insure that there is propellant in the delivery lines during stage separation.

Due to the large ratio of surface area to rate of propellant flow of these motors, regenerative cooling cannot be accomplished without some means of reducing the rate of heat flow through the motor walls. It will be necessary to ceramic line, sweat cool, or film cool these motors.

II. LIST OF SYMBOLS FOR PART A

η = efficiency

$C_{F_{th}}$ = theoretical thrust coefficient

C_F = actual thrust coefficient

p_c = combustion chamber pressure, psia

p_o = external static pressure, psia

A_t = throat area, sq in.

W_i = initial gross weight of a given stage, including the succeeding stages as pay load, lb

n_i = initial number of g's (acceleration of gravity at sea level = 32.174 ft/sec^2)

d_t = throat diameter of motor, in.

d_c = combustion chamber diameter of motor, in.

d_e = nozzle exit diameter, in.

ℓ_c = length of combustion chamber of motor, in.

ℓ_1 = length of converging section of nozzle, in.

ℓ_2 = length of diverging section of nozzle, in.

ℓ_a = overall length of rocket motor, in.

V_c = combustion chamber volume, cu in.

L^* = characteristic length of combustion chamber, in.

F = initial total thrust for given stage, lbs

III. INTRODUCTION

The optimum trajectory, gross weight, and fuel system studies presented in references 1, 2, and 3 have determined the requirements for a satellite rocket which will establish its payload on an orbit about the earth at an altitude of 350 miles. As hydrazine and liquid oxygen were selected as propellants, these requirements call for a three-stage rocket vehicle[±]. The power plant which uses these propellants in this satellite rocket is described in this report.

The requirements of a satellite power plant are that it be able to lift its own weight, the weight of the propellants, structure, and payload, to overcome the drag in the lower atmosphere, and to accelerate the payload to an orbital velocity of 23,500 feet per second at an altitude of 350 miles. The ramjet, turbo-jet, solid propellant rocket, and liquid propellant rocket were the high speed propulsion devices considered. Inasmuch as the maximum ceiling of an air-consuming engine is only about eleven miles and as it was desired to use a power plant with a minimum weight and minimum complexity of construction and operation, a liquid-propellant, pump-fed rocket power plant was chosen. Using hydrazine and liquid oxygen as the propellants in this power plant, the trajectory[1] and structural weight[2] studies have resulted in a satellite vehicle with a reasonable gross weight (86,400 pounds). The propellant studies[3] indicate that there are a number of possible propellant systems which will give comparable or lower gross weights. The hydrazine-liquid oxygen system was chosen for the satellite rocket because of its relatively high specific impulse, high density, and ease of handling in tanks, plumbing, pumps, and valves.

The feasibility, the description, and the problems encountered in the power plant for the satellite rocket using hydrazine-liquid oxygen are discussed in the report.

[±] A staged rocket is defined as one in which a complete rocket assembly (consisting of power plant, fuels, and structure) carries as its payload another complete rocket assembly. At the end of burning of the first assembly the two assemblies are separated from one another and the second proceeds alone.

The following pertinent factors are considered in detail:

a. Throatless vs conventional motors

b. Advisability of using multiple motors in each stage rather than a single large motor

c. Rocket motors whose thrust axis can be adjusted relative to the flight path vs jet vanes for attitude control of the satellite rocket

d. Rocket motor sizes and shapes

e. Gas-fed vs pump-fed propellant delivery system

f. The pumping problem

g. The problem of quick and reliable starting and continued operation of the propellant supply system under both the accelerated and acceleration free conditions.

IV. ANALYSIS AND DISCUSSION

A. GENERAL DISCUSSION

1. General Rocket Power Plant

In general a pump-fed rocket power plant consists of, in addition to the rocket motor itself and its propellants, a turbine and pump assembly with fuel for the turbine, gas to pressurize the turbine fuel tank and gas to maintain a certain minimum pressure in the propellant tanks, along with associated valves and piping. This general arrangement is used in stages I and II and for the main rocket motor of stage III.

The control rocket motors of stage III are operated as pressurized rockets. Such a system consists of, in addition to the rocket motors, only the propellants for the rocket motors, gas to pressurize the propellants to such pressure as required for delivery to the combustion chamber, and associated valves and piping.

2. Requirements of a Satellite Rocket Power Plant

The satellite studies on trajectories[1], structural weights[2], and fuels[3] resulted in the parameters given in Table I which are useful in the design of the power plants of the three stages.

Table I.

DESIGN PARAMETERS FOR THREE-STAGE SATELLITE VEHICLE
USING HYDRAZINE-LIQUID OXYGEN

	Stage I	Stage II	Stage III
Initial gross weight, lb	86,400.	18,600.	4615.
Initial load factor	1.44	1.72	1.72
Final load factor	5.0	5.0	5.0
Ratio of propellant weight to gross weight	.6532	.6532	.6532
Initial effective specific impulse, sec	243.	298.2	296.7
Final effective specific impulse, sec	287.5	298.2	296.7
Mixture ratio (weight of hydrazine to weight of oxygen)	1.5	1.5	1.5
Burning time, sec	112.	114.9	(A) 101.0 (B) 13.7
Propellant consumption, lb/sec	512.	107.5	26.4
Control thrust to total thrust	0.706	0.150	0.0151

From these data the unknowns such as total weight of propellant, weight rate flow of propellant, the number and size of rocket motors, the power and size of the turbine-pump assemblies, high pressure gas requirements, and the volumes of all the tanks can be determined. From these results the kind and quantity of turbine fuel, the method of motor cooling, and the method of operation of the power plant may be specified.

Before the rocket motors themselves can be designed it is necessary to answer several general questions such as whether to use throatless type[±] rocket motor combustion chambers or the conventional type, whether multiple motors for a given stage or one large motor be used, and whether jet vanes will be used for attitude control of the satellite rocket.

3. Throatless Rocket Motors

The use of throatless motors was considered because these motors have a higher thrust-weight ratio than the conventional throated motor. But analysis (see Appendix I) indicates that the loss in motor efficiency is greater than the saving in motor weight, especially for motors of long duration. Therefore, all the motors in the designed satellite vehicle have throated converging-diverging nozzles.

4. Multiple Rocket Motors

The question of whether one or more than one motor shall supply the required thrust in a given stage is a problem of weighing the advantages and disadvantages of

[±] Throatless type rocket motor refers to one in which the combustion chamber diameter is equal to the throat diameter of the nozzle used.

each. In general, multiple motors would have the following advantages:

1. Easier to manufacture due to smaller size (except in the very low thrust range: thrust < 100 pounds).

2. Easier to test and experimentally develop.

3. Thrust may be varied by operating a different number of motors.

4. The per cent saving in weight of the nozzle as given by

$$100\left(1 - \frac{1}{\sqrt{m}}\right)$$

where m is the number of motors (see Appendix II).

5. If the total throat area is larger than .00725 $(L^*)^2$, multiple motors result in lighter combustion chamber shells. (See Appendix III for the assumptions made.)

6. Thinner walls mean less temperature gradient and therefore less thermal stresses in the smaller motors.

7. Some of the multiple motors may be pivot-mounted so as to provide complete attitude control for the rocket vehicle.

In general, multiple rocket motors have the following disadvantages:

1. Greater complication of plumbing and operation, particularly the difficulty of simultaneous starting.

2. Greater space requirements in vehicle.

3. If the motors become too small, the ratio of surface area to propellant flow becomes so great that it is no longer possible to cool the motor solely by the regenerative principle.

4. If the total throat area is smaller than .00725 $(L^*)^2$, multiple motors result in heavier combustion chamber shells. (See Appendix III.)

Based on these advantages and disadvantages, the use of multiple motors in at least stages I and II of the satellite rocket appears desirable. However, the optimum sizes and number of motors per stage were compromised for reasons of arrangement, simplicity, and attitude control.

5. Control Rocket Motors vs Jet Vanes

For attitude control of rocket vehicles, jet vanes have generally been used in the past. An analytical investigation (see Appendix IV) indicates that auxiliary rocket motors, pivot mounted[‡], would be more desirable than jet vanes for steering the satellite rocket.

[‡] Rocket motors whose thrust axis can be adjusted relative to the flight path for attitude control.

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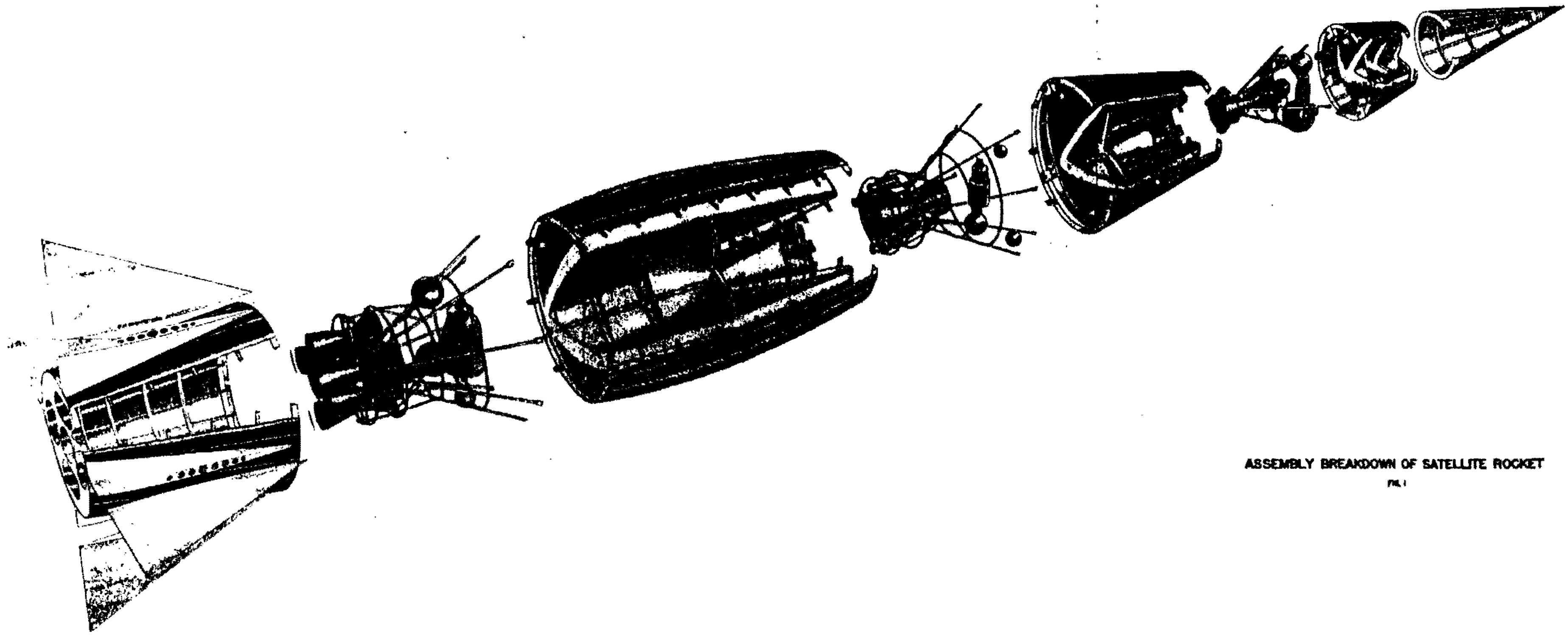

ASSEMBLY BREAKDOWN OF SATELLITE ROCKET

FIG. 1

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The primary advantage accruing with control rocket motors for attitude control is that the use of control motors would eliminate the high parasitic and angle of attack drag losses associated with jet vanes as discussed in Appendix IV. The problem of cooling long duration jet vanes would not be encountered. Weight saving will be gained by not using the heavy carbon jet vanes because the total weight of multiple motors per stage, including flexible propellant lines, should be at least as small as one large motor. By the use of large control motors, compared to the main motor, all the advantages of multiple motors discussed above may be gained. Furthermore, the requirements of the adopted trajectory call for vehicle control with the main motor off in stage III. This requirement is met simply by control motors, but if jet vanes were used a separate device for attitude control would be required during coasting as, of course, the main rocket motor is not operating during this time, hence the jet vanes are ineffective.

As a result of these considerations, control motors are used in all three stages of the satellite vehicle.

B. DESIGN OF ROCKET MOTORS FOR THE SATELLITE ROCKET

1. General Arrangement and Number of Rocket Motors

With the general background of the preceding section, the specific motors for the 86,400-lb hydrazine-liquid oxygen three-stage satellite vehicle may be designed. To simplify the control of yaw, pitch, and roll, four control motors are equally spaced about the main motor in each stage as shown in Fig. 1. The amount of thrust required from each of the control motors, based on a maximum deflection of 15 degrees, was determined from a study of control forces and moments required[4]. The ratio of control thrust to total thrust per stage is given in Table I. The control motors were designed just large enough to give the required thrust. Therefore, the full utilization of the advantages of multiple motors has not been achieved. The control motors are kept as small as possible because of this consideration: a large thrust from the control motor means only a small angular displacement is necessary for attitude control. When this necessary displacement becomes very small, it becomes of the same order of magnitude as the error in the deflection sensing device, and then the servo-motors cannot give the proper control.

2. Throat Area Required

For the determination of the dimensions of the individual motors, it is necessary to know the total motor throat area required for each stage. This is found from the general expression for rocket thrust

$$F = C_{F_{th}} \cdot \eta_V \cdot \eta_F \cdot p_c \cdot A_t \,, \tag{1}$$

where

$C_{F_{th}}$ = theoretical thrust coefficient =

$$\sqrt{\frac{2\gamma^2}{\gamma-1}\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}}\left[1-\left(\frac{p_e}{p_c}\right)^{\frac{\gamma-1}{\gamma}}\right]} + \left(\frac{p_e - p_0}{p_c}\right)\frac{A_e}{A_t} \tag{2}$$

η_V = efficiency factor which accounts for the loss due to the transverse component of the exhaust velocity

η_F = efficiency factor which includes the following in this analysis:

1. Frictional losses in the combustion chamber and nozzle
2. Burning efficiency
3. Deviations from the adiabatic expansion process (heat loss through shell, etc.)
4. Loss in thrust due to deflection of the control motors from the thrust axis (a maximum of 2%).

The thrust required at the start of each stage is simply

$$F = W_i n_i \quad . \tag{3}$$

The total throat area per stage is (combining Eqs. (1) and (3))

$$A_t = \frac{W_i n_i}{C_{F_{th}} \eta_V \eta_F p_c} \quad . \tag{4}$$

The values of W_i, n_i, and p_c are obtained from the optimum gross weight studies[1,2], and are listed in Table II.

Table II

THROAT AREA REQUIRED PER STAGE OF SATELLITE ROCKET

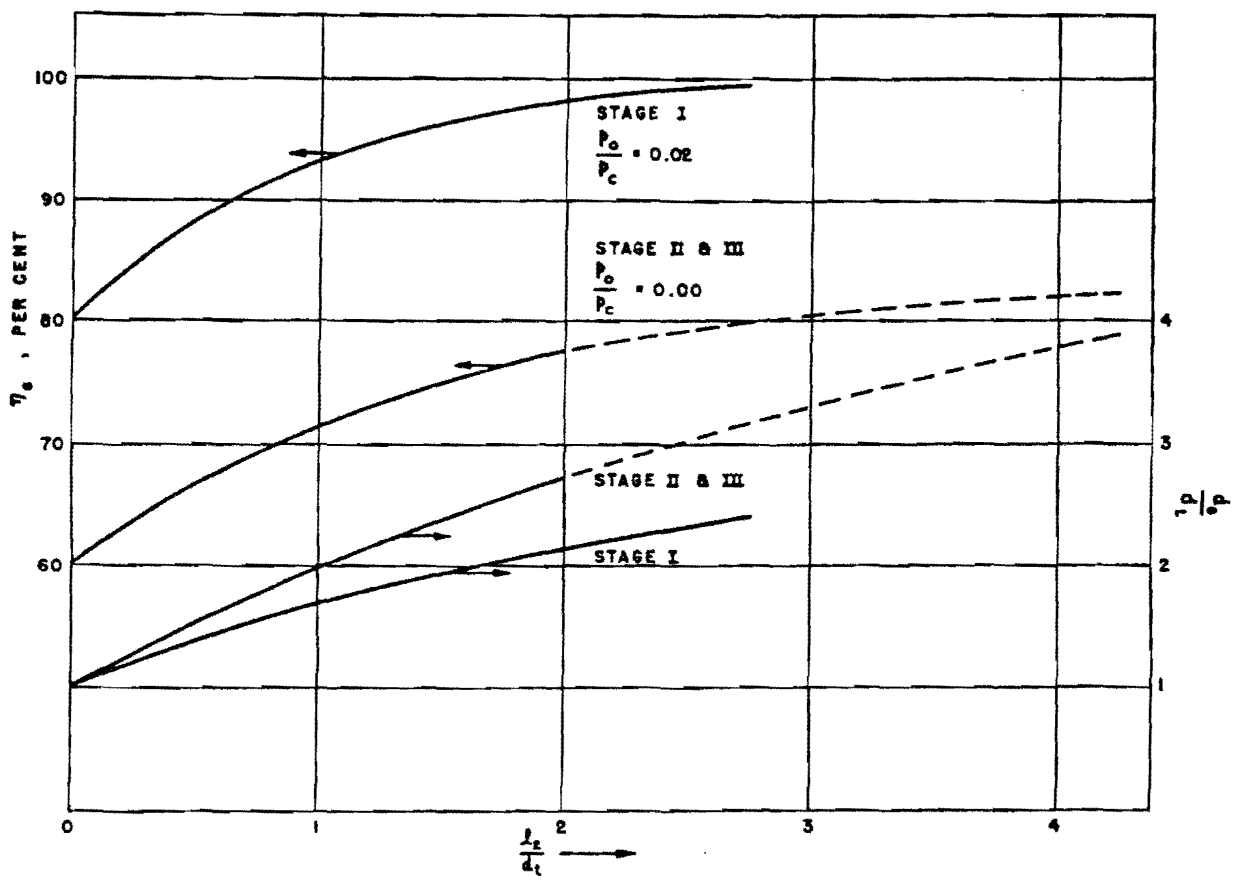
Stage	Gross Wt lbs W_i	Initial Load Factor n_i	Thrust lbs F	Chamber Pressure psia p_c	Theoretical Thrust Coefficient $C_{F_{th}}$	% η_V	% η_F	sq in. A_t
I±	86,400	1.44	124,400	400	1.39	99.5	90.5	249
II	18,635	1.72	32,100	300	1.76	98.5	90.5	69
III	4,615	1.72	7,940	150	1.76	98.5	90.5	33.6

± Values for take-off at sea level.

The values of $C_{F_{th}}$ were calculated from Eq. (2). (Note that the value of $C_{F_{th}}$ for stage I, 1.39, is for sea level.) The motor efficiency, η_F, which was defined above, was assumed to be equal to 0.905 for all motors.

In order to obtain the highest value of η_V, the diverging portion of the de Laval exhaust nozzle is designed according to ref. 6. This design provides that for a given pressure ratio and nozzle length over throat diameter ratio, the maximum possible thrust will be produced. Essentially this means that for the given conditions the incremental velocity vectors of the exhaust jet are more nearly parallel to the central thrust axis than with any other nozzle contour. Stage I of the satellite rocket has a mean expansion ratio very close to 0.02 and stages II and III operate with ratios approximately equal to zero. Therefore, the 0.02 and 0.00 contours and their efficiencies were used in all the satellite motors and are reproduced from ref. 6 in Fig. 2. The nozzle efficiency referred to above is denoted by η_G and is defined as follows:

$$\eta_G = \eta_V \times \frac{C_{F_{th}}\text{(for expansion to a given area ratio, parallel-flow exhaust)}}{C_{F_{th}}\text{(for complete expansion, parallel-flow exhaust)}} \qquad (5)$$



NOZZLE CONTOURS AND EFFICIENCIES FOR BEST POSSIBLE DIVERGENT SECTION OF NOZZLE

FIG. 2

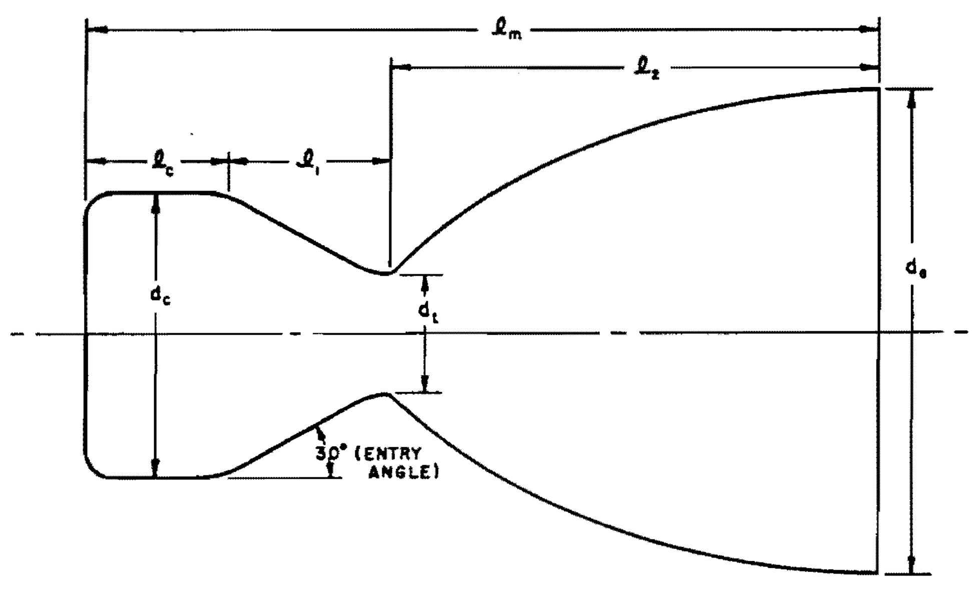
The p_0/p_c = 0.02 contour at a ℓ_2/d_t ratio of 2.76 gives the efficiency η_G = 99.4% and from Eq. (5) η_V is found to be 99.5%. This contour and length over diameter ratio is used in stage I motors. However, the p_0/p_c = 0.00 curve was shown only to a ℓ_2/d_t ratio of 2.05 in ref. 6 and this ratio gives a value of η_G = 77.6%. As a better efficiency is desired, the contour and efficiency are extrapolated in Fig. 2, as suggested in the reference, to an ℓ_2/d_t ratio of 4.25 which gives an area ratio of 15. This results in the efficiencies of η_G = 82.4% and η_V = 98.5%. The possibility of error in the efficiency and contour at the new area ratio is recognized. These curves may be accurately calculated by a lengthy mathematical process which was not done for this paper but which would be necessary before such a nozzle could be built. The important items obtained from Fig. 2 are the length and shape of the nozzles which will give reasonably high efficiencies (neglecting the absolute values of these efficiencies).

The length of the nozzles described in this report were determined only from the consideration of the η_V and the weight. A further study should be conducted to determine the optimum lengths with the friction losses and the cooling requirements also taken into consideration.

With all the values now known for Eq. (4), the total throat area per stage may be calculated and is given in Table II.

3. Rocket Motor Dimensions

The dimension notation and general shape of the rocket motors for the satellite rocket are given in Fig. 3. Inasmuch as the thrust determines the throat area, all the dimensions are given as a function of the throat diameter.



GENERAL LAYOUT AND NOTATION USED IN ROCKET MOTOR DESIGN

FIG. 3

Combustion Chamber Diameter

An area ratio of 6.25 based on current rocket design practice is chosen for the converging portion of the exhaust nozzle, therefore,

$$d_c = 2.5\ d_t \quad , \tag{6}$$

except where $2.5\ d_t = d_c < \ell_c/2.4$. In this case d_c is taken as $\ell_c/2.4$. This is done to prevent the combustion chamber aspect ratio from becoming excessive in the small motors.

Length of the Converging Section of Nozzle

The entry angle of the nozzle is chosen as large as possible in order to reduce the length of the nozzle, but this angle should not be too large or the throat curvature will be too sharp. This is undesirable for two reasons: first, greater difficulties with nozzle erosion will be experienced; second, the transition between subsonic and supersonic flow will move downstream from the throat resulting in separation losses at the sharp throat profile. Previous testing of various rocket motors has indicated that 30° is a good compromise value for the entry angle. The length of the converging section of the nozzle is then

$$\ell_1 = \frac{1}{2} \cot 30^\circ \left(d_c - d_t\right) \ . \tag{7}$$

This straight-sided cone in the actual motor would be faired at both ends as shown roughly in Fig. 3. It is particularly important that the fairing between the cone and the throat produce parallel flow at the throat.

Length of Combustion Chamber

The length of the combustion chamber can be determined from one of two parameters which are characteristic of the propellant used. These parameters are relative volume, which is the ratio of combustion chamber volume, V_c, to the rate of propellant consumption and characteristic length, L^*, which is the ratio of the combustion chamber volume to throat area. Inasmuch as the throat area is proportional to the propellant flow rate, these parameters are practically identical. They both are proportional to the chemical reaction rate of a given combination of propellants. If the combustion volume is assumed to include the converging portion of the nozzle, the length of the cylindrical portion of the combustion chamber is found as follows:

$$V_c = L^* \frac{\pi d_t^2}{4} = \ell_c \frac{\pi d_c^2}{4} + 1/3\ \pi \ell_1\ 1/4 \left(d_c^2 + d_c d_t + d_t^2\right) \ ,$$

$$L^* d_t^2 = \ell_c d_c^2 + 1/3\ \ell_1 \left(d_c^2 + d_c d_t + d_t^2\right) \ ,$$

$$\ell_c = \frac{L^* d_t^2}{d_c^2} - \frac{\ell_1}{3 d_c^2} \left(d_c^2 + d_c d_t + d_t^2\right) \ . \tag{8}$$

The characteristic length, L^*, is at present unknown for the hydrazine and liquid-oxygen combination and must be experimentally determined. The value of L^*

estimated for this propellant combination is 70 inches. This estimate was based on known values of L^* for the acid-aniline and alcohol-oxygen systems. When the correct value of L^* has been determined the necessary adjustment in ℓ_c may be made.

Length of the Diverging Portion of the Nozzle

The length of the diverging portion of the nozzle is obtained by multiplying the ℓ_2/d_t ratio (obtained from Fig. 2) for the proper pressure ratio by the throat diameter.

Exit Diameter of Exhaust Nozzle

The exit diameter of the exhaust nozzle is obtained by multiplying the d_e/d_t ratio (obtained from Fig. 2) for the proper pressure ratio by the throat diameter. The actual shapes of the diverging portion of the nozzles will follow the contours given in ref. 6 and reproduced in Fig. 2.

The dimensions of the main and control motors in all the stages have been calculated as outlined above and the results are given in Table III.

Table III

ROCKET MOTOR DIMENSIONS FOR SATELLITE VEHICLE

Stage	Motor	Thrust per Motor lbs	Diameters (in.)			Lengths (in.)			
			Throat d_t	Exit d_e	Comb. Chamber d_c	Comb. Chamber ℓ_c	Converging Nozzle ℓ_1	Diverging Nozzle ℓ_2	Overall ℓ_m
I	Main	36,400	9.67	23.2	24.2	4.65	12.6	26.7	43.95
	Control	22,000	7.46	17.9	18.66	6.15	9.73	20.6	36.48
II	Main	27,300	8.64	33.3	21.6	5.34	11.27	36.5	53.11
	Control	1,200	1.817	7.01	4.54	9.97	2.37	7.67	20.01
III	Main	7,820	6.48	25.0	16.20	6.82	8.46	27.3	42.58
	Control	30	0.398	1.538	1.605	3.86	1.05	1.68	6.59

The relative size of the main and control motors in each stage is also given in Table III. The thrust in the first stage is supplied by five motors in this proportion: 1 - 1 - 1 - 1 - 1.68. This division of size gives most of the multiple motor advantages and allows the combustion chambers to approach the optimum shape in stage I (see Appendix III).

In the second stage, the main motor is much larger than the control motors. This means that the proportioning in stage II is quite different from the optimum. However, as is pointed out in the discussion of multiple motors, some of their advantages disappear when the total thrust decreases. Hence, the weight penalty incurred by not using optimum multiple motors is not so great as in stage I. Another reason for using small control motors in stage II, besides the one of control precision previously discussed, is the composite nozzle exit diameter consideration. With the present arrangement of the stage I oxygen tank around the stage II motors (see Fig. 1), the stage II trailing edge forms the maximum diameter of the satellite rocket. It is desirable from an aerodynamic standpoint to keep this diameter at a reasonably low value. The equal division of thrust to the five motors in stage II, without the use of a heavier notched oxygen tank, would greatly increase this diameter.

In stage III the control motors are made small for three reasons. There is again the precision of control, and it is again desirable to keep the vehicle diameter a minimum value at this station. The other reason is that the control motors must operate during seven minutes of coasting and it is imperative that the propellant consumption be very small during this period. The main motor in stage III has the optimum combustion chamber shape if an L^* of 70 inches is assumed. This is most important in stage III because a saving in motor weight here means an equivalent increase in useful payload. The control motors in this stage, each producing 30 pounds of thrust, are about the size of the smallest liquid rocket motor which has been built to date.

The largest individual motor recommended for the satellite rocket is required to produce 36,400 pounds of thrust. It is of interest to note that this is only 66% as much as the thrust produced by the German A-4.

4. Predicted Performance of Rocket Motors

A summary of the predicted performance of the motors of the satellite rocket is given in Table IV.

Table IV

PREDICTED PERFORMANCE OF THE ROCKET MOTORS OF THE SATELLITE VEHICLE
(Control and Main)

Stage	Thrust	Specific Impulse	Specific Propellant Consumption	Est. Thrust Weight Ratio	Actual Thrust Coefficient	Effective Exhaust Velocity	Characteristic Velocity	Chamber Area to Throat Area	Exit Area to Throat Area
	lbs	sec	1/sec		C_F	ft/sec	ft/sec		
I	124,400	243+	.00412+	73.2+	1.25	7810	6250	6.25	5.76
II	32,100	298.2	.00335	80.0	1.56	9600	6150	6.25	15.0
III	7,940	296.7	.00337	80.0	1.56	9540	6110	6.25±	15.0

\+ Take-off values (sea level)

± Ratio of chamber area to throat area in the control motors of stage III is 16.25. In stages I and II this ratio is the same for both the main and control motors and is 6.25.

5. Motor Cooling

As the density of heat flow (BTU per sq in. per sec) is unknown at present for rocket motors using hydrazine and liquid oxygen in a molecular ratio of 1.5 as propellants, a detailed analysis of the problem of cooling these motors is impossible at this time. However, an indication of whether the rocket motors can be cooled regeneratively can be obtained by assuming a value of the average density of heat flow based on known values for various propellant combinations and the difference in combustion temperature.

The investigation (Appendix V) is based on an overall rise of coolant temperature, a specific heat of 0.583 BTU/# °F for hydrazine (the coolant) and an average density of heat flow of 1-1/2 BTU per sq in. per sec. The assumption is made that the most difficult condition for regeneratively cooling a rocket motor is that of a low thrust motor operated at low chamber pressure and exhausting to a vacuum (large surface area, and low rate of propellant flow which is the condition of the rocket motors of stage III). The sample calculation in Appendix V indicates that all of the motors can probably be successfully regeneratively cooled with the exception of the control rocket motors in stage III. These motors require some method of reducing the average density of heat flow from the hot combustion gases to the cooling liquid which is hydrazine if the coolant is not to boil in the coolant chamber. A possible solution would be the use of a ceramic liner on the inner surface of the motor walls. However, further research in this direction is necessary because of the high combustion temperature.

The proper design of the injectors and the combustion chamber also reduces the density of heat flow. This, in addition to the use of ceramic liners, should provide a solution to the cooling problem. If this does not provide a solution it will be necessary to sweat cool or film cool the motors. Sweat and film cooling have the disadvantage that the coolant used for this purpose does not produce as much thrust as when burned directly in the chamber. It is true the percentage of coolant used in this manner is small, but it is significant from a weight standpoint for a long duration rocket.

C. PROPELLANT DELIVERY SYSTEM

1. Pump-Feed vs Gas-Feed

A propellant delivery system for a three-stage satellite vehicle must meet the varying requirements of each stage and result in a minimum vehicle gross weight. With this end in view an optimum study of gas-feed and pump-feed propellant delivery systems was conducted and is described herein. The system chosen for each stage was based on the minimum weight and operational reliability considerations.

Gas-Feed System

In this system, gas stored at a high pressure is used to pressurize the propellant tank to a pressure large enough to give the desired combustion chamber pressure. Nitrogen has been used as this high pressure gas in the past. For the satellite rocket, helium is recommended because it is lighter in weight for a given volume, and it is much less soluble in liquid oxygen than is nitrogen. The weight of helium (if it is stored at 3000 psi) required to pressurize a tank of volume V_p to a pressure p_p is

$$W_{HE} = 6.45 \times 10^{-6} \; \frac{V_p p_p}{15} \left(1 + \frac{p_p}{3000}\right). \tag{9}$$

This weight of helium plus the weight and size of the helium tanks for each stage may be found graphically in Appendix VI.

Quick stopping and starting of the rocket motors is readily accomplished by the use of the gas-pressure feed system.

Pump-Feed System

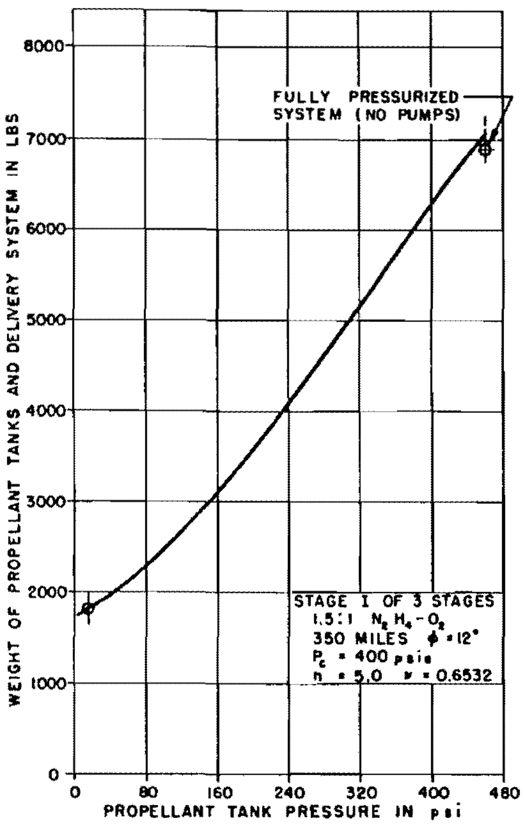
In this system, the propellants are stored at a low pressure (approximately one atmosphere) and a pump raises the propellant pressure to the desired feed value. For most of the propellant flow rates and pressures required in the satellite rocket motors, a high speed, two-stage centrifugal pump directly driven by a gas turbine is indicated as being the lightest pumping system. In order to give the desired quick motor starting, it is necessary to bring the turbine and pumps up to speed prior to the time the propellants are needed. This is possible with centrifugal pumps without a recirculating line, as a centrifugal pump may be operated with its discharge line closed for a short period without overheating the pump. However, a recirculating line will probably be necessary around the oxygen pump, otherwise enough oxygen will boil off during starting to produce a vapor lock. It is also necessary in this system to replace the propellants in the tanks with gas (helium or oxygen) in order to keep the pressure high enough to prevent cavitation at the pump inlet.

Pump-Feed vs Gas-Feed

An investigation was conducted to determine the optimum propellant tank pressure for each of the stages from a weight standpoint (see Appendix VII). The results are given in Figs. 4, 5, and 6. From Figs. 4 and 5 it is readily seen that a pump-feed system with the propellant tank pressure at 15 psi would result in the lowest rocket gross weight in stages I and II. In stage III (Fig. 6), however, the advantage of the pump-feed system over the gas-feed system is very marginal. But in this investigation the tanks were considered spherical and if the actual shapes of the tanks had been used, the fully and partly gas pressurized system weights would have been greater. Therefore, in all three stages of the satellite rocket, a turbine-pump arrangement similar to that used in the German A-4 has been used to raise the propellant tank pressure of 15 psi to this required feed pressure. An exception to this is the control rocket motors in stage III which use a complete gas-feed system having a weight of about 8 pounds, but this weight increase is justifiable in order to simplify the third stage operation.

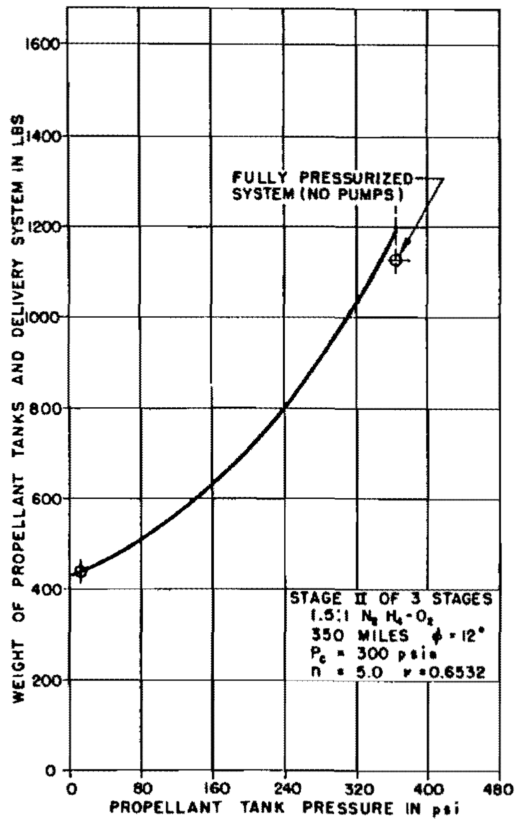
2. Power and Size of Turbine and Pumps

The power and size of the turbine and pumps required in each stage are calculated in Appendix VII. The results are given in Table V.



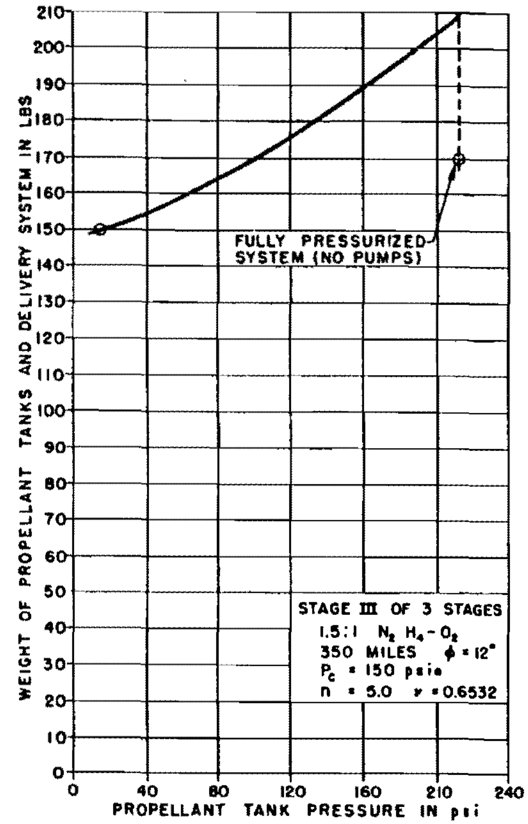
WEIGHT OF PROPELLANT TANKS AND DELIVERY SYSTEM vs PROPELLANT TANK PRESSURE

FIG. 4



WEIGHT OF PROPELLANT TANKS AND DELIVERY SYSTEM vs PROPELLANT TANK PRESSURE

FIG. 5



WEIGHT OF PROPELLANT TANKS AND DELIVERY SYSTEM vs PROPELLANT TANK PRESSURE

FIG. 6

Table V

PUMPS AND TURBINES FOR THE SATELLITE ROCKET

Stage	Hydrazine Centrifugal Pumps				Oxygen Centrifugal Pumps			Turbine Power Output	Overall Length of Turbine and Pumps	Max. Dia. of Unit	Hydrogen-Peroxide Required
	Power Output	Speed	Dia.	Efficiency	Power Output	Speed	Efficiency				
	HP	RPM	in.	%	HP	RPM	%	HP	in.	in.	lbs
I	568 Two Stage	4590	24.8	65	311 Two Stage	4590	61	1384	45	30	429
II	93.7 Two Stage	6720	13.0	53	50.5 Two Stage	6720	47	284.3	26	14	84.4
III	12.2 Single Stage	7760	9.4	57	6.0 Single Stage	7760	50	33.4	20	10	10.6

As is shown in Appendix VI, the diameters were determined by using a similarity relation between the desired centrifugal pump and an existing centrifugal pump. A specific speed was chosen for each pump which would result in high efficiency without making the actual speed excessive. The actual speeds and the efficiencies were then determined from these specific speeds and the desired capacities and pressure heads. In stages I and II a two-stage pump is necessary for both propellants, but in stage III a single-stage pump can give the desired head. The turbine and pumps in each stage were designed to operate at the same speed, making the use of reduction gears unnecessary.

The oxygen pump bearings may be lubricated by liquid oxygen. If the hydrazine pump bearings cannot be lubricated by the hydrazine itself, a grease must be found which will not react with hydrazine. Hydrazine is known to dissolve sulphur which is present in small quantities in most greases.

In stages I and II the turbine, besides driving the pumps, will drive a generator which will provide power for the servo-motors which position the rocket control motors. In stage III this power will come from lightweight, short duration batteries. A separate auxiliary power plant will supply power for the instruments and telemetering equipment during the entire trajectory.

3. Turbine Fuel Requirements

A 90% solution of hydrogen-peroxide is suggested as the fuel in the gas generator for the turbine. This propellant has a heat content of 1137 BTU/lb and a combustion temperature of 1786°R[7] which is about the maximum allowable temperature for present turbine materials. However, as the operational life of the turbines will be of the order of one or two minutes, higher temperatures and stresses may be allowed than in a turbine designed for long duration operation. On the other hand, inasmuch as the weight of turbine fuel required is only about 7/1000 of the weight of the propellants, any saving of turbine fuel weight due to higher energy fuels, and consequently higher turbine temperatures, will not affect the gross weight of the vehicle appreciably. Though research on the turbine blade metals to enable higher reaction temperatures, as well as research to find fuels with high specific impulses at a given temperature is desirable, it is not imperative. A mono-propellant, such as hydrogen-peroxide, is preferable to a fuel requiring an oxidizer and possibly a coolant additive, in view of the increased difficulty of precisely governing the turbine speed when two or more propellants have to be regulated. Furthermore, the products of the peroxide reaction are all gaseous and hence there is little possibility of eroding the turbine blades with solid particles. With the turbine power required, the efficiencies, and the heat content of the turbine fuel all known, the weight of turbine fuel required may be found and is given in Table V.

4. Heat Exchanger to Maintain Oxygen Tank Pressure in Stage I at One Atmosphere

The weight of the helium and helium tank can be reduced in stage I by the use of a heat exchanger (similar to the one used on the A-4) to vaporize a portion of the oxygen to keep the oxygen tank at a pressure of one atmosphere. About 1/258 of the liquid oxygen flow rate will have to be bled through the heat exchange and led to the main oxygen tank as gaseous oxygen. An examination of the use of heat exchangers for stages II and III indicated that the saving in helium and helium tank weights is so small that in these stages the extra complexity due to heat exchangers was not warranted.

D. METHOD OF OPERATION OF THE ROCKET POWER PLANT

In a liquid rocket type of heat engine, the moving parts consist only of the propellants themselves and the valves and pumps which control their movement. The prime requisites are that the initiation of flow be prompt and reliable, that the flow continue steady and uninterrupted during the entire burning period, and that the cessation of flow be sharply defined so there will not be a rough explosive stop. These objectives, plus the varied requirements in each stage of the satellite rocket, governed the design of the fuel systems shown in Figs. 7, 8, and 9.

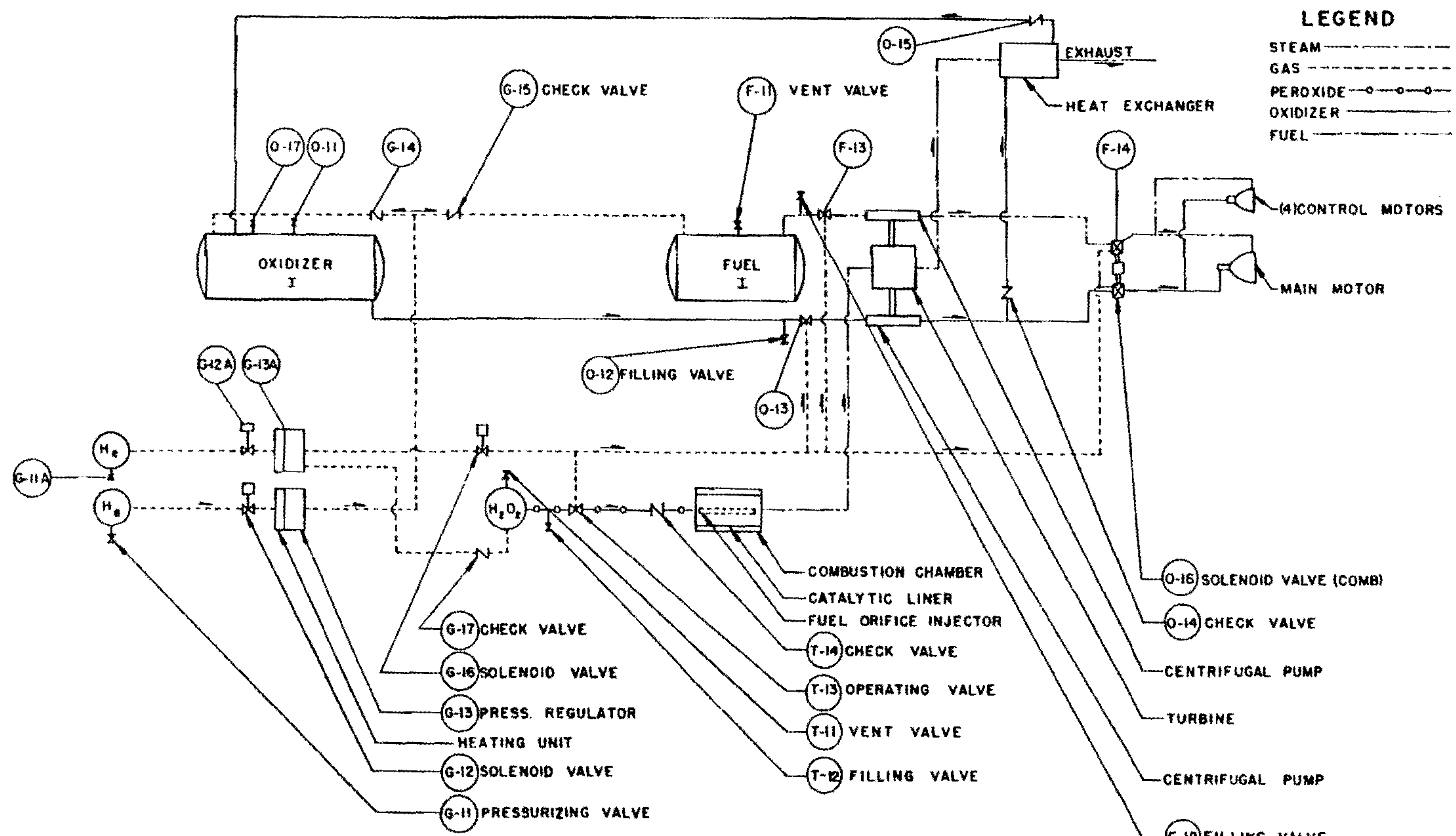
1. Stage I Method of Operation

Fig. 7 is a schematic diagram of the stage I fuel system. The filling and operation of this system is described below:

Filling

Fuel Tank - F-11, a hand-operated vent valve and F-12, a hand-operated filling valve, are opened. The fuel is forced into the tank by a pump in the ground servicing equipment. Upon completion of the filling operation, F-11 and F-12 are closed.

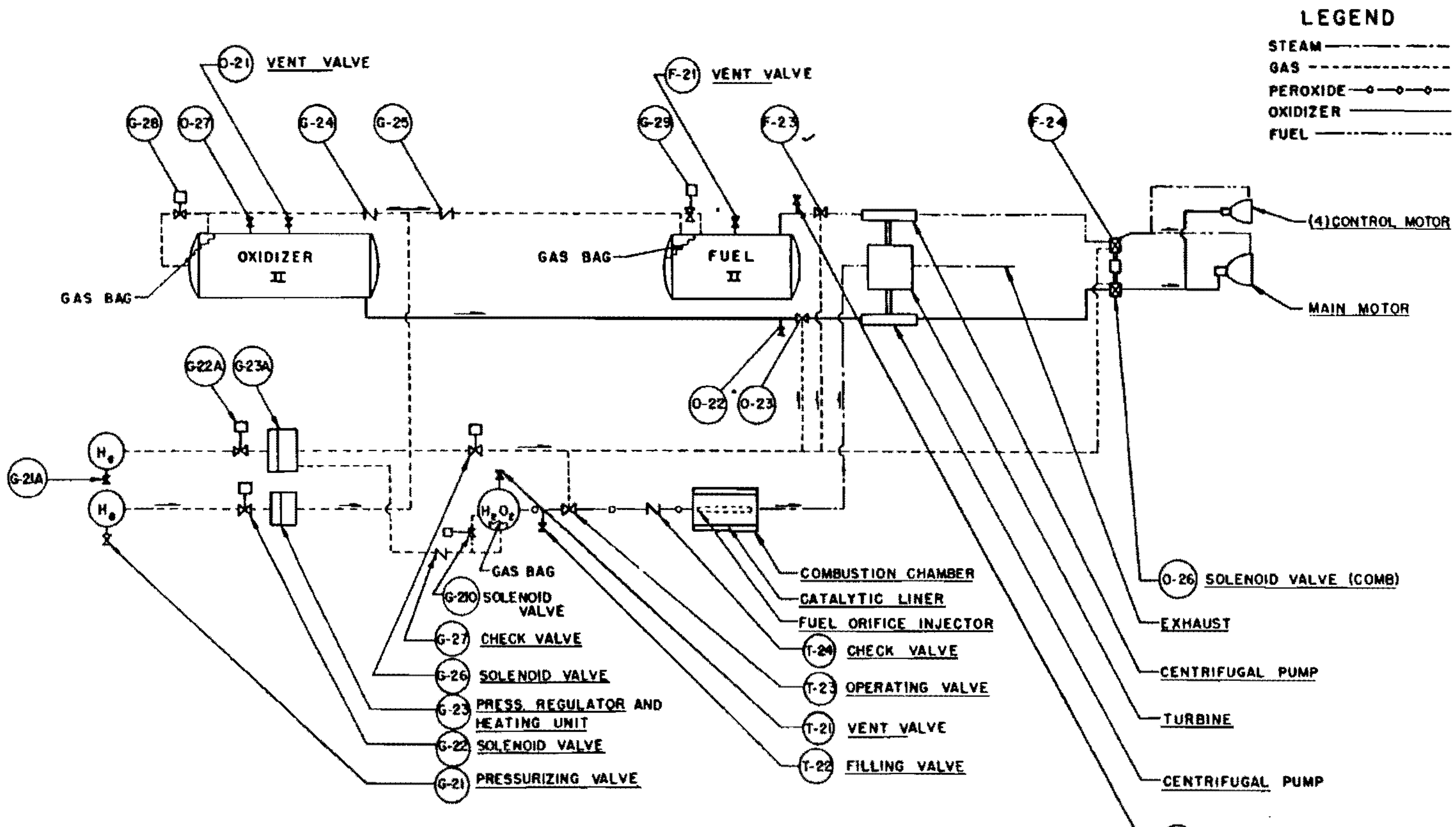
Oxidizer Tank - O-11, a hand-operated vent valve and O-12, a hand-operated filling valve, are opened. The oxidizer is forced into the tank by a pump in the ground servicing equipment. Upon completion of the filling operation, O-11 and O-12 are closed.



FUEL SYSTEM DIAGRAM FOR THREE-STAGE SATELLITE "STAGE NO. I"

FIG. 7

UNCLASSIFIED

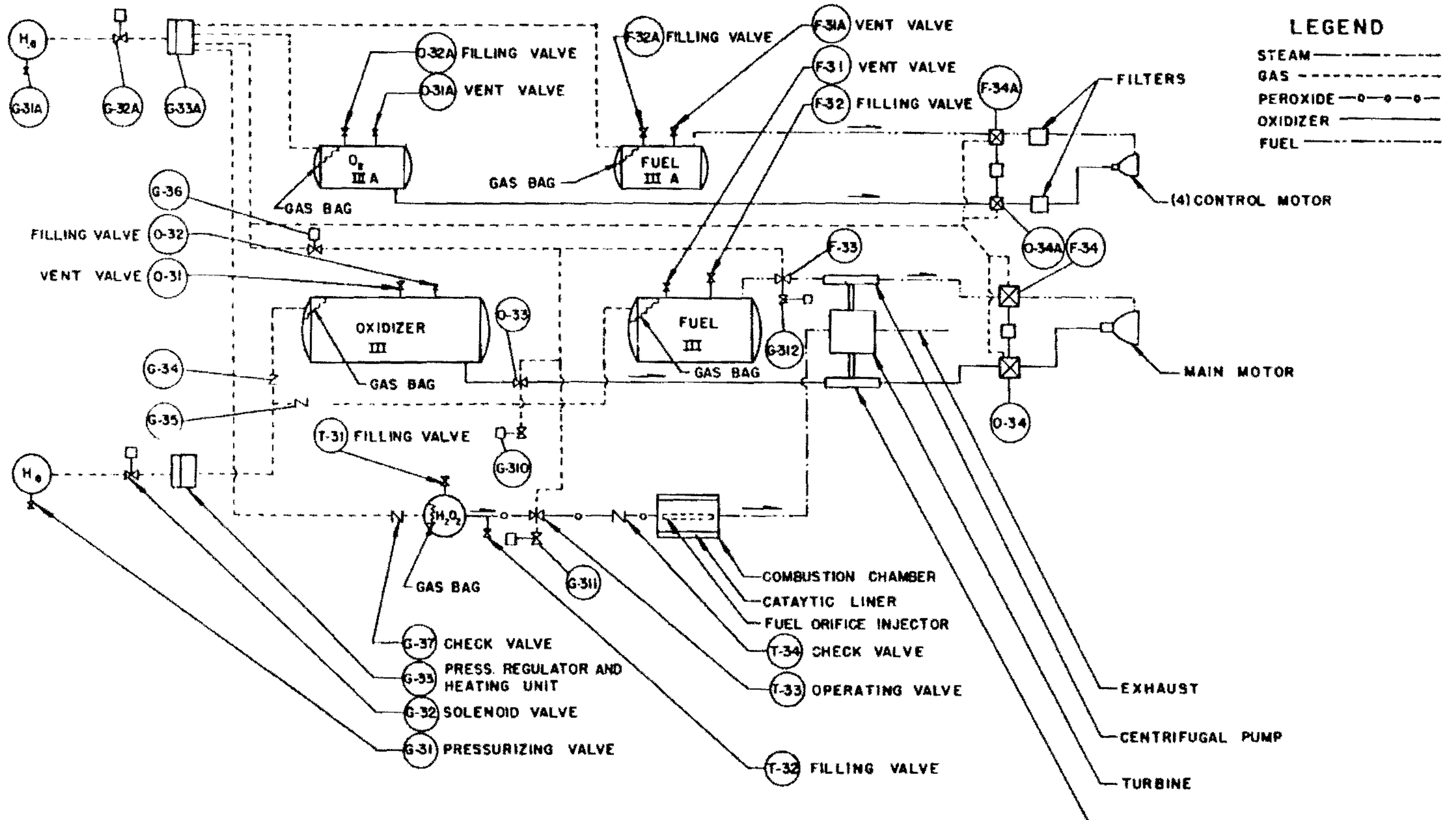


FUEL SYSTEM DIAGRAM FOR THREE-STAGE SATELLITE "STAGE NO. II"

FIG. 8

UNCLASSIFIED

February 1, 1947



FUEL SYSTEM DIAGRAM FOR THREE-STAGE SATELLITE "STAGE NO. III"

FIG. 9

UNCLASSIFIED

Turbine Fuel Tank - T-11, a hand-operated vent valve and T-12, a hand-operated filling valve, are opened. The turbine fuel is forced into the tank by a pump in the ground servicing equipment. Upon completion of the filling operation, T-11 and T-12 are closed.

Gas Pressure Tanks - The gas tanks are pressurized by ground servicing equipment through valves G-11 and G-11A.

Operation

G-12 and G-12A, solenoid valves, are opened allowing high pressure gas to flow from the gas tank into G-13 and G-13A, pressure reducing and regulator valves. Gas at various lower pressures is taken from G-13 and G-13A in three lines.

The line from G-13 leads directly to the fuel and oxidizer tanks in order to maintain a certain minimum pressure within these tanks throughout operation. Check valves, G-14 and G-15, are in the lines to prevent any gaseous fuel and oxidizer from coming together in the gas pressure lines. At the end of stage I operation the helium supply pressure for G-13 is reduced to about 20 psi.

One line from G-13A leads through check valve G-17 to pressurize the turbine fuel tank throughout operation. A second line from G-13A supplies pressure to the pneumatically operated valves in the system. At the end of stage I operation the helium supply pressure for G-13A is reduced only to about 420 psi. G-16, a solenoid valve, is opened allowing the gas to open pneumatic valves T-13, O-13, F-13, and to operate the electrically controlled valves O-16 and F-14.

When T-13 is opened the pressurized turbine fuel flows through check valve T-14 into the turbine combustion chamber, where it is sprayed against the catalyst-lined walls of the chamber. The hot gaseous products of combustion operate the turbine for driving the fuel and oxidizer pumps. T-14 prevents the products of combustion from flowing back into the turbine fuel tank. After being exhausted from the turbine the hot gases pass through the heat exchanger to the atmosphere.

When F-13 is opened, gravity plus the one atmosphere pressure of helium above the fuel forces the fuel into the centrifugal fuel pump. The fuel is then pumped through F-14 into the cooling jackets and the combustion chambers of the motors. F-14 is an electrically controlled, pneumatically operated valve used to vary the fuel flow during the starting operation on the ground.

When O-13 is open, the oxidizer flows into the centrifugal oxidizer pump and is pumped into two lines. One of these lines leads through O-16 to the combustion chamber. O-16 is an electrically controlled, pneumatically operated valve used to vary the oxidizer flow during the starting operation. A small oxidizer line leads through check valve O-14 into the heat exchanger, where the hot turbine exhaust gases transform the oxidizer to the gaseous state. O-14 prevents the gaseous oxidizer from flowing back into the line leading to the motor. The gaseous oxidizer is led from the heat exchanger through check valve O-15 back to pressurize the oxidizer tank. O-15 prevents the initial pressurizing gas of the pressure regulator from escaping out of the oxidizer tank before the turbine begins to operate. O-17 is a pressure relief valve preventing the pressure in the oxidizer tank from reaching excessively high values. The motors in stage I are allowed to run for about 5 seconds at 1/3 thrust. Then, if the turbine and the five motors are operating properly and there are no leaks, combination valve O-16 and F-14 is opened to give full thrust and take-off commences. No allowance has been made in the satellite gross weight for the propellant consumption during this warm-up, but since this weight is consumed before take-off it will not influence the trajectory calculations. It only means that the stage I propellant tanks must be about 5% larger than is shown in Fig. 2. The power plant is now allowed to run a given length of time based on a constant fuel flow rate and a given initial amount of fuel, the values of which are determined from trajectory considerations. At the end of the pre-determined powered-flight time, oxidizer valve O-16 is closed and combustion is precisely ended. The first stage is then separated from the second and third stages.

2. Stage II Method of Operation

Filling

Fuel Tank - F-21, a hand-operated vent valve and F-22, a hand-operated filling valve, are opened. The fuel is forced into the tank by a pump in the ground servicing equipment. Upon completion of the filling operation, F-21 and F-22 are closed.

Oxidizer Tank - O-21, a hand-operated vent valve and O-22, a hand-operated filling valve, are opened. The oxidizer is forced into the tank by a pump in the ground servicing equipment. Upon completion of the filling operation O-21 and O-22 are closed.

Turbine Fuel Tank - T-21, a hand-operated vent valve and T-22, a hand-operated filling valve, are opened. The turbine fuel is forced into the tank by a pump in the ground servicing equipment. Upon completion of the filling operation T-21 and T-22 are closed.

Gas Pressure Tank - The gas tanks are pressurized by ground servicing equipment through valves G-21 and G-21A.

Operation

G-22 and G-22A, solenoid valves, are opened allowing the high pressure gas to flow from the gas tank into G-23 and G-23A, pressure reducing and regulating valves. Gas at various lower pressures is taken from G-23 and G-23A in three lines.

The line from G-23 leads to the fuel and oxidizer tanks through solenoid valves G-29 and G-28 in order to maintain a certain minimum pressure within those tanks throughout operation. Check valves G-24 and G-25 are in the lines to prevent any gaseous fuel and oxidizer from coming together in the gas pressure lines.

A unique condition exists at the beginning of stages II and III. When the vehicle is coasting, the propellants and propellant tanks have zero apparent gravity. Therefore, if helium is applied to the top of the tank it is not certain whether helium or propellant will enter the pump feed line. To make sure that propellant will enter this feed line a small gas bag is incorporated in the fuel and oxidizer tanks. Gas from G-23 is fed directly to these bags which expand and take up any excess volume in the tank and thus force the fuels into the fuel lines leading to the rocket motor as far as pneumatically operated valves O-23 and F-23.

This same condition exists in the case of the turbine fuel tanks. Though the turbine is started for stage II during the latter part of burning of stage I, at the end of burning of stage I the turbine fuel and fuel tank of stage II are also being acted upon only by gravity, and there is a possibility that the pressurizing gas might enter the turbine fuel feed line and hence stop the turbine. For this reason it has been necessary to incorporate a small gas bag in the turbine fuel tank. Thus gas from G-23A is fed directly from check valve G-27 to the gas bag in the turbine fuel tank until the stage II motors begin to produce thrust, after which solenoid valve G-210 is opened allowing the gas to pressurize the turbine fuel directly.

A second line from G-23A supplies pressure to the pneumatically operated valves in the system.

G-26, a solenoid valve, is opened allowing the gas to open pneumatic valves T-23, O-23, and F-23. At this time solenoid valves G-28 and G-29 are opened, allowing the gas from G-23 to pressurize the oxidizer and fuel tanks directly. When T-23 is opened, the pressurized turbine fuel flows into the turbine combustion chamber where it is sprayed against the catalyst-lined walls of the chamber. The hot gaseous products of combustion operate the turbine for driving the fuel and oxidizer pumps. Check valve T-24 prevents the products of combustion from flowing back into the turbine fuel tank. After being exhausted from the turbine the hot gases pass directly to the atmosphere.

When F-23 is opened, the pressurized fuel flows into the centrifugal fuel pump and is pumped to the electrically controlled pneumatically operated valve F-24. Similarly, when O-23 is opened, the pressurized oxidizer flows into the centrifugal oxidizer pump and is pumped to O-26. The fuel and oxidizer are stopped at F-24 and O-26 in order that the turbine may develop its full power during stage separation without having propellant combustion in the rocket motor.

F-24 and O-26, which are used to regulate the fuel and oxidizer flow during the starting operation, are now opened and fuel and oxidizer are admitted to the combustion chamber.

The power plant is allowed to run a given length of time based on constant fuel flow rate and a given initial amount of fuel which values are determined from trajectory considerations.

At the end of the predetermined powered-flight time, oxidizer valve O-26 is closed and combustion is precisely ended. The second stage is then separated from the third stage.

3. Stage III Method of Operation

Filling

Fuel Tank - F-31, a hand-operated vent valve and F-32, a hand operated filling valve, are opened. The fuel is forced into the tank by a pump in the ground servicing equipment. Upon completion of the filling operation, F-31 and F-32 are closed.

Oxidizer Tank - O-31, a hand-operated vent valve and O-32, a hand-operated filling valve, are opened. The oxidizer is forced into the tank by a pump in the ground servicing equipment. Upon completion of the filling operation O-31 and O-32 are closed.

Auxiliary Fuel Tank - F-31A, a hand-operated vent valve and F-32A, a hand-operated filling valve, are opened. The fuel is forced into the tank by a pump in the ground servicing equipment. Upon completion of the filling operation, F-31A and F-32A are closed.

Auxiliary Oxidizer Tank - O-31A, a hand-operated vent valve and O-32A, a hand operated filling valve, are opened. The oxidizer is forced into the tank by a pump in the ground servicing equipment. Upon completion of the filling operation, O-31A and O-32A are closed.

Turbine Fuel Tank - T-31, a hand-operated vent valve and T-32, a hand operated filling valve, are opened. The turbine fuel is forced into the tank by a pump in the ground servicing equipment. Upon completion of the filling operation, T-31 and T-32 are closed.

Gas Pressure Tanks - The gas tanks are pressurized by ground servicing equipment through valves G-31 and G-31A.

Operation

In this case a separate fuel system is provided for the main rocket motor and another for the control rocket motors. They will be discussed separately.

Main Rocket Motor - G-32 and G-32A, solenoid valves are opened, allowing the high pressure gas to flow from the gas tanks into G-33 and G-33A pressure reducing and regulating valves. Gas at various lower pressures is taken from G-33 and G-33A in five lines. The line from G-33 leads to the main fuel and oxidizer tanks through solenoid valves G-38 and G-39 in order to maintain a certain minimum pressure within these tanks throughout operation. Check valves G-34, G-35 are in the lines to prevent any gaseous fuel and oxidizer from coming together in the gas pressure lines.

Inasmuch as this power plant is also required to start when the vehicle is not accelerating as well as restart after a period of coasting, gas bags are incorporated in the fuel and oxidizer tanks. However, these bags differ from those in stage II in that they are sufficiently large when expanded to fill the whole volume of each tank and the propellants are never pressurized by the gas directly. Gas from G-33 is fed directly to these bags which expand and take up any excess volume in the tank and thus force the fuels into the fuel lines leading to the rocket motor as far as the pneumatically operated valves O-33 and F-33.

Again the turbine fuel tank requires a gas bag for the same reasons as discussed in stage II; however, as above, this gas bag is also large enough when expanded to fill the whole volume of the turbine fuel tank and the turbine fuel is never pressurized by the gas directly.

One line from G-33A leads from the pressure regulator through check valve G-37 to pressurize the turbine fuel tank throughout operation.

A second line from G-33A supplies pressure to the pneumatically operated valves in the system.

A third line from G-33A leads to the auxiliary fuel tank IIIA to pressurize it sufficiently to be operated as a component of the pressurized fuel system used for the control rocket motors.

A fourth line from G-33A leads to the auxiliary oxidizer tank IIIA to enable it to operate as the other component of the pressurized fuel system mentioned above.

When this power plant is to be started, G-36, a solenoid valve, is opened allowing gas to open T-33, F-33, and O-33. When T-33 is opened, the pressurized turbine fuel flows into the turbine combustion chamber where it is sprayed against the catalyst-lined walls of the chamber. The hot gaseous products of combustion operate the turbine for driving the fuel and oxidizer pumps to supply fuel and oxidizer to the main rocket motor only. Check valve T-34 prevents the products of combustion from flowing back into the turbine fuel tank. After being exhausted from the turbine, the hot gases pass directly to the atmosphere.

When F-33 is opened, the pressurized fuel flows into the centrifugal fuel pump and is pumped to the electrically controlled pneumatically operated valve F-34. Similarly when O-33 is opened the pressurized oxidizer flows into the centrifugal oxidizer pump and is pumped to the electrically controlled, pneumatically operated valve O-34. The fuel and oxidizer are stopped at F-34 and O-34 in order that the turbine may develop full power during stage separation without having propellant combustion in the rocket motor. Solenoid-controlled valves F-34 and O-34 which are used to regulate the fuel and oxidizer flow are now opened and fuel and oxidizer are admitted to the main motor and combustion starts.

It is now desired that the main rocket motor be stopped. O-34 is closed, stopping the flow of oxidizer to the motor. The turbine is permitted to run for a few seconds to pump fuel through the cooling jacket of the motor to absorb the heat from the last part of the combustion. Then F-34 is closed, stopping the flow of fuel to the motor.

Solenoid G-36 is closed, shutting off gas supply to pneumatic valves T-33, F-33, and O-33. Solenoid valves G-310, G-311, and G-312 are opened, permitting gas pressure to escape from O-33, T-33, and F-33, respectively, and thus close those valves, and the plant is shut down.

After a period of coasting it is desired that the main rocket motor be started again. This is accomplished as before, starting with the opening of solenoid valve G-36.

The power plant is now allowed to run a given length of time based on constant fuel flow rate and a given initial amount of fuel, which values are determined from trajectory consideration.

Due to the undesirable effect of a rocket motor explosion in this stage, because of unabsorbed heat at the end of combustion, the power plant will be completely shut down in the same manner as described in the first part of stage III and not similarly to stage I and II.

Control Rocket Motors - G-32A, solenoid valve, is opened, allowing the high pressure gas to flow from the gas tank into G-33A, a pressure reducing and regulating valve. For the operation of the control motors, gas is taken from G-33A in three lines. One line leads directly to the auxiliary fuel tank to pressure it sufficiently to be operated as a component of the pressurized fuel system. A second line leads to the auxiliary oxidizer tank to enable it to operate as the other component of the pressurized fuel system mentioned above. A third line supplies pressure to the pneumatically operated valves in the system.

As these motors are required to start when the rocket vehicle is not accelerating, gas bags are incorporated in the fuel and oxidizer tanks. Gas from G-33A is fed directly to these bags which expand and take up any excess volume in the tank and thus force the fuels into the fuel lines leading to the rocket motors as far as the electrically controlled, pneumatically operated valves F-34A and O-34A. When it is desired that these motors be started, F-34A and O-34A, simultaneously with F-34 and O-34, are opened and the fuel and oxidizer are admitted to the combustion chamber through filters and combustion starts. Filters are necessary in this stage because of the small size of the control rocket motors and the consequent small size of the injectors with their inherent danger of clogging.

F-34A and O-34A are used to regulate the flow of fuel and oxidizer to the motors during the starting operation. These motors are allowed to run a given length of time based on constant fuel flow rate and a given initial amount of fuel, values of which are determined from trajectory considerations. These control motors will operate constantly during the two powered portions, as well as the coasting portion of stage III. To stop the motors O-34A is closed, stopping the flow of oxidizer. Fuel is permitted to continue to flow through the cooling jacket and thus absorb the heat from the last part of the combustion. Then F-34A is closed, stopping the flow of fuel to the motors.

V. CONCLUSIONS AND RECOMMENDATIONS

Rocket power plants can be built which will propel a staged rocket vehicle to a satellite orbit at an altitude of 350 miles. The power plants recommended here consist of five regeneratively cooled liquid rockets in each of the three stages. The fuel used is liquid oxygen and hydrazine. The steering of the vehicle is accomplished by the pivot mounting and control of the four smaller rocket motors in each stage. All rocket motors are fed by centrifugal pumps, except the stage III control motors which are pressure fed with helium gas. The products of decomposition of 90% hydrogen-peroxide operate a gas turbine which directly drives both propellant pumps. A small amount of research plus the usual experimental development work necessary for any rocket power plant will be required before the final satellite power plant is constructed. The prime unknowns which must be determined are:

1. The heat flow density from a chamber containing the high temperature products of combustion of hydrazine and liquid oxygen to the hydrazine in the coolant jacket.

2. The optimum combustion chamber volume to throat area ratio (L^*) for the hydrazine-liquid oxygen system (Alternatively the chemical reaction rate of these propellants.)

3. The optimum nozzle length for each motor with η_V, friction losses, nozzle weight, and cooling requirements taken into consideration.

4. The optimum injector design based on the maximum combustion efficiency and the minimum heat flux to the motor shell.

PART B.

AUXILIARY POWER PLANT

I. SUMMARY

An auxiliary power plant is necessary in the satellite rocket to provide electrical power for the operation of attitude control motors, data taking, radio contact, and telemetering while the satellite is on its orbit and when the main rocket motors are not operating. As essentially the same equipment will be used for data taking, radio contact, and telemetering both on the ascending trajectory and on the orbit, it was decided that the auxiliary power plant would provide the power required for these functions during the ascent as well as on the orbit. In addition, this power plant will also energize the attitude control mechanism for the orbital flight.

A preliminary estimate of the power needed indicates 300 watts will be necessary, and this figure is used in this study.

This study reveals that it is possible to construct a 300 watt power plant to operate for an estimated required period, two weeks, within the allowable weight of 400 pounds. The power plant to meet these conditions is a radioactive cell (heat source), a low pressure mercury vapor turbine condensing system with an electrical generator, and a mercury vapor radiant condenser. (Cooling is accomplished by radiation as there is no atmosphere to provide conduction or convection at the orbit altitude of 350 miles.)

The total weight of this system is estimated to be 157 pounds. Therefore, if it is thought desirable, the watt output of this system can be more than doubled and the weight will still be below the allowable 400 pounds. Increasing the watt output is beneficial as it increases the size of the turbine to be used, which is critically small for 300 watts. Further, it will provide more power to accomplish the various requirements given above, and therefore simplify the equipment design problems.

There is an alternative power plant which is within the weight limit which can be used for auxiliary power; however, it will not provide power for the full two week period, but only for about 5½ days. This is the system wherein a gas turbine is driven by the products of combustion of a chemical fuel at relatively high pressure.

To avoid difficulties in this case, with an extremely small turbine required for continuous power output, a duty cycle is suggested in which power at a reasonable level is generated for a short period and stored in a battery. Power requirements are taken from the battery and when the battery reserve becomes low, the generator is started again.

II. INTRODUCTION

During the ascending trajectory and on the orbital path auxiliary power is required for satellite attitude control, radio contact, data taking, and telemetering. Exact power requirements for these functions are at present unknown, however, qualitatively the requirements are as follows:

A. ASCENDING TRAJECTORY

1. **Power for Data Taking Instruments**
   a. To provide information on satellite rocket performance
   b. To provide scientific information of interest

2. **Power for Communication**
   a. Radio beacons and receiver
   b. Telemetering
   c. Doppler altimeter

3. **Power for Satellite Attitude Control Mechanisms**
   a. Servo-indicator (gyroscopic)
   b. Servo-amplifier
   c. Servo-motors which position the rocket control motors

B. ORBITAL PATH

1. **Power for Data Taking Instruments**
   a. To provide information on satellite rocket attitude
   b. To provide scientific information of interest

2. **Communication**
   a. Radio beacon and receiver
   b. Telemetering
   (These are the same instruments used in the ascending trajectory)

3. **Power for Satellite Attitude Control Mechanism**
   a. Five molecular beam detectors to indicate pitch and yaw[5]
   b. Electro-magnetic compass to indicate roll
   c. Device for indicating the variation of a and b and computing the correct signal to be sent to the flywheel controls
   d. Motors for rotating the flywheel controls.

During stages I and II of the satellite rocket an electrical generator coupled to the main fuel pump turbine will supply all the power for the control mechanisms. In stage III a light weight, short duration battery will supply the power for the control mechanisms as well as for the Doppler altimeter which is used in the final powered portion of the trajectory after coasting to determine when altitude is constant. A separate auxiliary power plant will be used to supply power for the communication devices and data taking instruments during both the ascending trajectory and orbital flight, and for the control mechanisms during orbital flight.

A preliminary estimate of the power which this auxiliary power plant will be required to produce is 300 watts, if satellite rocket attitude is to be controlled continuously and if data are to be taken and transmitted to the earth continuously. This requirement of 300 watts is taken as a basis for evaluating the various systems considered.

The auxiliary power plant will be required to supply the necessary power for a long duration (on the order of two weeks), in practically a vacuum, with no gravity or acceleration forces available, and have a very small total weight (including fuel). It is readily seen that no conventional power source can meet these severe conditions. As a result, a number of unconventional power sources were investigated. These include a solar power plant, a high power electro-magnetic ground beam, an atomic pile, a radioactive material, and chemical fuels. To make the energy from these sources usable it is necessary to convert it to electrical power. Therefore, these energy sources have been investigated for use with a turbine-generator and a thermopile.

III. ANALYSIS AND DISCUSSION

A. SOLAR POWER PLANT

The most obvious source of auxiliary power for the satellite rocket is the sun itself. At the orbital altitude of 350 miles, practically none of the radiant energy of the sun is absorbed by the earth's atmosphere. However, the investigation made in Appendix VIII shows that the entire projected area of the stage III satellite rocket would intercept only enough radiant energy to supply 242 watts of useful electric power. The present estimated power requirements for data taking, control, and telemetering is about 300 watts. But even if the power requirements are later reduced to 242 watts, a solar absorber covering one side of the rocket vehicle would be very heavy. To this weight must be added the weight of the working fluid, the boiler-feed pump, the turbine, and a large condenser on the shady side of the vehicle. The physical size of the solar absorber might be reduced by installing a large condensing lens on the sunny side of the vehicle, but it is unlikely that this would result in any overall weight reduction. Therefore, the solar power plant must be discarded because of its inherently large weight.

B. HIGH ENERGY GROUND TRANSMITTERS

Another possibility for supplying power for the satellite is to transmit high-frequency electro-magnetic waves from a series of ground stations spaced around the earth. An antenna on the satellite would intercept these waves producing an electric current. The order of magnitude of the power required by fourteen stations, located at the equator, equally spaced about the earth, and which would transmit 300 watts to the satellite with an orbital altitude of 350 miles, is now calculated. The maximum distance from the vehicle to any ground station is approximately

$$R = \sqrt{(350)^2 + \left(\frac{25000}{28}\right)^2}$$

$$R = 1008 \text{ miles} \quad .$$

The power output required from each of the ground stations in watts is

$$P = \frac{4\pi R^2 S}{G \times A_{eff}}$$

where S = power required by satellite = 300 watts

G = gain of antenna over isotropic radiation = 10^5

A_{eff} = effective area of receiver antenna = 10 square feet.

Therefore

$$P = \frac{4\pi\,(5280 \times 1008)^2\ 300}{10^5 \times 10}$$

$$= 10.7 \times 10^{10} \text{ watts or } 107{,}000 \text{ megawatts.}$$

This excessive power requirement of 107,000 megawatts at each station excludes the use of high energy ground beams for this purpose.

C. CHEMICAL FUEL-GAS TURBINE SYSTEM

1. Long, Continuous Operation System

The orbital power plant requiring the least amount of development is a chemical fuel-gas turbine system similar to the system used for driving the pumps for the rocket motors of the satellite vehicle. The decomposition products of hydrogen-peroxide drive a turbine which drives an electrical generator instead of pumps. The drawback to this system is the large weight of fuel required to operate the generator for a long duration. If the generator is to supply 300 watts continuously for two weeks, the total power plant weight would be 1455 pounds (see calculation 1 of Appendix IX). This is larger than the combined auxiliary power plant and payload allotment in the present design satellite vehicle. This weight might be reduced a few per cent if a higher energy propellant system which will give the same combustion chamber temperature were used, but the total weight would still be excessive. A higher temperature than the 1786°R given by the decomposition of 90% H_2O_2 cannot be tolerated by present turbine materials for a long duration even though a partial admission turbine is used. The difficulties in building the watch-pocket size turbine required with a high efficiency would be very great. It is probable that a small reciprocating engine which would supply the 0.47 horsepower required could be more readily made. Also, a propellant combination with a higher chamber temperature could be tolerated in a reciprocating engine thereby lowering the total fuel weight.

However, a miniature reciprocating engine with its large number of moving parts is more likely to have a failure over a two week period than a gas turbine with its one moving part.

2. Light-Weight Intermittent Operation System

As an alternative to overcome small turbine size and large total fuel weight difficulties, an intermittent power generation and telemetering cycle has been set up (calculation 2 of Appendix IX) which includes a battery for storage of power. With this duty cycle and a total power plant weight limited to 400 pounds, the duration of contact between the earth and the satellite could continue for 136.7 hours and a 5 1/2 horsepower turbine could be used. If a stand-by turbine-generator is also included in the 400 pounds (for greater reliability) the contact duration is reduced to 113.8 hours.

3. General Method of Operation

At the time of take-off, a valve would be opened allowing helium gas to pressurize the turbine fuel tank. The hydrogen-peroxide would flow to the combustion chamber where an injector sprays the peroxide on potassium-permanganate-impregnated walls. The permanganate is a catalyst and therefore is not consumed in the reaction. But for long duration operation the momentum of the hot gases striking the wall can be expected to carry some of the catalyst out the exhaust. It is therefore necessary to intermittently pressurize a small container of the catalyst and thereby replenish the amount of catalyst in the walls.

The combustion gases then pass to the turbine through a nozzle diaphragm, whose area is only a small fraction of the total area existing between the turbine buckets. A partial admission turbine such as this makes possible a small but still efficient turbine and also helps cool the turbine blades.

The direct current generator which is driven by the turbine, feeds electrical power to a storage battery or directly to the control mechanisms and electronic devices.

D. ATOMIC ENERGY POWER PLANT

1. Pile of Fissionable Material

The main disadvantage of the chemical fuel system described above is that a long duration power plant requires too large a weight of fuel. A much higher concentration of energy can be obtained from a given weight of material if the atomic energy of the material is utilized. An obvious method of using the energy of the neutron-chain-reaction of fissionable materials is to use a uranium or plutonium pile. The heat from this pile could be used to generate steam or mercury vapor in a boiler and then operate a turbine-generator unit.

However, the use of a pile for the auxiliary power plant has serious disadvantages. The main disadvantage is the inherent inefficiency of a pile used to produce the orbital power requirements of only 300 watts. There must be a minimum (critical) mass of fissionable material present before the unit will operate, whether 300 or 30,000,000 watts are desired. Another disadvantage lies in the penetrating gamma-ray and neutron emission which comes from such an atomic power plant. For use in the satellite rocket, it is very desirable to have this power plant begin operation before take-off so that a complete check-out can be made of the auxiliary power plant, control, and telemetering systems before the satellite leaves the ground. This means that thick walls of shielding material would have to surround the power plant while the crew prepared the rocket for launching.

2. Radioactive Cell

However, if instead of using U-235 or plutonium directly, use is made of one of the radioactive substances which are by-products of the plutonium manufacturing process, all the above disadvantages can be eliminated. The by-product which most closely meets the needs of the orbital power plant, according to a memorandum by RAND consultants Alvarez, McMillan, and Serber, of Berkeley,[11] is radioactive strontium 89. The half-life of Sr 89 is 55 days and it emits beta rays with an upper limit of 1.52 mev. Also, it is believed that an adequate amount of Sr 89 already exists in stored waste products. It would be very difficult to separate Sr from the radioactive barium which emits strong gamma rays with a half-life of 12.5 days. But if the fission products are allowed to cool for 100 days or so, the shorter-lived barium will decay greatly with respect to the Sr, so that the separation can be much more easily effected. Since there is so much activity at any one time in the cooling tanks, it is not necessary to hurry the separation to make sure of getting the strontium out before it has decayed too much.

The weight of strontium 89 required to produce 300 watts for two weeks is calculated in Appendix X and is equal to 3.02 grams. The total weight of the radioactive cell cannot be determined until the per cent concentration of the isotope, which can be obtained at reasonable cost, is known; as well as the per cent dilution which would be desirable to control the temperature when the strontium cell is installed in a power plant.

Two methods are suggested for converting the heat from the strontium cell into electrical energy. These are a boiler, turbine-generator, and condenser - and a hot and cold plate thermopile.

E. RADIOACTIVE CELL - MERCURY VAPOR SYSTEM

1. Description

Inasmuch as there is no cooling medium available in the upper atmosphere, the greatest difficulty encountered in a satellite power plant using any vapor system is the condensing of the vapor after it has expanded in a turbine. The only possible method of condensing in the satellite is with the use of a radiant condenser. Since the rate of heat transfer from a radiating surface is proportional to the fourth power of the temperature, it is imperative that the working fluid condense at a high temperature at the low turbine exhaust pressure. Therefore, the boiling point of the working fluid should be high enough to permit sufficient heat transfer in the condenser and at a low enough temperature-pressure combination to permit light boiler construction with existing materials. Mercury meets these requirements and was therefore chosen as the working fluid. Also, considerable research and development has already been done with mercury boilers and turbines.

In the mercury vapor cycle described in Appendix XI, the mercury is vaporized in the radioactive cell boiler at an absolute pressure of 40 psi and a temperature of 1245° R. The vapor is condensed at 5 psi and 1036° R. A higher cycle efficiency may be obtained by the use of a higher boiler pressure and temperature, but the efficiency is not critical in this strontium cell auxiliary power plant from a weight standpoint. If the cycle efficiency was reduced by 1/2, the 300 watts power required would still only require about 6 grams of strontium 89. On the other hand, operating the mercury cycle at a low pressure has the following advantages:

1. Lower pressure reduces the weight of boiler tubes, piping, turbine casing, and boiler feed pump.

2. Lower pressure results in a higher specific volume at the turbine inlet. This is very important when designing a turbine small enough to operate on one or two pounds of flow per minute.

3. The relatively lower operation temperature increases the duration of turbine operation before failure can be expected.

The rate of heat released from the strontium cell cannot be stopped or even varied, therefore, it is desirable to start the turbine-generator system when the strontium cell and boiler coil are assembled so that the temperature of the strontium cell may be maintained at some allowable value. Therefore, the auxiliary power plant will operate on the ground before launching and during the ascending trajectory, as well as during the orbital flight. The problem of keeping the condenser cool on the ground can be solved simply by some cooling device external to the satellite itself. But on the ascending trajectory the solution is not as simple. The condenser cannot be located adjacent to the external skin of the stage III rocket because this skin reaches temperatures of about 1300°R due to friction in the lower atmosphere. But at the same time, if the condenser is to radiate the necessary amount of heat during the orbital flight, it must be located on an external surface of the stage III vehicle. Therefore, the condenser is located around the motor compartment of stage III. The condenser will then be protected by the fuel tank of stage II in the lower atmosphere and will be exposed to space during the stage III trajectory and during orbital flight. During the ascending trajectory a cake of solid carbon dioxide can be spring-fed against an area at the exit of the condenser coil at a rate which will condense but not freeze the mercury. Making this conservative assumption, that the radiant condenser will not operate until the satellite is on the orbit after a fourteen-minute trajectory, it will theoretically require about 9 pounds of carbon dioxide to condense the mercury vapor assuming no heat addition except that released by the condensing mercury vapor. The condenser consists of a corrugated 0.03-inch-thick steel plate welded to a flat rectangular 0.03-inch-thick steel plate mounted in the motor compartment as shown in Fig. 1. When the satellite rocket is travelling in its orbit, this condenser plate will be oriented parallel to the earth satellite radius vector with the corrugations facing spaceward to obtain maximum heat dissipation. A sheet of highly polished aluminum is located between the condenser and the stage III rocket motors in order to intercept the heat radiated from the stage III rocket motors during their operation. The mechanics of vapor and fluid flow in a gravitationless condenser are not fully understood at the present time. However, it is believed that the low adhesion and high surface tension properties of mercury will facilitate the flow from the condenser inlet to the boiler feed pump inlet. Furthermore, inasmuch as the flow cycle is started on the ground when gravity is present, it is believed that flow will continue later in the absence of gravity, although restarting while on the orbit is probably not possible. The flow rate of mercury, the heat which must be radiated from the condenser, and the design of the condenser are given in Appendix XI. The weight of this condenser and the estimated weights of the other units in the radioactive-boiler, mercury-vapor system which will produce 300 watts are given below:

	Weight in pounds
Mercury condenser	40
Mercury (working fluid)	50
Mercury turbine	15
Electrical generator	12
Boiler feed pump and motor drive	5
Carbon dioxide cake and feed mechanism	15
Radioactive boiler (includes boiler tubes and strontium cell)	20
Total weight of the radioactive cell and mercury vapor system	157 pounds

2. **Method of Operation**

Immediately after the assembly of the strontium cell and the mercury boiler, the boiler feed pump driven by a battery in the ground servicing unit will be started. The entire auxiliary power plant may then be checked out by the ground crew. Also, all the control mechanism and telemetering equipment may be checked out at the same time. As soon as the mercury turbine acquires full speed the boiler feed pump motor will be driven directly by power from the system's electric generator. The condenser may be cooled by a water coil or carbon dioxide blocks during this ground operation. As soon as the ground checks and adjustments have been made and the trajectory block of carbon dioxide has been placed in its feed mechanism, the auxiliary power plant is ready for satellite take-off. During the trajectory after stage III separates from stage II and after the carbon dioxide is consumed, the mercury vapor is condensed solely by radiation from the condenser.

If the strontium 89 has decayed for 110 days before the rocket takes off as assumed in Appendix X, the calculated strontium cell will produce 360 watts of electrical power. If the maximum consumption of electric power by the equipment is 300 watts, the extra 60 watts may be dissipated in an electrical resistance which would be located as far as possible from the mercury condenser. At the end of fourteen days the auxiliary power plant will produce only the required 300 watts. The strontium would still have 100 watts available at the end of 102 days of orbital flight. But the turbine and condenser will stop operating within a few days after the fourteenth day, because the turbine and condenser cannot be expected to operate much below their design conditions.

F. RADIOACTIVE CELL - THERMOPILE SYSTEM

In order to utilize the reduced watt output of the strontium's later days of activity, as well as to avoid the other difficulties of the condenser and miniature turbine, another means of power generation was investigated. This method would consist of a thermopile with the hot junctions imbedded in a strontium 89 cell and the cold junctions attached to a plate which will radiate heat out into space. This thermopile circuit would feed power directly to the electronic equipment or indirectly through a battery. In the past, the use of this thermo-electric effect to measure temperature required only minute quantities of power. However, by using two conductors (or semi-conductors) which have a high thermo-electric power as the two wires of the thermocouple, and by having a large number of thermocouples in series, it is possible to produce a reasonable quantity of power.

Again, as in the case of the mercury vapor system, the important criteria of a thermopile power plant in a satellite rocket are minimum weight and the amount of heat which is lost by conduction through the wires and which must be radiated from the vehicle. The general expressions for these criteria are developed in Appendix XII and are given below.

The weight in pounds of the thermopile wires is

$$W_{gt} = \frac{16\omega\rho\, P\, d^2}{(Q\Delta T)^2}\ , \tag{75}$$

where

ω = average specific weight of the wires, gms/cm^3

ρ = average electrical resistivity of the wires, ohm cm

P = power in load in watts

Q = thermo electric power per thermocouple, volts per $^\circ$C

ΔT = temperature difference between hot and cold junctions in $^\circ$C

d = distance between junctions, cms.

The heat lost by conduction through the wires in watts:

$$H = \frac{16\rho h\, P}{Q^2\, \Delta T} \tag{78}$$

where

h = average thermal conductivity of the wires in calories/cm/sec/$^\circ$C.

From Eqs. (75) and (78) above, it is readily seen that best materials for the thermopile should have these properties: low specific weight, low electrical resistivity, low thermal conductivity, and high thermo-electric power. A number of thermopile combinations were investigated, including: iron-constantin, carbon-silicon, molybdenum-silicon, germanium-silicon, germanium-platinum, and germanium type N - germanium type P. None of these systems gave small enough weights and required radiating areas to be satisfactory for use in the satellite vehicle. However, in view of the simplicity of such a power plant, a research program to discover a combination of materials which would make a satisfactory thermopile power plant is advisable.

IV. CONCLUSIONS

It is possible to build an auxiliary power plant which will produce 300 watts for two weeks with an overall weight of less than 400 pounds.

The power plant which is recommended for use is the strontium 89, mercury vapor system. This system would weigh approximately 157 pounds. The next best method would be a hydrogen-peroxide gas turbine system with an intermittent duty cycle. This power plant with a gross weight of 400 pounds would supply power long enough to maintain earth-to-satellite contact for 137 hours.

APPENDIX I

I. THROATLESS vs CONVENTIONAL MOTORS

A preliminary approximate comparison has been made between the conventional liquid rocket combustion chamber and the 'throatless' chamber (a chamber of the same diameter as the throat of the nozzle being used), based on the overall efficiency of the two types.

The possible advantage of the latter, if it were as efficient as the conventional design, would be the saving in combustion chamber size and weight. On the other hand, there is the possible disadvantage of a more difficult cooling problem due to the higher average velocity of gases in the chamber and the resulting higher coefficient of heat transfer.

It is found that a comparison of the two motors using a given amount of fuel, the same injection pressure, and the same exhaust pressure favors the conventional design by a small percentage of thrust produced. The percentage decreases as the exhaust pressure decreases.

1. LIST OF SYMBOLS

ρ = Density

v = Velocity

A = Area

T = Temperature

p = Pressure

θ = Heat of vaporization of propellants

C_p = Specific heat at constant pressure

R = Gas constant for particular gas under consideration

γ = C_p/C_v = Ratio of specific heat at constant pressure and constant volume

a = Velocity of sound

M = v/a Mach number

$p_{(\ )_t}$ = Stagnation pressure

I = Specific impulse

F = Thrust

m = Mass rate of flow of propellant

p_e = Pressure of gases inside the nozzle at the exit section

p_v = Atmospheric pressure at altitude for which motor was designed to have perfect expansion

p_c = Combustion chamber pressure

h = Heat liberated by the reaction (heat added)

2. THROATLESS ROCKET MOTOR

It is assumed that the problem of combustion in a throatless rocket motor can be approximated as combustion in a duct of constant cross section closed at one end.

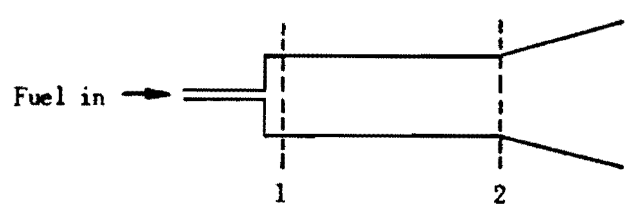


DIAGRAM OF THROATLESS MOTOR
FIG. 10

Further assumptions:

1. At section (1) the fuel is evenly distributed over area A_1.

2. At section (2) combustion is complete.

3. $\gamma = \dfrac{C_p}{C_v}$ is constant.

Continuity equation

$$\rho_1 v_1 A_1 = \rho_2 v_2 A_2 \tag{10}$$

Momentum equation

$$(\rho_2 v_2 A_2) v_2 - (\rho_1 v_1 A_1) v_1 = p_1 A_1 - p_2 A_2 \tag{11}$$

Energy equation

$$(C_p T_2 + \tfrac{1}{2} v_2^2 + \theta) - (C_p T_1 + \tfrac{1}{2} v_1^2) = h$$

$$(C_p T_2 + \tfrac{1}{2} v_2^2) - (C_p T_1 + \tfrac{1}{2} v_1^2) = h - \theta = H \tag{12}$$

Equation of state

$$p/\rho = RT \tag{13}$$

Now:

$$C_p T = \frac{C_p}{R} RT = \frac{C_p}{R}\frac{p}{\rho} = \frac{\gamma}{\gamma - 1}\frac{p}{\rho} .$$

Then the energy equation becomes

$$\left(\frac{\gamma}{\gamma - 1}\frac{p_2}{\rho_2} + \tfrac{1}{2} v_2^2\right) - \left(\frac{\gamma}{\gamma - 1}\frac{p_1}{\rho_1} + \tfrac{1}{2} v_1^2\right) = H.$$

Now defining:

$$\frac{H}{C_p T_1 + \tfrac{1}{2} v_1^2} = \frac{\text{heat added}}{\text{total energy in inflow}} = C_h.$$

The energy equation becomes

$$\left(\frac{\gamma}{\gamma - 1}\frac{p_2}{\rho_2} + \tfrac{1}{2} v_2^2\right) - \left(\frac{\gamma}{\gamma - 1}\frac{p_1}{\rho_1} + \tfrac{1}{2} v_1^2\right)\left(1 + C_h\right) = 0.$$

But:

$$a_1^2 = RT_1 = \frac{p_1}{\rho_1}.$$

Dividing energy equation by $\frac{\gamma}{\gamma - 1}\frac{p_1}{\rho_1} = \frac{1}{\gamma - 1} a_1^2$,

$$\left[\frac{p_2}{p_1}\frac{\rho_1}{\rho_2} - ½\left(\frac{v_2}{v_1}\right)^2 M_1^2 (\gamma - 1)\right] - \left[1 + \frac{\gamma - 1}{2} M_1^2\right]\left(1 + C_h\right) = 0 ,$$

$$\left[\frac{p_2}{p_1}\frac{\rho_1}{\rho_2} - \left(1 + C_h\right)\right] + \frac{\gamma - 1}{2} M_1^2 \left[\left(\frac{v_2}{v_1}\right)^2 - \left(1 + C_h\right)\right] = 0.$$

From continuity equation because $A_1 = A_2$,

$$\frac{\rho_1}{\rho_2} = \frac{v_2}{v_1} .$$

And rewriting the momentum equation

$$1 - \frac{p_2}{p_1} = \frac{\rho_1 v_1^2}{p_1}\left(\frac{v_2}{v_1} - 1\right) ;$$

$$\frac{p_2}{p_1} = 1 - M_1^2\left(\frac{v_2}{v_1} - 1\right) .$$

Substitution of these values for $\frac{\rho_1}{\rho_2}$ and $\frac{p_2}{p_1}$ in the energy equation yields:

$$\left(\frac{v_2}{v_1}\right)\left(1 + M_1^2\right) - M_1^2\left(\frac{v_2}{v_1}\right)^2 - \left(1 + C_h\right) + \frac{\gamma - 1}{2} M_1^2 \left(\frac{v_2}{v_1}\right)^2 - \frac{\gamma - 1}{2} M_1^2 \left(1 + C_h\right) = 0,$$

or:

$$\frac{1 + \gamma}{2} M_1^2 \left(\frac{v_2}{v_1}\right)^2 - \left(1 + \gamma M_1^2\right)\left(\frac{v_2}{v_1}\right) + \left(1 + \frac{\gamma - 1}{2} M_1^2\right)\left(1 + C_h\right) = 0 ,$$

solving the quadratic for $\frac{v_2}{v_1}$,

$$\frac{v_2}{v_1} = \frac{1 + \gamma M_1^2}{(1 + \gamma) M_1^2} \pm \sqrt{\left(\frac{1 + \gamma M_1^2}{(1 + \gamma) M_1^2}\right) - \frac{\left(2 + (\gamma - 1) M_1^2\right)\left(1 + C_h\right)}{(1 + \gamma) M_1^2}} . \qquad (14)$$

Consistent with the initial pressure and exit pressure conditions and that a Mach Number $M_2 = 1$ exist at the end of the combustion chamber, there is, for a given value of C_h, only one Mach Number M_1 that can exist and that is the one corresponding to the value of the radical in the above equation being zero.

Therefore, substituting this value of $\frac{v_2}{v_1}$ the momentum equation gives:

$$\frac{p_2}{p_1} = 1 + \gamma M_1^2 - \gamma M_1^2 \; \frac{1 + \gamma M_1^2}{(1 + \gamma)M_1^2}$$

$$= 1 + \gamma M_1^2 - \gamma \left[\frac{1 - \gamma M_1^2}{1 + \gamma}\right] = \left(1 + M_1^2\right)\left[1 - \frac{\gamma}{1 + \gamma}\right]$$

$$= \frac{1 + \gamma M_1^2}{1 + \gamma}. \tag{15}$$

Now as incoming fuel will have a negligible velocity at station (1) the ratio of 'throat' pressure to fuel inlet pressure becomes:

$$\frac{p_2}{p_1} = \frac{1}{1 + \gamma}. \tag{16}$$

3. COMPARISON OF 'THROATLESS' AND CONVENTIONAL ROCKET COMBUSTION CHAMBER

It is of interest to find the loss of specific impulse caused by using a throatless rocket combustion chamber as compared with one of conventional design.

Assume:
1. Fuel flow rate is the same in both cases.
2. Fuel feed pressure is the same in both cases.
3. Area ratio of the nozzles is the same in both cases.

Static pressure ratio which was derived above is:

$$\frac{p_2}{p_1} = \frac{1 + \gamma M_1^2}{1 + \gamma}$$

but:

$$\frac{p_2}{p_1} \times \frac{p_{2_t}}{p_{1_t}} \times \frac{p_{1_t}}{p_{2_t}} = \frac{1 + \gamma M_1^2}{1 + \gamma}$$

therefore:

$$\frac{p_{2_t}}{p_{1_t}} = \frac{1 + \gamma M_1^2}{1 + \gamma} \times \frac{p_1}{p_{1_t}} \times \frac{p_{2_t}}{p_2}$$

now:

$$\frac{p_{x_t}}{p_x} = \left(1 + \frac{\gamma - 1}{2} M_x^2\right)^{\frac{\gamma}{\gamma - 1}}$$

therefore:

$$\frac{p_{2_t}}{p_{1_t}} = \frac{1 + \gamma M_1^2}{1 + \gamma} \frac{\left(1 + \frac{\gamma - 1}{2} M_2^2\right)^{\frac{\gamma}{\gamma - 1}}}{\left(1 + \frac{\gamma - 1}{2} M_1^2\right)^{\frac{\gamma}{\gamma - 1}}} \tag{17}$$

but as $M_2 = 1$ and as before M_1 is negligible:

$$\frac{p_{2_t}}{p_{1_t}} = \frac{\left(\frac{\gamma + 1}{2}\right)^{\frac{\gamma}{\gamma - 1}}}{(1 + \gamma)} . \tag{18}$$

For the fuel used in the satellite, namely, hydrazine and liquid oxygen, γ is approximately 1.24.

Thus:

$$\frac{p_{2_t}}{p_{1_t}} = \frac{1}{2.24} (1.12)^{5.16} = 0.805 .$$

Now, from the well-known theory of the de Laval nozzle, the ratio of average specific impulses for stage I can be given by the following:

$$\frac{I_{throatless}}{I_{conventional}} = \frac{\dfrac{F_t}{m}}{\dfrac{F_c}{m}} = \frac{F_t}{F_c}$$

$$= \frac{\sqrt{\dfrac{2\gamma^2}{\gamma - 1}\left(\dfrac{2}{\gamma + 1}\right)^{\frac{\gamma+1}{\gamma-1}}\left[1 - \left(1/2\,\dfrac{p_0}{p_c} \times \dfrac{p_{1_t}}{p_{2_t}}\right)^{\frac{\gamma-1}{\gamma}}\right]}}{\sqrt{\dfrac{2\gamma^2}{\gamma - 1}\left(\dfrac{2}{\gamma + 1}\right)^{\frac{\gamma+1}{\gamma-1}}\left[1 - \left(1/2\,\dfrac{p_0}{p_c}\right)^{\frac{\gamma-1}{\gamma}}\right]}}$$

$$= \frac{0.748}{0.762} = 0.982 \quad . \tag{19}$$

And the ratio of specific impulses for stages II and III by:

$$\frac{I_t}{I_c} = \frac{\sqrt{\dfrac{2\gamma^2}{\gamma - 1}\left(\dfrac{2}{\gamma + 1}\right)^{\frac{\gamma+1}{\gamma-1}}\left[1 - \left(\dfrac{p_e}{p_c} \times \dfrac{p_{1_t}}{p_{2_t}}\right)^{\frac{\gamma-1}{\gamma}}\right]}\; p_c f_t + p_e f_e}{\sqrt{\dfrac{2\gamma^2}{\gamma - 1}\left(\dfrac{2}{\gamma + 1}\right)^{\frac{\gamma+1}{\gamma-1}}\left[1 - \left(\dfrac{p_e}{p_c}\right)^{\frac{\gamma-1}{\gamma}}\right]}\; p_c f_t + p_e f_e}$$

$$= \frac{33{,}700 + 1937}{34{,}120 + 1937} = 0.988 \tag{20}$$

Thus, the throatless rocket motor would require 1.8 per cent more propellant in the first stage and 1.2 per cent more propellant in the second and third stages. Inasmuch as this increase in propellant weight is greater than the savings in motor weight, a conventional design is used.

APPENDIX II

NOZZLE WEIGHT SAVINGS BY THE USE OF MULTIPLE ROCKET MOTORS(10)

The weight in pounds of the nozzle of one rocket motor can be expressed as

$$W = \gamma \cdot C \cdot \ell \cdot t \tag{21}$$

where

γ = specific weight of metal, lb/in.^{3}

C = average circumference of metal shell, in.

$$= \pi d_a \tag{22}$$

where

d_a = average diameter of nozzle

ℓ = length of nozzle, in.

t = thickness of the shell necessary to withstand high internal pressure (e.g., the outer shell of a regeneratively cooled motor)(±)

$$= \frac{pd_e}{2\sigma} \tag{23}$$

where

p = pressure differential across the shell, psi

σ = allowable tensile stress of the metal, psi

d_e = exit diameter of nozzle .

Therefore Eq. (21) becomes

$$W = \gamma \cdot \pi d_a \cdot \ell \cdot \frac{pd_e}{2\sigma} \ .$$

(±)Note that for the case of high external pressure (e.g., the inner shell of a regeneratively cooled motor) the expression for thickness is not a simple function due to buckling. However, since buckling increases with size, it becomes a good first approximation to assume the thicknesses of both the inner and outer shells are a function of the diameter.

But since for a given nozzle contour such as those drawn in Figure 2 of Part A of this report the average diameter, $d_a = k_1 d_t$, the length, $\ell = k_2 d_t$, and the exit diameter $d_e = k_3 d_t$

$$W = \frac{\gamma \pi p k_1 k_2 k_3}{2\sigma} \left(d_t\right)^3 . \tag{24}$$

The thrust of the motor is given as

$$F = C_F p_c A_t = C_F p_c \frac{\pi d_t^2}{4} . \tag{25}$$

The thrust of each of 'm' smaller motors having the same total thrust, thrust coefficient and chamber pressure is

$$F = m \cdot C_F \, p_c \frac{\pi d_s^2}{4} \tag{26}$$

where

d_s = throat diameter of the smaller motors.

Combining Eqs. (25) and (26)

$$d_s = \frac{d_t}{\sqrt{m}} . \tag{27}$$

The weight in pounds of the nozzles of 'm' smaller motors is from Eqs. (26) and (27), if the smaller nozzles are made from the same material and are geometrically similar.

$$W_s = \frac{m \pi p k_1 k_2 k_3}{2\sigma} \left(\frac{d_t}{\sqrt{m}}\right)^3 . \tag{28}$$

The ratio of the weight of the nozzles of 'm' smaller motors to that of one large motor from Eqs. (24) and (28) becomes

$$\frac{W_m}{W} = \frac{1}{\sqrt{m}} . \tag{29}$$

The approximate saving in nozzle weight by the use of five motors in stage I of the 86,400 pound ($N_2H_4 + O_2$) satellite rocket is thus

$$100 \left(1 - \frac{1}{\sqrt{5}}\right) = 55.4\% .$$

APPENDIX III

UNIT MOTOR SIZE FOR OPTIMUM COMBUSTION CHAMBER SHAPE

For a combustion chamber to enclose a given volume at a given pressure and to have minimum thickness and surface area, and therefore a minimum weight and surface area to cool, it should be fabricated in the shape approximating a sphere. Therefore

$$V_c = \frac{\pi d_c^3}{6} \quad . \tag{30}$$

Also

$$V_c = L^* A_t \quad . \tag{31}$$

If a value of $d_c/d_t = 2.5$ is chosen, the throat area for an optimum combustion chamber shape becomes

$$A_t = .00725 \, (L^*)^2 \quad . \tag{32}$$

Using the estimated L^* for the hydrazine-liquid oxygen combination of 70 in:

$$A_t = 35.5 \text{ sq in}$$

and in stage I with the following conditions

$$p_c = 400 \text{ psia}$$

$$C_{F_{th}} = 1.39$$

$$\eta_V = 99.5\%$$

$$\eta_F = 90.5\%$$

$$F = 124{,}400 \text{ lbs.}$$

Since

$$A_t = \frac{F}{C_{F_{th}} \times \eta_V \times \eta_F \times p_c} \tag{1}$$

$$A_t = \frac{124{,}400}{1.39 \times 0.995 \times 0.905 \times 400} = 249 \text{ sq in.}$$

Therefore the number of motors that will give optimum combustion chamber shape is

$$\frac{249}{35.5} = 7 \text{ motors.}$$

Because of the arrangement and construction difficulties of 7 motors, 5 motors are used in this design as a compromise number.

APPENDIX IV

STEERING CONTROL WHILE ON ASCENDING TRAJECTORY

An investigation has been made to determine whether jet vanes or movable control rocket motors are more efficient on a basis of fuel used per pound of control force produced.

The study reveals that the control rocket method is more efficient than vanes by at least an amount equal to the parasite drag of the jet vanes as long as the deflection of the control rockets does not exceed twice the deflection of the jet vanes when producing the same control force.

This development points the way to the control of missiles with present day engineering knowledge, for it permits control with movable rockets for as long a period as fuel is available. Vanes, on the other hand, present a problem in materials and cooling that has not yet been solved.

The analysis which follows is intended to determine whether or not a minimum required fuel weight results from controlling a missile by movable rockets or by jet vanes. Further, if a saving results by the control rocket system it is desired to find the optimum division of thrust between the main rocket motor and the control rocket motors.

1. SYMBOLS USED IN ANALYSIS

Jet Vane System

θ = Thrust of rocket motor if jet vanes were absent

D_0 = Parasite drag of one jet vane

D_v = Angle of attack drag of one jet vane

α = Angle between trajectory and longitudinal axis of satellite rocket

β_1 = Angle between longitudinal axis of jet vane and longitudinal axis of satellite rocket

L_v = Lifting force of each jet vane

S_v = Area of one jet vane

S_j = Cross section area of rocket motor nozzle at section where jet vanes are located

M = Mach number

Rocket Control System

F = Thrust of main rocket motor

T = Thrust of each of the control rocket motors

L = Force normal to longitudinal axis of satellite rocket due to angle β_s of each control rocket motor (lift force)

α = Angle between trajectory and longitudinal axis of satellite rocket

β_s = Angle between longitudinal axis of control rocket motor and longitudinal axis of satellite rocket

Weight Effects

T = Thrust of rocket motor

dm/dt = Mass rate of flow of propellants

m = Instantaneous mass of rocket vehicle

c = Exhaust velocity of rocket motor

dv/dt = Acceleration of rocket vehicle

m_o = Initial mass of rocket vehicle

m_i = Final mass of rocket vehicle

v_o = Initial velocity of rocket vehicle

v_i = Final velocity of rocket vehicle

F = Fuel mass

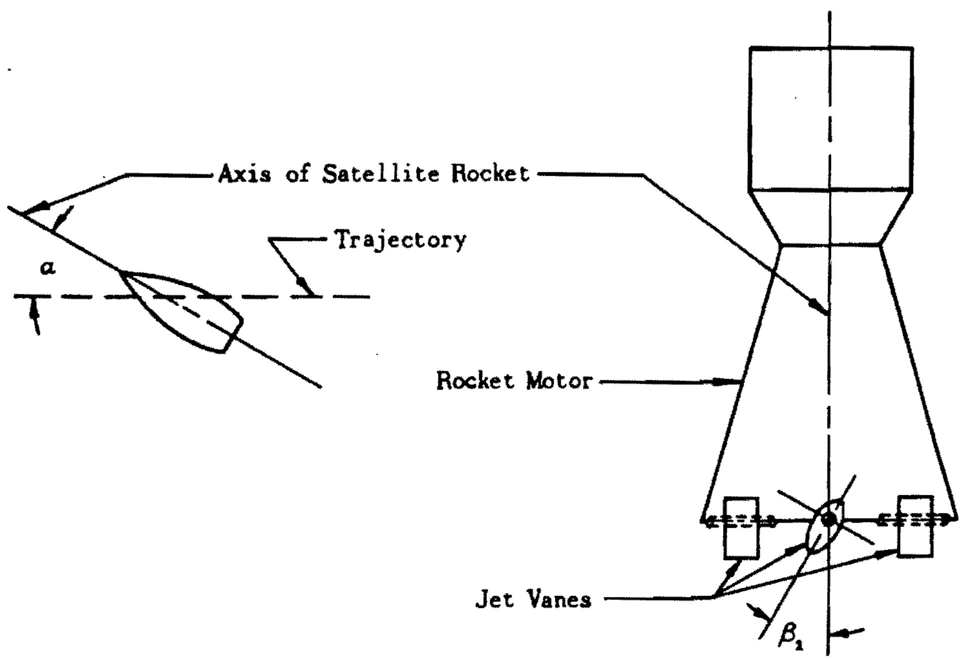
S = Structural mass

P = Payload mass

R = Per cent of fuel (F) saved due to use of control rockets

Q = Per cent of structure and payload ($S+P$) which may be added due to saving R

2. JET VANE SYSTEM (4 CONTROL VANES)



JET VANE CONTROL

FIG. 11

When steering correction is being made:

(1) Thrust of rocket motor along axis of vehicle

$$= \theta - 4\,D_0 - 2\,D_v\,. \tag{33}$$

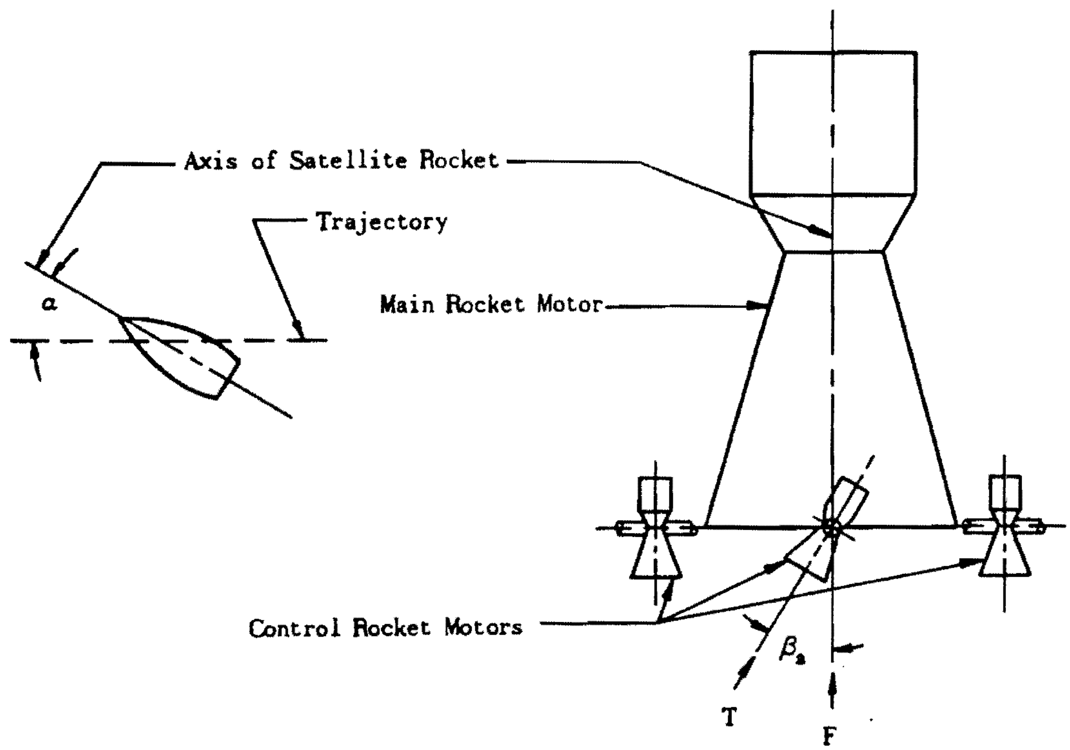
(2) Lift of jet vanes (considering effect of one pair only)

$$2\,L_v = \frac{2.2\,\beta_1 S_v}{\sqrt{M^2 - 1}\,S_j}\,. \tag{34}$$

(3) Thrust of rocket motor along trajectory

$$= (\theta - 4\,D_0 - 2\,D_v)\cos a - 2\,L_v \sin a \tag{35}$$

3. ROCKET CONTROL SYSTEM (4 CONTROL ROCKETS)



ROCKET MOTOR CONTROL
FIG. 12

When steering correction is being made:

(1) Thrust of all rockets along axis of vehicle

$$= F + 2\,T + 2\,T \cos \beta_2 \,. \tag{36}$$

(2) Lift

$$2\,L = 2\,T \sin \beta_2 \,. \tag{37}$$

(3) Thrust of all rockets along trajectory

$$= (F + 2\,T + 2\,T \cos \beta_2) \cos \alpha + 2\,T \sin \beta_2 \sin \alpha \,. \tag{38}$$

4. COMPARISON

As fuel consumption is proportional to thrust along the trajectory and this thrust must be the same for either system, to find relative fuel consumption it is only necessary to equate thrusts along trajectory.

Initial condition (no control acting, zero missile angle of attack)

$$\theta - 4\,D_0 \;=\; F + 4\,T$$

$$\theta \;=\; (F + 4\,T) + 4\,D_0 . \tag{39}$$

Thus, initially the thrust required of the missile using jet vanes is larger than that of the missile using control rockets by the amount of the parasite drag of the jet vanes.

Dynamic condition (control force acting, finite missile angle of attack)

$$(\theta - 4\,D_0 - 2\,D_v)\cos\alpha + 2\,L_v \sin\alpha$$

$$= (F + 2\,T + 2\,T\cos\beta_2)\cos\alpha + 2\,T\sin\beta_2 \sin\alpha . \tag{40}$$

Assume the angle of attack is the same in both cases and the control forces to be equal.

Then

$$(\theta - 4\,D_0 - 2\,D_v) \;=\; (F + 2\,T + 2\,T\cos\beta_2). \tag{41}$$

Before proceeding with the comparison of the two control systems it will be necessary to find the optimum distribution of thrust between F and T as this will affect the loss of thrust due to the angle β_2 at which the control rocket motors operate. The criterion is that distribution which, when giving a given control force, produces the maximum thrust along the trajectory.

5. BEST SIZE OF CONTROL ROCKET MOTORS

The thrust along the trajectory is given by:

$$(F + 2\,T + 2\,T\cos\beta_2)\cos\alpha + 2\,T\sin\beta_2\sin\alpha \;=\; X \tag{42}$$

but from the conditions of the problem:

$$F + 4\,T \;=\; \text{constant} \;=\; K$$

$$2\,T\sin\beta_2 \;=\; \text{constant} \;=\; J.$$

Therefore:

$$(K - 2\,T + 2\,T \cos \beta_2) \cos \alpha + J \sin \alpha = X$$

$$\left[K + 2\,T\,(\cos \beta_2 - 1)\right] \cos \alpha + J \sin \alpha = X$$

and:

$$\frac{dX}{dt} = \cos \beta_2 - 1 = 0$$

$$\cos \beta_2 = 1$$

$$\beta_2 = 0° \,. \tag{43}$$

That is, from second condition above, as $\beta_2 = 0$, T must be infinite. This indicates control rockets should be as large as possible. From the first condition above this is seen to be $T = 1/4\,K$. That is, the four control rockets are each 1/4 of the total thrust. However, there are other reasons for not choosing this condition of size of control motors which are discussed in Ref. 5.

6. DEFLECTION OF CONTROL ROCKETS

Because for practical reasons the control motors cannot be made the optimum size, the comparison will be made with an arbitrary distribution of thrust between F and T,

$$\theta - 4\,D_0 - 2\,D_v = F + 2\,T + 2\,T \cos \beta_2 .$$

Now as:

$$\theta - 4\,D_0 = F + 4\,T$$

$$F + 4\,T - 2\,D_v = F + 2\,T + 2\,T \cos \beta_2$$

$$2\,T - 2\,D_v = 2\,T \cos \beta_2$$

$$D_v = T\,(1 - \cos \beta_2)$$

but:

$$D_v = \frac{2\,\beta_1^{\,2}\,S_v\,\theta}{\sqrt{M^2 - 1}\,S_j}$$

and

$$1 - \cos \beta_2 = \frac{\beta_2^{\,2}}{2} \quad (\text{for small } \beta_2)$$

therefore:

$$\frac{2\,\beta_1^{\,2}\, S_v\, \theta}{\sqrt{M^2 - 1}\; S_j} = \frac{T\,\beta_2^{\,2}}{2}$$

now as the lift of one jet vane is

$$\frac{2\,\beta_1\, S_v\, \theta}{\sqrt{M^2 - 1}\; S_j}$$

and the lifts of both systems are equal

$$\frac{2\,\beta_1\, S_v\, \theta}{\sqrt{M^2 - 1}\; S_j} = T \sin \beta_2 = T\,\beta_2 \quad (\text{for small } \beta_2)$$

then:

$$\beta_1\, T\, \beta_2 = T\frac{\beta_2^{\,2}}{2}$$

$$\beta_2 = 2\,\beta_1\,. \tag{44}$$

Therefore, the control rocket is more efficient than the jet vane system by an amount at least D_0 per vane as long as $\beta_2 \leqq 2\,\beta_1$. Thus, it is seen that the control rocket is more efficient in all cases under consideration as T can be adjusted to make above condition true.

The above determines the maximum β_2. The minimum is defined as that at which the servomechanism is not sensitive enough to precisely control the control motors, hence, T cannot be larger than the value determined by this minimum β_2.

7. WEIGHT EFFECTS

It is of interest to determine how much the structural weight or payload of the missile can be increased at the expense of saving on fuel load due to lack of drag term (D_0 and part of D_v) when control rockets are used for steering.

Simplified method: (assume vacuum trajectory and no gravity)
Using jet vanes:

$$T = -\frac{dm}{dt}c = m\frac{dv}{dt} \tag{45}$$

$$\int_{v_0}^{v_1} dv = -c\int_{m_0}^{m_1} \frac{dm}{m}$$

$$v_1 - v_0 = \Delta v = -c \log_e \frac{m_1}{m_0}$$

but:

$$m_0 = F + S + P$$

$$m_1 = S + P$$

and if:

$$F = Xm_0$$

then:

$$S + P = (1 - X)\, m_0$$

and

$$\Delta v = -c \log_e (1 - X)$$

now if rockets are used instead of vanes, and rockets save R per cent of F, then Q per cent of $(S + P)$ may be added.

$$\Delta v = -c \log_e \frac{(1 - X)(1 + Q)}{X(1 - R) + (1 - X)(1 + Q)} \;. \tag{46}$$

For the rocket vehicles to have the same performance, equate Δv's before and after saving was made:

$$-c \log_e(1 - X) = -\frac{c \log_e(1 - X)(1 + Q)}{\left[(1 - R) - (1 + Q)\right]X + (1 + Q)}$$

$$1 - X = \frac{(1 - X)(1 + Q)}{\left[(1 - R) - (1 + Q)\right]X + (1 + Q)}$$

$$(Q - R)X + (1 + Q) = (1 + Q)$$

$$Q = R \;. \tag{47}$$

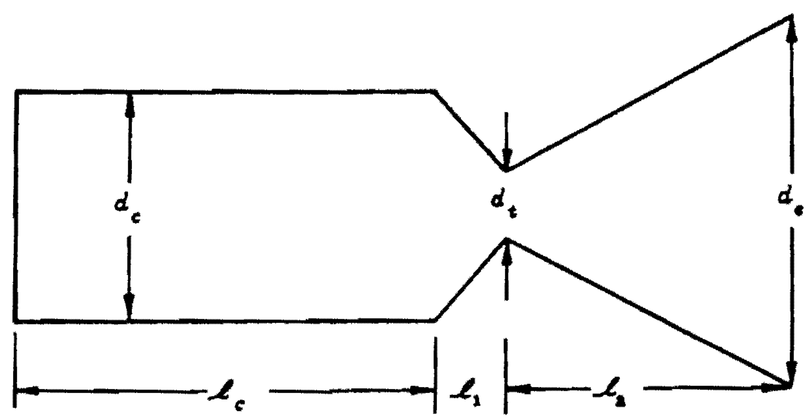
For a given saving of $R\%$ of fuel (F) the structural weight ($S + P$) can increase $Q\%$ of ($S + P$). When R and Q are numerically equal the same Δv (performance) will result as before the saving in fuel was made.

This per cent Q of ($S + P$) will be smaller for a given saving of $R\%$ of (F) were the performance measured by the true trajectory equations, but in no case will it go to zero.

APPENDIX V

SAMPLE CALCULATIONS FOR REGENERATIVE MOTOR COOLING

The motor design was assumed as shown in the drawing below:



SIMPLIFIED SKETCH OF ROCKET MOTORS
FIG. 13

Thus, the expression for the surface area to be cooled is:

$$A = \pi d_c \ell_c + \frac{\pi}{4} d_c^2 + \pi \sqrt{\left(\frac{d_c - d_t}{2}\right)^2 + \ell_1^2}\left(\frac{d_c + d_t}{2}\right) + \pi \sqrt{\left(\frac{d_e - d_t}{2}\right)^2 + \ell_2^2}\left(\frac{d_e + d_t}{2}\right) \tag{48}$$

The expression for the overall rise in coolant temperature is:

$$\Delta T_f = \frac{qA}{W_f C_p} \tag{49}$$

where:

q = density of heat flow (BTU/in^{2}sec)

A = surface area (in^2)

W_f = wt. rate of coolant flow (lbs/sec)

C_p = specific heat at constant pressure (BTU/lbs°F).

Using these relations and the assumed value of density of heat flow as 1¼ BTU/in^{2}sec, the overall mean rise in coolant temperature can be found.

a) For control rocket motors of stage III:

Chamber pressure = 150 psi

Thrust = 30 lbs

Specific fuel consumption = 0.00384 (sec^{-1})

ΔT_f = 684°F.

The boiling point of hydrazine at the fuel pressure of 225 psi is 421°F. Thus, straight regenerative cooling is not possible in this case.

b) For main rocket motor of stage III:

Chamber pressure = 150 psi

Thrust = 7820 lbs

ΔT_f = 200°F.

With the increase in rocket motor size the possibility of regenerative cooling increases.

c) For control rocket motors of stage II:

Chamber pressure = 300 psi

Thrust = 1200 lbs

ΔT_f = 165°F.

The boiling point of hydrazine at the fuel feed pressure of 338 psi is 462°F. With the increase of rocket motor pressure the ratio of surface area to rate of fuel flow decreases and again the possibility of regenerative cooling improves.

APPENDIX VI

GAS FEED VS PUMP FEED

A method has been set up for rapidly determining the best combination of gas pressurization and pump feeding for a liquid rocket power plant from a weight consideration. A sample calculation is given for stage I of the hydrazine-liquid oxygen, three-stage satellite rocket. This method indicated by the sample calculation and illustrated in Fig. 14 through 20 can be used for any propellant combination, size of vehicle, combustion chamber pressure, and maximum load factor. The dynamic pressure head due to this load factor has been neglected for both the gas feed and pump feed systems because when the two systems are compared at the same load factor the additional tank weight will cancel out. Furthermore, this additional tank weight is small because with an 'n' of 5 the tank pressure would be in order of only 10 psi above the static value.

All the tanks were considered spherical in order to simplify the comparison, therefore, if the gas feed system results in the lower weight, then a more careful calculation should be made of the tank weight using the actual propellant tank shapes. The derivation of the relations which are plotted in Fig. 14 through 20 is given in Ref. 2.

If the effect of combustion chamber pressure on specific impulse is included in the calculation the optimum combination of propellant tank pressure and combustion chamber pressure for any stage of the satellite may be determined.

SAMPLE CALCULATION

Given Data:

Stage I of a three-stage satellite rocket using a liquid oxygen-hydrazine pump-fed-rocket power plant.

Propellant tank pressures (p_p) are maintained at 15 psia.

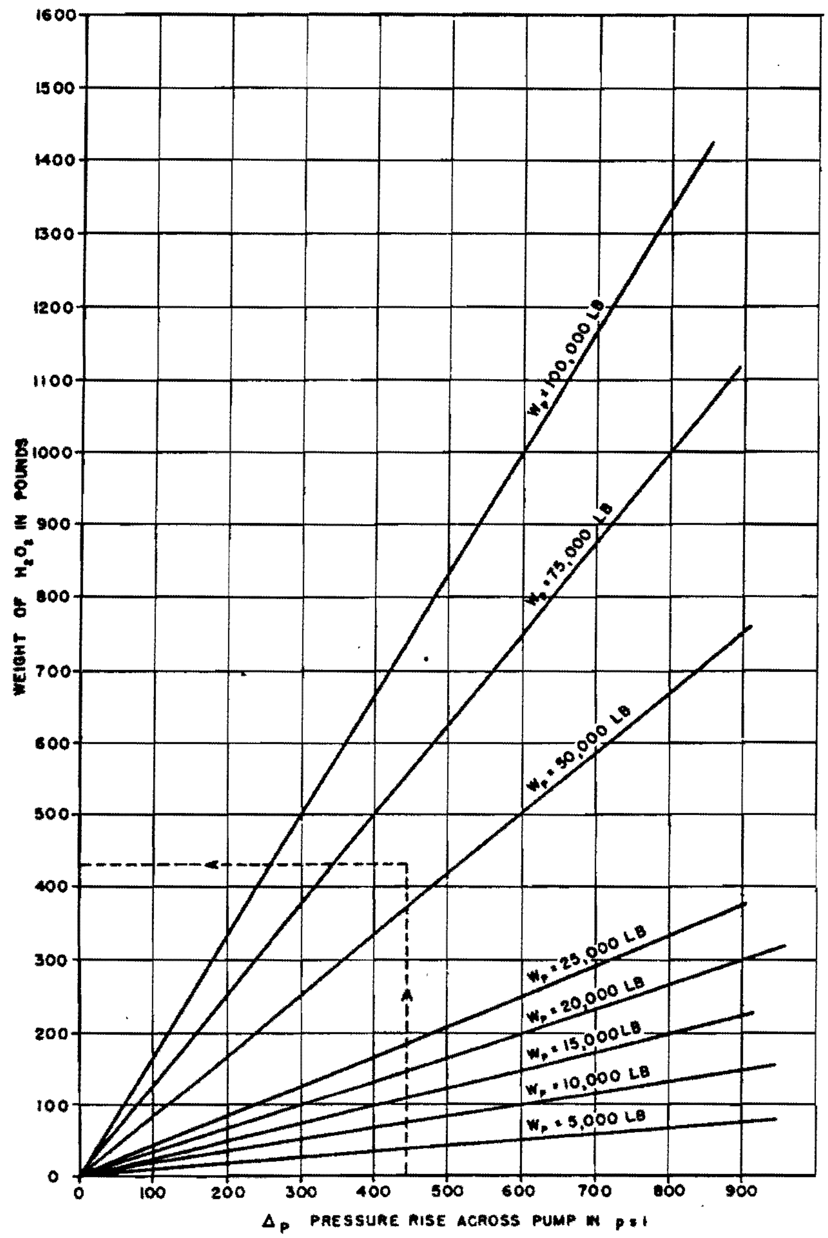
Combustion chamber pressure (p_c) is 400 psia.

Average static head times average load factor = nh = 316 inches of average propellant.

Pressure drop through the cooling coils and injector (Δp_L) is 72 psi. (This pressure drop was used for both the fuel and oxidizer. Actually only the fuel passes through the cooling coils, but inasmuch as the 72 psi is only an estimate and it was used for both the pump feed and gas feed calculations, this assumption is justified.)

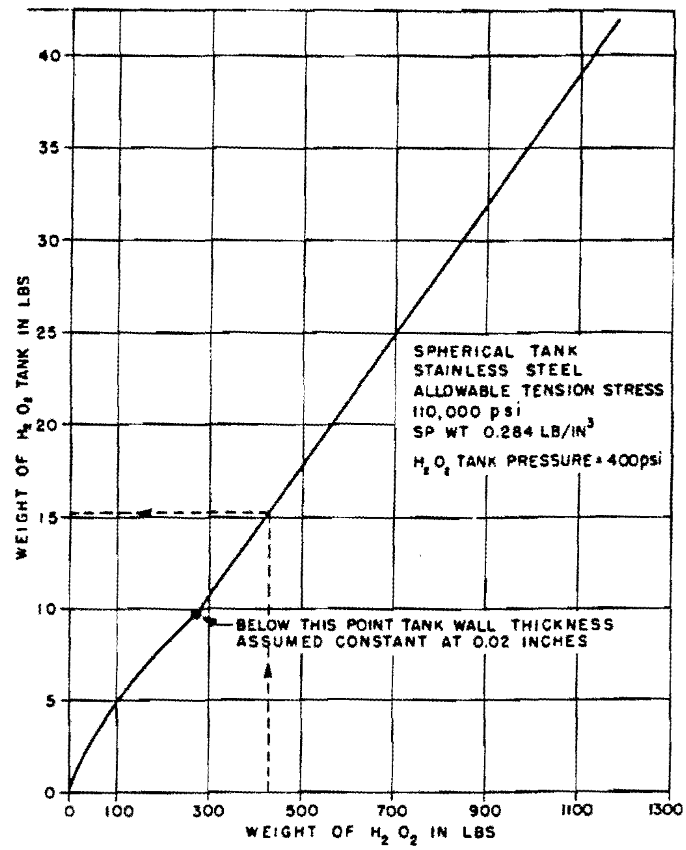
Hydrogen peroxide tank pressure is 400 psia.

Helium tank pressure is 3000 psia.



WEIGHT OF H_2O_2 REQUIRED vs PRESSURE RISE ACROSS PUMP WITH TOTAL WEIGHT OF PROPELLANTS AS A PARAMETER

FIG. 14



WEIGHT OF H_2O_2 TANK VS WEIGHT OF H_2O_2

FIG. 15

Total propellant weight is 57,574 lbs.

Propellant flow rate is 512 lbs/sec.

Fuel tank volume is 945,000 cu in.

Oxidizer tank volume is 560,000 cu in.

Find:

Weight of the propellant delivery system plus the weight of the propellant tanks.

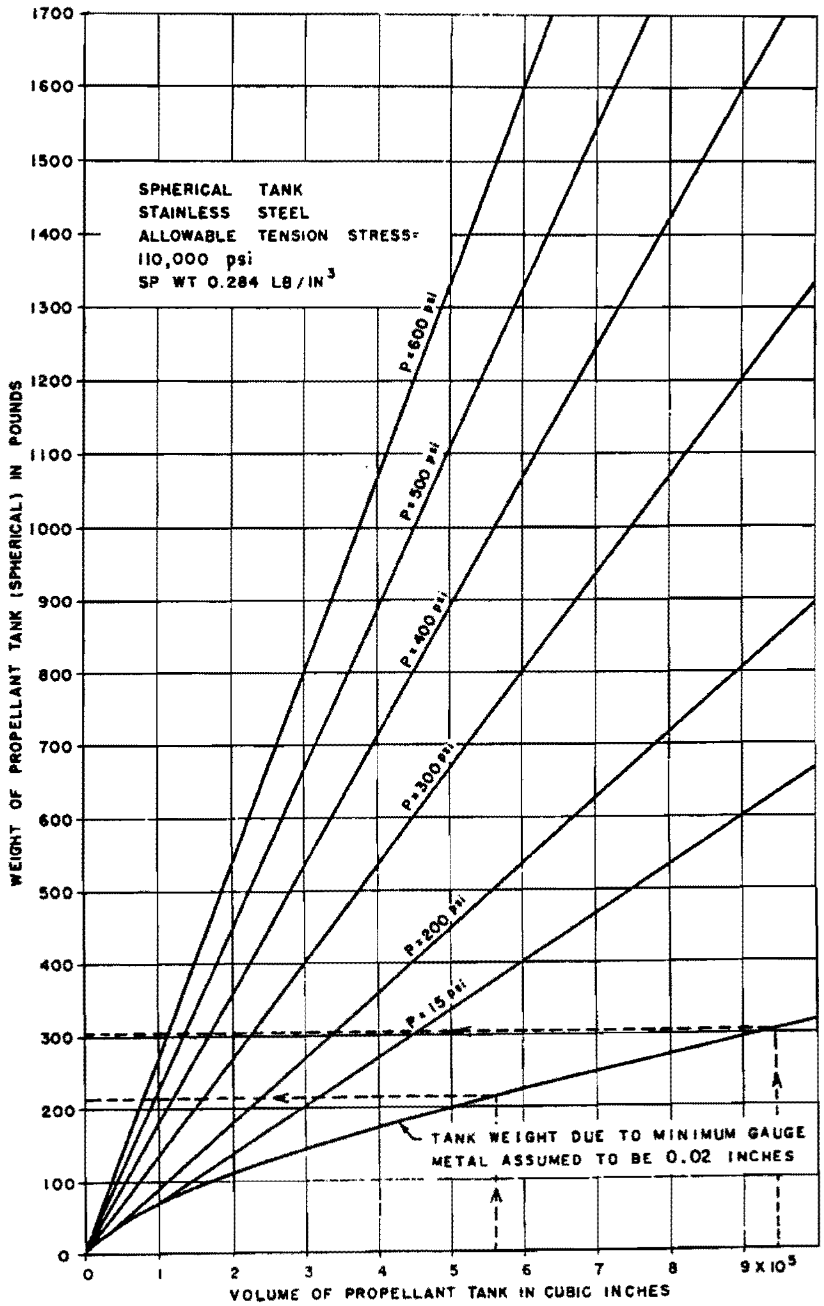
Calculations:

The pressure increase desired from the pump

$$= p_c + \Delta p_L - \frac{\gamma_o + \gamma_F}{2}(nh) - p_p \tag{50}$$

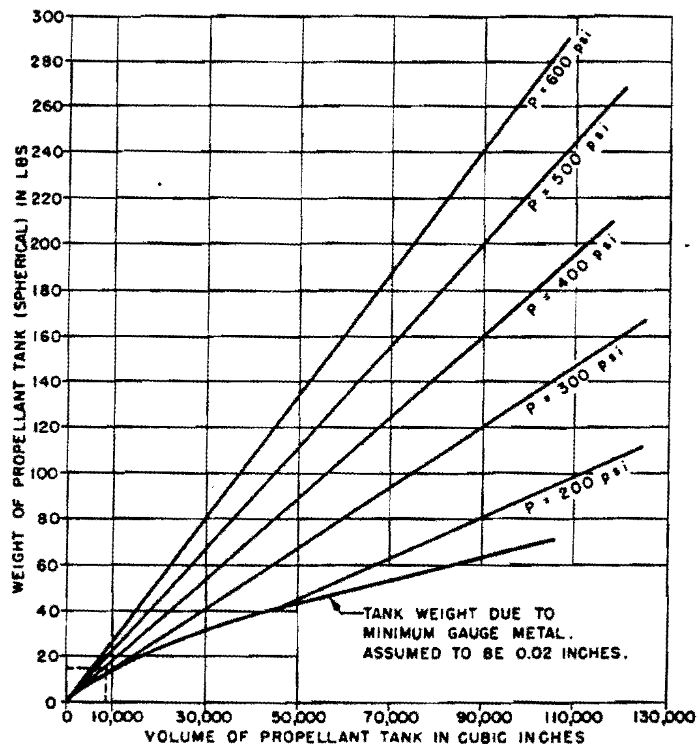
$$= 400 + 72 - \frac{.0412 + .0365}{2}(316) - 15$$

$$= 400 + 72 - 12.2 - 15 = 445 \text{ psi.}$$



WEIGHT vs VOLUME OF PROPELLANT TANK WITH PROPELLANT TANK PRESSURE AS A PARAMETER

FIG. 16(a)



WEIGHT VS VOLUME OF PROPELLANT TANK WITH PROPELLANT TANK PRESSURE AS A PARAMETER

FIG. 16 (b)

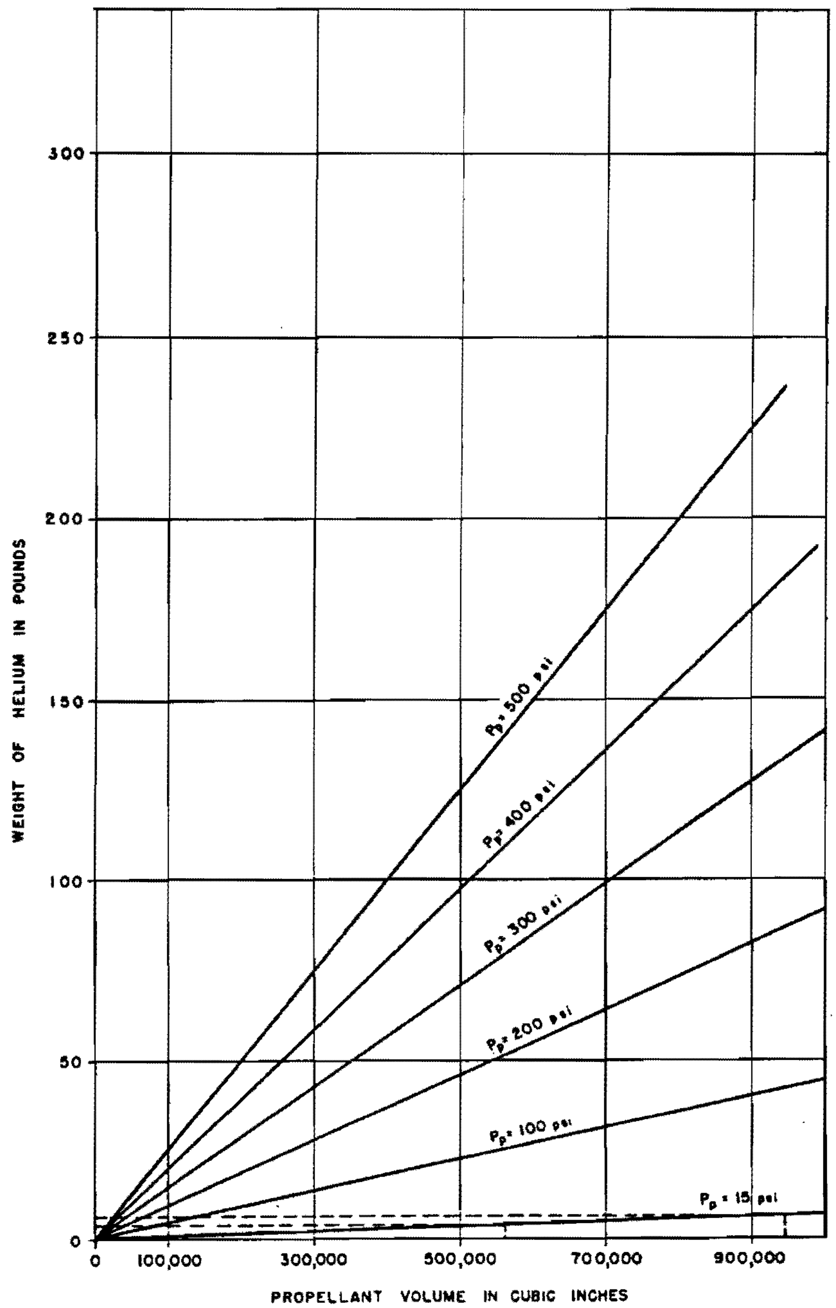
Entering Fig. 14 with this pressure rise and the total weight of propellants to be pumped, the weight of H_2O_2 necessary to drive the pumps is found to be 430 pounds (8500 cu in.). From Fig. 15 the weight of the H_2O_2 tank required to contain this amount of H_2O_2 at 400 psia is found to be 15 pounds.

Knowing the volumes of the fuel and oxidizer and the propellant tank pressure (p_p), Fig. 16a gives the weights of the fuel and the oxidizer tanks as 305 pounds and 215 pounds, respectively.

Fig. 16b, an expanded scale of Fig. 3, in this case permits a check of the weight of the H_2O_2 tank. However, for stage III of the satellite rocket, the main propellant tanks fall on this graph.

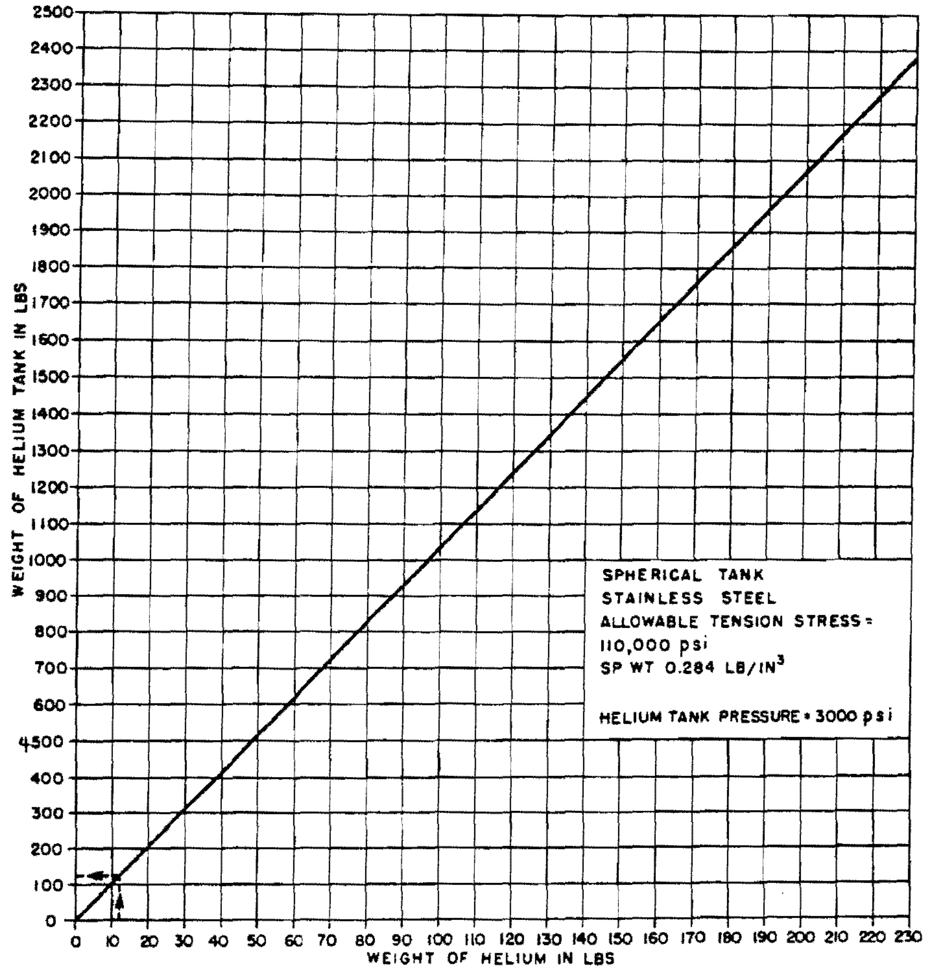
Having the volumes of the fuel, oxidizer, and H_2O_2 tanks and their respective tank pressures, Fig. 17 gives the weight of the helium required to pressurize these tanks. These figures are:

For the fuel tank . . . . . 6 pounds of helium
For the oxidizer tank . . . 4 pounds of helium
For the peroxide tank . . . 2 pounds of helium.



WEIGHT OF HELIUM VS PROPELLANT VOLUME WITH PROPELLANT TANK PRESSURE AS A PARAMETER

FIG. 17



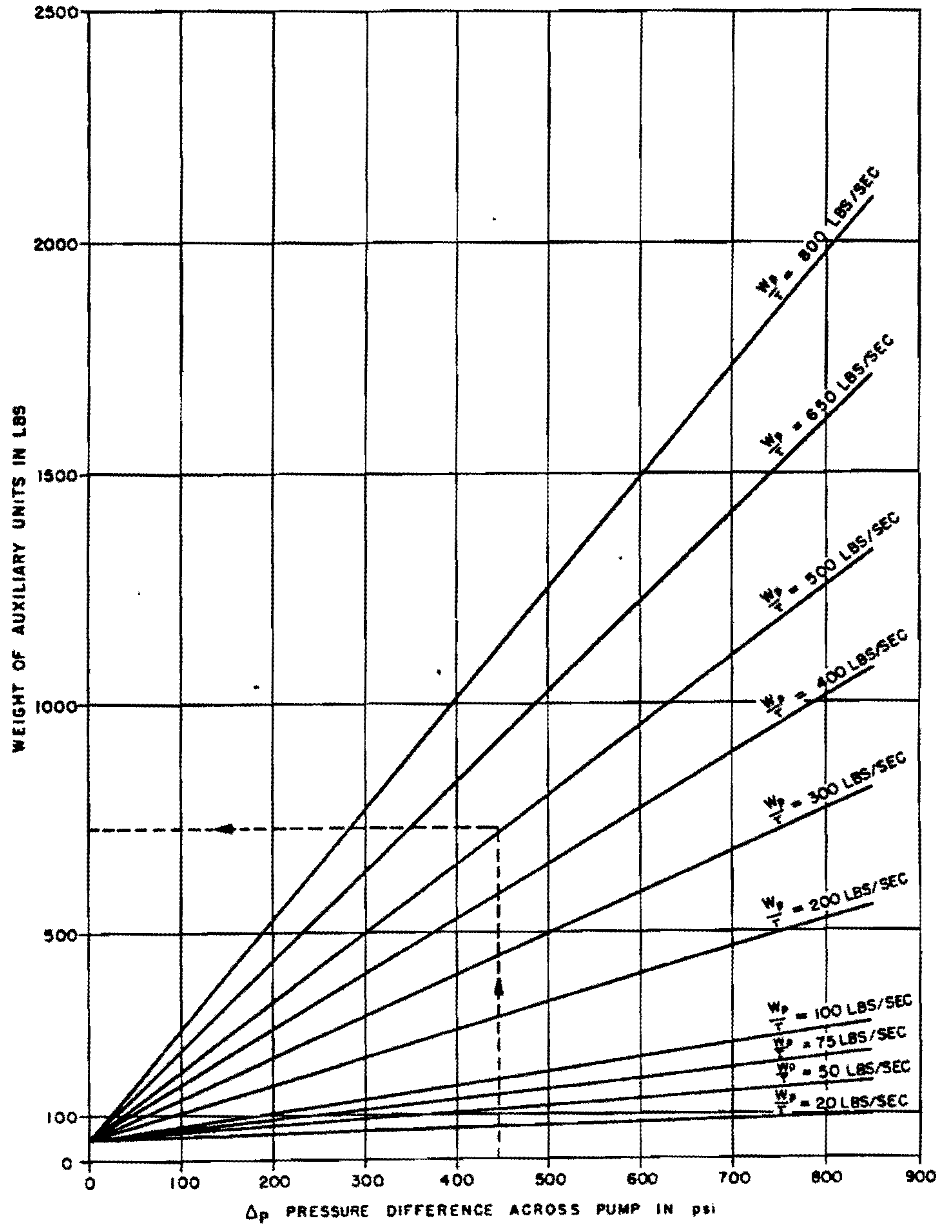
WEIGHT OF HELIUM TANK VS WEIGHT OF HELIUM

FIG. 18

Entering Fig. 18 with the total weight of helium required, the weight of the helium tanks to store this gas at 3000 psia is found to be 124 pounds.

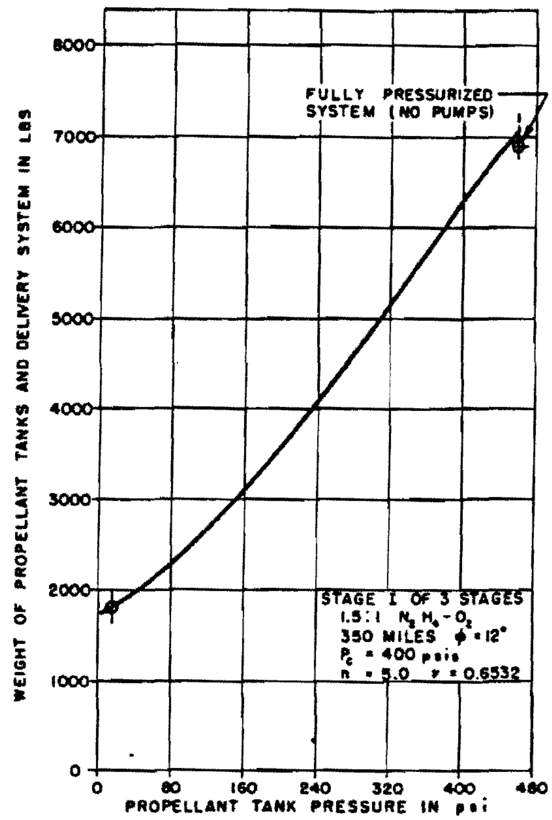
The weight of the auxiliary units used in the pump-feed rocket such as pumps, turbine, valves and plumbing is obtained from Fig. 19, knowing the pressure rise across the pumps and the propellant flow rate. In this example the weight is 730 pounds.

The total weight of the propellant delivery system plus propellant tank weights is thus the sum of the above weights or 1831 pounds.



WEIGHT OF AUXILIARY UNITS VS PRESSURE DIFFERENCE ACROSS PUMP WITH WEIGHT FLOW RATE OF PROPELLANTS AS A PARAMETER

FIG. 19

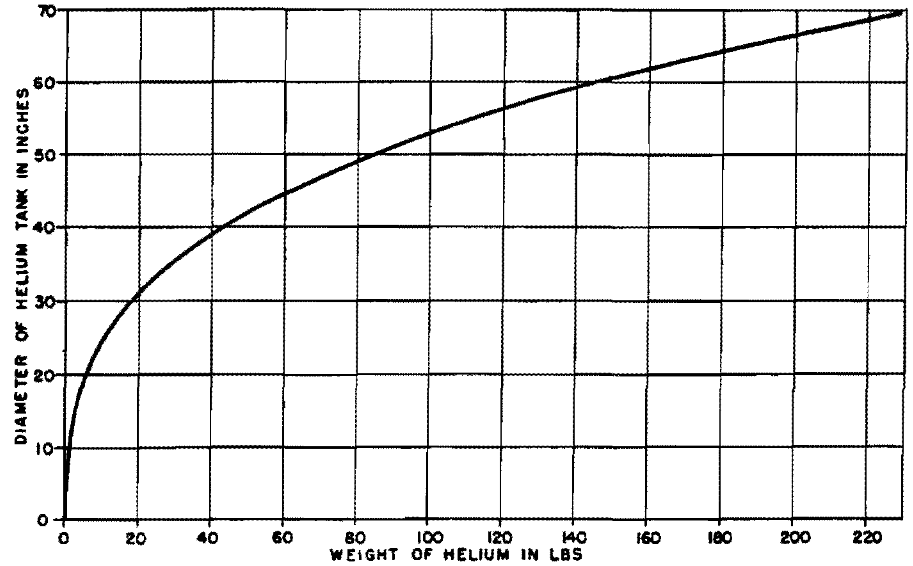


WEIGHT OF PROPELLANT TANKS AND DELIVERY SYSTEM VS PROPELLANT TANK PRESSURE

FIG. 20

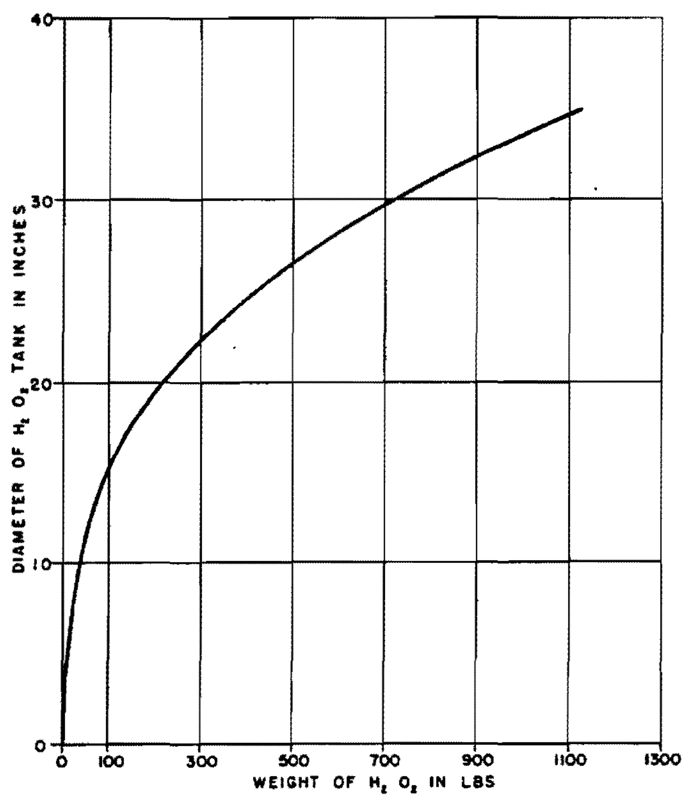
Now repeating the above calculation for several different initial tank pressures, to and including a fully pressurized system in which no turbines or pumps are needed, results in a curve such as in Fig. 20. From this curve of the satellite rocket power plant weights vs. propellant tank pressures, the tank pressure for the minimum rocket gross weight may be chosen.

The only tank sizes which vary with the propellant tank pressure are the helium and hydrogen peroxide tanks. The diameters of these tanks are given as a function of their contents in Figs. 21 and 22. In this case the diameter of the H_2O_2 tank is 25.3 inches and the helium tank diameter is 26 inches. Consideration of these diameters plus the relative size of the auxiliary units determines the comparative bulk of the pump-feed and gas-feed systems.



DIAMETER OF HELIUM TANK vs WEIGHT OF HELIUM

FIG. 21



DIAMETER OF H_2O_2 TANK vs WEIGHT H_2O_2

FIG. 22

APPENDIX VII

SAMPLE CALCULATIONS FOR TURBINES AND PUMPS

STAGE I - PUMPING SYSTEM

Hydrazine Pump

The mass rate of flow of hydrazine to the constant mass flow motors in stage I is equal to the total weight of hydrazine divided by the burning time.

$$\dot{m} = \frac{34,600}{112} = 308 \text{ lb/sec}$$

The necessary capacity of the pump, Q, is therefore

$$Q = \frac{308 \times 60}{.0365 \times 231} = 2190 \text{ gal/min.}$$

The pressure increment desired from the pump = combustion chamber pressure + pressure drop in the injector and cooling coils − pressure at the pump inlet due to the acceleration force − pressure in the propellant tank. The pump discharge velocity is assumed to be equal to the pump inlet velocity.

$$\Delta p = p_c + \Delta p_L - \gamma\,(nh) - p_p$$

$$= 400 + 72 - 12 - 15$$

$$= 445 \text{ psi} = 1029 \text{ feet of water.}$$

The power output of the pump in horsepower is

$$P = \frac{\dot{m}\,\Delta p}{6600\,\gamma} \qquad (51)$$

where

$\dot{m}$ = weight rate of propellant flow, lb/sec

Δp = pressure rise across pump, psi

γ = specific weight of propellant, lb/in^3 .

The power output of the stage I hydrazine pump must therefore be

$$P = \frac{(308)(445)}{(6600)(.0365)}$$

$$= 568 \text{ horsepower}.$$

For large centrifugal pumps a head of 500 feet of water is about the largest which is practical at present; therefore, it is necessary to use a double-stage, single suction centrifugal pump. For a capacity of 2190 gal/min the maximum efficiency is at a specific speed of 2000 (see page 1897 of Ref. 8).

$$\text{Specific speed} = N_s = \frac{NQ^{1/2}}{(\Delta p)^{3/4}} = 2000 \tag{52}$$

$$\Delta p \text{ (per stage)} = \frac{1029}{2} = 514.5 \text{ ft}$$

$$\therefore N = \frac{2000\ (514.5)^{3/4}}{(2190)^{1/2}} = 4590 \text{ rpm}.$$

To estimate the outside diameter, D, of the impeller casing, the following equation of similarity was used:

$$\frac{Q}{D^3 N} = \text{constant} \tag{53}$$

where

N = actual speed of pump.

To determine this constant, the following specifications of a Worthington, single-stage centrifugal pump type 2½ - R - 2 were used.

Δp = 425 feet

Q = 350 gal/min

N = 3500 rpm

P = 60 horsepower

Outside diameter of impeller casing = 14.75.

The estimated diameter of the stage I hydrazine pump is thus found:

$$\frac{350}{(14.75)^3\ 3500} \equiv \frac{2190}{(D)^3\ 4590}$$

$$D = 24.8 \text{ in.}$$

Liquid Oxygen Pump

The mass rate of liquid oxygen flow

$$\dot{m} = \frac{23100}{112} = 204 \text{ lb/sec} .$$

The capacity is

$$Q = \frac{204 \times 60}{.0412 \times 231} = 1287 \text{ gal/min.}$$

The pressure increment desired from the pump is the same as for the hydrazine pump except the liquid oxygen does not have the assumed 30 psi pressure drop through the cooling coils. Therefore

$$\Delta p = 415 \text{ psi} = 959 \text{ feet of water.}$$

Therefore, this pump also will have to have two stages. The power output of the stage I liquid oxygen pump is

$$P = \frac{(204)(415)}{(6600)(.0412)} = 311 \text{ horsepower.}$$

Inasmuch as it is desirable to operate this pump at the same actual speed of the hydrazine pump to eliminate use of reduction gears, the resulting specific speed of this pump is

$$N_s = \frac{(4590)(1287)^{1/2}}{(480)^{3/4}} = 1615.$$

This specific speed and the 1287 gal/min capacity will give a satisfactory efficiency.

Turbine And Turbine Fuel

In order to determine the power output of the turbine it is necessary to find the overall efficiencies of the two pumps. This is done as suggested in Ref. 8 by multiplying the efficiencies of the individual stages and deducting 8 per cent. Estimated overall efficiency of hydrazine pump

$$= (0.85 \times 0.85) - (0.08) \cong 0.65 = 65\%.$$

Estimated overall efficiency of oxygen pump

$$= (0.83 \times 0.83) - (0.08) \cong 0.61 = 61\%.$$

The power output required from the turbine in stage I is therefore:

$$P = \frac{568}{0.65} + \frac{311}{0.61} = 1384 \text{ horsepower.}$$

The total heat input in BTU to the turbine in stage I assuming a turbine efficiency of 25 per cent is

$$= \frac{P\ (\text{turbine}) \times t_B \times 550}{0.25 \times 778} \tag{54}$$

$$= \frac{1384 \times 112 \times 550}{0.25 \times 778}$$

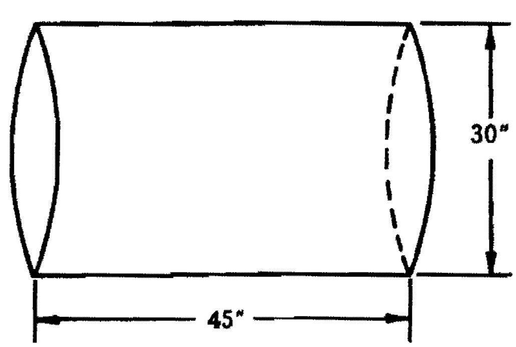
$$= 439{,}000 \text{ BTU.}$$

The weight of 90% hydrogen peroxide required if the combustion efficiency is assumed to be 90 per cent and if the heat content of the fuel is 1137 BTU/lb (see Ref. 7) is therefore:

$$= \frac{439{,}000}{0.90 \times 1137} = 429 \text{ pounds.}$$

Estimated Size Of Turbine And Pumps

It is assumed that the maximum height of this unit is equal to the diameter of the impeller casing of the largest pump plus the required space for the inter-stage piping. Thus, the turbine casing is equal to or smaller than this height. The overall length of the pumping unit including space for two two-stage pumps, one turbine, and four bearings is estimated for each stage. In stage I the space requirement is



STAGE II - PUMPING SYSTEM

The same calculations and assumptions are made in stage II as in stage I, except a specific speed of 1600 is used for the hydrazine pump.

STAGE III - PUMPING SYSTEM

Inasmuch as the control motors in stage III have an independent gas-feed system, only the propellant flow to the main motor was used in this calculation. A specific speed of 800 was used here for the hydrazine pump and a single-stage pump can supply the 459 and 389 foot heads of hydrazine and oxygen, respectively. A specific speed as low as 800 was used here because otherwise the actual speed becomes excessive. This decreases the efficiency of the hydrazine pump to about 57 per cent but this is better than using a specific speed of 1600 and a double-stage pump which would have an efficiency of only about 40 per cent.

The pumps in stage III will be considerably smaller than the Worthington pump described in stage I above; therefore, a different pump is used as a reference in estimating the size. This reference pump is:

Worthington single-stage centrifugal pump Type $2 - R - 1$

Δp = 240 feet

Q = 100 gal/min

N = 3500 rpm

P = 15 horsepower

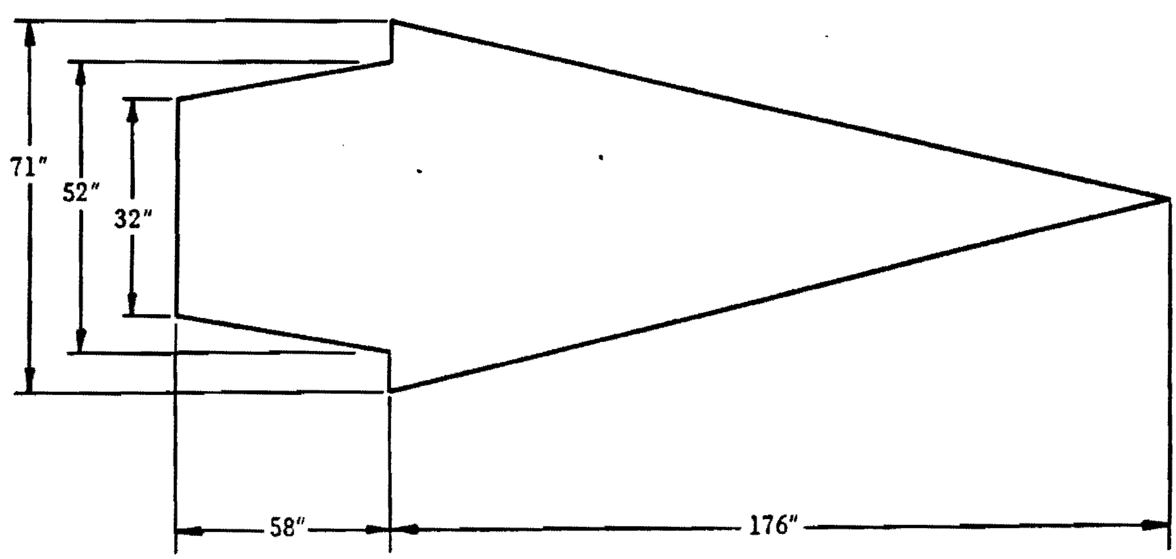
Outside diameter of impeller casing = 12 inches.

APPENDIX VIII

SOLAR ENERGY ABSORBER TO SUPPLY AUXILIARY POWER

The maximum effective absorber area of the stage III rocket of the three-stage 86,400-pound satellite would be the projected area of one side (see Fig. 23).

$$\text{Area} = 1/2 \times 52 \times 176 + 1/2\ (32 + 52)\ 58 = 8690 \text{ sq in.} = 60.3 \text{ sq ft}$$



PROJECTED AREA OF STAGE III OF THE SATELLITE ROCKET
FIG. 23

Total radiant energy coming from the sun striking a surface perpendicular to the sun's rays outside of the atmosphere = 7.15 BTU/sq ft/min[9]. A thin flat-tank absorber has been built in the United States giving an efficiency of 50.1%[9]. Therefore, the maximum heat that would be absorbed by a satellite boiler is:

$$7.15 \times 60.3 \times 0.50 = 215 \text{ BTU/min.}$$

As the satellite orbit is in the plane of the earth's equator, the average projected area of the satellite as seen by the sun when the satellite is on the sunward side of the earth is approximately 1/2 of the total projected area of the satellite. If it is assumed that a solar absorber covering the total projected area of the satellite is used, the absorber will be effectively perpendicular to the sun's rays for 1/4 of the orbital revolution. Therefore, if the satellite completes the orbital circle in 103.2 minutes, the BTU added to the working fluid per revolution is:

$$1/4 \times 215 \times 103.2 = 5560 \text{ BTU.}$$

If the turbine efficiency is 25%, the work output of the turbine is:

$$5560 \times 0.30 \times 778 = 12.98 \times 10^{5} \text{ ft lb.}$$

If the electrical generator has an efficiency of 85%, the work output of the generator is:

$$12.98 \times 10^{5} \times 0.85 = 11.02 \times 10^{5} \text{ ft lb.}$$

If a battery is used for storing this energy, the power available for continuous telemetering and attitude control during each orbit is:

$$\frac{11.02 \times 10^{5}}{103.2} = 10.68 \times 10^{3} \text{ ft lb/min}$$

$$= 242 \text{ watts.}$$

APPENDIX IX

HYDROGEN PEROXIDE, STEAM TURBINE SYSTEM FOR AUXILIARY POWER

Two calculations will be made: (1) to determine the overall power plant weight for a hydrogen peroxide system which will supply 300 watts continuously for two weeks; (2) if an energy saving telemetering duty cycle is devised in order to keep the power plant weight at 400 pounds, to determine the total contact time between the earth and the satellite.

CALCULATION 1.

Weight of Fuel

The heat required from the fuel if the generator efficiency is 85% and the turbine efficiency is 30%.

$$\text{Heat required} = \frac{300}{.85 \times .30} = 1175 \text{ watts} = 66.8 \text{ BTU/min} .$$

The suggested fuel is a 90% solution by weight of H_2O_2 in H_2O. If the fuel temperature is 32°F and the chamber pressure is 300 psi, the latent chemical energy is 1137 BTU/lb and the resulting vapor temperature is 1326°F[7]. If a combustion efficiency of 95% is assumed the weight rate of fuel will be

$$\frac{66.8}{.95 \times 1137} = .0619 \text{ lbs/min.}$$

Total fuel consumption for a two-week duration is

$$.0619 \times 60 \times 24 \times 14 = 1248 \text{ pounds.}$$

Weight of Fuel Tank

The specific gravity of a 90% solution of H_2O_2 is 1.393, therefore:

$$\text{Volume of } H_2O_2 \text{ solution} = \frac{1}{1.393 \times 62.4}$$

$$= 14.33 \text{ cu ft}$$

$$= 24,770 \text{ cu in.}$$

If the fuel tank is built in the shape of a sphere $V = \frac{\pi d^3}{6}$, the diameter would be:

$$d = \left(\frac{6 \times 24,770}{\pi}\right)^{1/3} \tag{55}$$

$$d = 36.2 \text{ inches.}$$

Effective maximum pressure inside the fuel tank equals the combustion chamber pressure plus drop in the fuel lines and injector plus the pressure force due to the acceleration of the rocket during its ascending trajectory. This maximum pressure, p, is about 335 psi for a combustion chamber pressure of 300 psi. The material used for the tank is '18-8' stainless steel, 1/2 hard with an allowable tensile stress, σ, equal to 89,500 psi. Therefore, the thickness of the tank, t, is

$$t = \frac{pd}{4\sigma} = \frac{335 \times 36.2}{4 \times 89,500} \tag{56}$$

$$t = .0339 \text{ inches.}$$

If the density of the steel, ρ, is .284 pounds per cu in., and if 15% is added to the weight of the shell for fittings, welds, etc., the weight of the tank becomes

$$\pi d^2 \times t \times \rho \times 1.15 = \pi(36.2)^2(.0339)(.284)(1.15) = 45.6 \text{ pounds.}$$

Weight of Catalyst and Catalyst Tank

Inasmuch as the catalyst is not chemically consumed during the reaction and as the range of per cent catalyst for a maximum heat release is quite wide, this weight is not calculated. Based on a previous similar system the weight of potassium permanganate and container is estimated at 45 pounds. This weight can perhaps be reduced if a solid catalyst such as manganese dioxide impregnated in the combustion chamber walls is used.

Weight of Helium

The weight of the helium stored at 3000 psi required for the fuel tank at its normal operating pressure of 325 psi is

$$W_{He} = 6.45 \times 10^{-6}\,\frac{V_p p_p}{15}\left(1 + \frac{p_p}{3000}\right) \tag{9}$$

$$= 6.45 \times 10^{-6}\,\frac{(24770)(325)}{15}\left(1 + \frac{325}{3000}\right)$$

$$W_{He} = 3.75 \text{ pounds.}$$

Weight of the Helium Tank

By using the same method that was used for finding the weight of the fuel tank, the weight of the helium tank is found.

$$\text{Volume of helium tank} = \frac{3.75}{6.45 \times 10^{-6}} \times \frac{15}{3000} = 2910 \text{ cu in.}$$

The diameter of the spherical helium tank is

$$d = \left(\frac{6 \times 2910}{\pi}\right)^{1/3} = 17.7 \text{ inches.}$$

The weight of the stainless steel helium tank is

$$W_T = \pi d^2 \times \frac{pd}{4\sigma} \times \rho \times 1.15$$

$$= \frac{\pi (17.7)^3 \times 3000 \times 0.284 \times 1.15}{4 \times 89,500}$$

$$= 48 \text{ pounds.}$$

Weight of the Turbine

The horsepower rating of the turbine would be

$$HP = \frac{300}{.85} \times \frac{1}{746} = 0.473 \text{ horsepower.}$$

Inasmuch as it is desired to expand the combustion products from 300 psia to 2 psia or less, and to do it as efficiently as possible, a multi-stage turbine would be required. In the larger horsepower ranges, turbines can be built to give one horsepower per one pound of weight. It is probable, however, that a multi-stage turbine with a two-week duration as small as this one cannot be built for less than 15 pounds. It would have to be a partial admission (probably one nozzle inlet) type of turbine because of the low mass flow. This inherently makes a multi-stage turbine have low efficiency because of 'windmill' type losses. It is probable that a reciprocating engine could be used to advantage here, however, because of the low power required and long duration of operation it is unlikely that any weight saving could be accomplished. Further, because of the large number of moving parts of a reciprocating engine as compared with a turbine, the reliability is correspondingly decreased.

Weight of Plumbing, Filters, Valves, and Combustion Pot

This equipment, as in the case of the turbine, would have to be built with fine watch-like precision. This feed and combustion system would not weigh more than 5 pounds. Inasmuch as the turbine and combustion system weight is such a small fraction of the weight, it would be desirable to have two turbines and combustion systems connected to the same peroxide and helium tank. Then if one power unit fails, the other unit could be set to automatically go into operation.

Weight of DC Electrical Generator

The generator weight estimate was based on the following aircraft generator specifications:

Generator model M-2 AC # 32285

24 volt - 1.2 kilowatts

maximum rpm = 4500

weight = 18.5 pounds

It is probable that a lighter generator could be designed to produce 300 watts at the higher speeds attainable with the turbine for a weight of about 12 pounds.

Weight Summation of Continuous Peroxide Power System

	Weight in pounds
Hydrogen peroxide solution	1248
Hydrogen peroxide tank	46
Weight of catalyst and container	45
Helium	4
Helium tank	48
Turbine assembly	15
Plumbing, filters, valves, and combustion pot	5
Electrical generator	12

Total weight of two-week, 300-watt power plant = 1423 pounds

CALCULATION 2.

If an energy saving telemetering duty cycle is devised in order to keep the power plant weight at 400 pounds, what will be the duration of contact between the earth and the satellite?

The following assumptions are made for the duty cycle:

1. Radio receiver and control mechanism will continuously draw 100 watts.

2. Data taking instruments, radio beacon, and telemetering devices will draw 200 watts for 2 minutes while passing over each of 4 ground stations during each orbital revolution (orbital period = 103.2 minutes).

3. Hydrogen peroxide turbine will operate for 15 minutes of every 6 hours and charge a light weight aircraft battery.

4. This battery will discharge for 5-3/4 hours before being recharged.

The duration of contact between the earth and the satellite is computed as follows:

The total energy required for each six-hour period is

$$100 \times 6 + \frac{2 \times 4}{103.2} \times 200 \times 6 \;=\; 693 \text{ watt hours.}$$

If the battery efficiency is 80% and the turbine generator is to run 1/4 of an hour every six hours, the generator output must be

$$\frac{693}{0.80} \times 4 \;=\; 3.46 \text{ kilowatts.}$$

If the turbine efficiency is 30%, combustion efficiency is 95%, and the heat content of the hydrogen peroxide is 1137 BTU/lb, the weight rate of fuel is

$$\frac{5.46 \times 42.5}{0.30 \times 0.95 \times 1137} \;=\; 0.718 \text{ lb/min.}$$

Some of the power plant weights can now be determined.

Weight of the Turbine

A 5-1/2 horsepower turbine with 0.718 lb/min mass flow rate could have more than one nozzle inlet with a minimum diameter wheel and therefore would probably weigh about the same as the miniature turbine in calculation 1, Appendix IX above, which is 15 pounds.

Again it is desirable to have two complete turbine generator units in case one breaks down. In other words, the greater reliability that this entails is well worth the shortening of the duration of satellite to earth contact. Both turbine generator units would be operated by the same hydrogen peroxide and helium tanks.

Weight of Plumbing, Filters, Valves, and Combustion Pot

This equipment is still in the minimum weight range and would probably weigh no more than the set in calculation 1 above, which is 5 pounds per set.

Weight of the Generator

The weight of this generator is again based on the aircraft generator specification given in calculation 1. The weight of a 3-1/2 kilowatt generator operating at a speed higher than 4500 rpm is estimated to be 25 pounds.

Weight of Catalyst and Catalyst Tank

The weight of a 10% solution of potassium permanganate and its container, based on the expected duration and combustion chamber size is 20 pounds. Though, again, a weight saving may be realized by using a solid catalyst impregnating the walls of the combustion chamber.

Weight of the Storage Battery

The battery is required to have a capacity of 300/24 × 6 = 75 ampere-hours for a six-hour duration and a voltage of 24 volts. Two present aircraft batteries have the following specifications:

> Exide 6-FHM-13 aircraft battery has a capacity of 88 amp-hr (5 hr.) at 12 volts and a maximum weight of 78 pounds.
>
> AN-W-B-152 aircraft battery has a capacity of 34 amp-hr (5 hr.) at 24 volts and a maximum weight of 76 pounds.

Based on these two batteries, the satellite battery with a small amount of development should not weigh more than 70 pounds.

Weight of the Helium, Helium Tank, and Hydrogen Peroxide Tank

As can be seen in calculation 1, these weights are proportional to the total weight of fuel. If the same tank materials and storage pressures are used as in calculation 1, these two approximate relations may be written

$$\text{weight of the hydrogen peroxide tank} = 0.04\ W_F$$

$$\text{and the weight of the helium tank plus the helium} = 0.04\ W_F$$

where W_F = total weight of hydrogen peroxide.

A summation is now made of the power plant weight excluding the weight of the propellant and propellant weight variables.

	Weight in pounds
Turbine assembly	15
Plumbing, filters, valves, and combustion pot	5
Electrical generator	25
Weight of catalyst and catalyst container	20
Weight of storage battery	70
Weight of power plant excluding propellant weight variables	135 lbs

Therefore, if the total power plant weight is limited to 400 pounds

$$400 = 135 + W_F + 0.04\ W_F + 0.04\ W_F$$

$$W_F = 245 \text{ pounds.}$$

The total duration of operation of this system is therefore

$$\frac{245}{.718} \times \frac{360}{15} \times \frac{1}{60} = 136.7 \text{ hours} = 5.7 \text{ days.}$$

If two sets of combustion chambers, turbines, and generators are carried, the total duration will be reduced to 113.8 hours.

APPENDIX X

RADIOACTIVE CELL CONTAINING STRONTIUM 89

The weight of radioactive strontium 89 (Beta emitter) required to produce 300 watts of useful electrical energy is determined here.

The general radioactive decay equation is

$$N = N_o e^{-\lambda t} \tag{57}$$

where

N = number of atoms disintegrating at time t

N_o = number of atoms disintegrating at beginning of life, t_o

λ = decay constant $= \dfrac{\ln_e 2}{t_H}$

where

t_H = half life of the isotope

= 55 days for strontium 89.

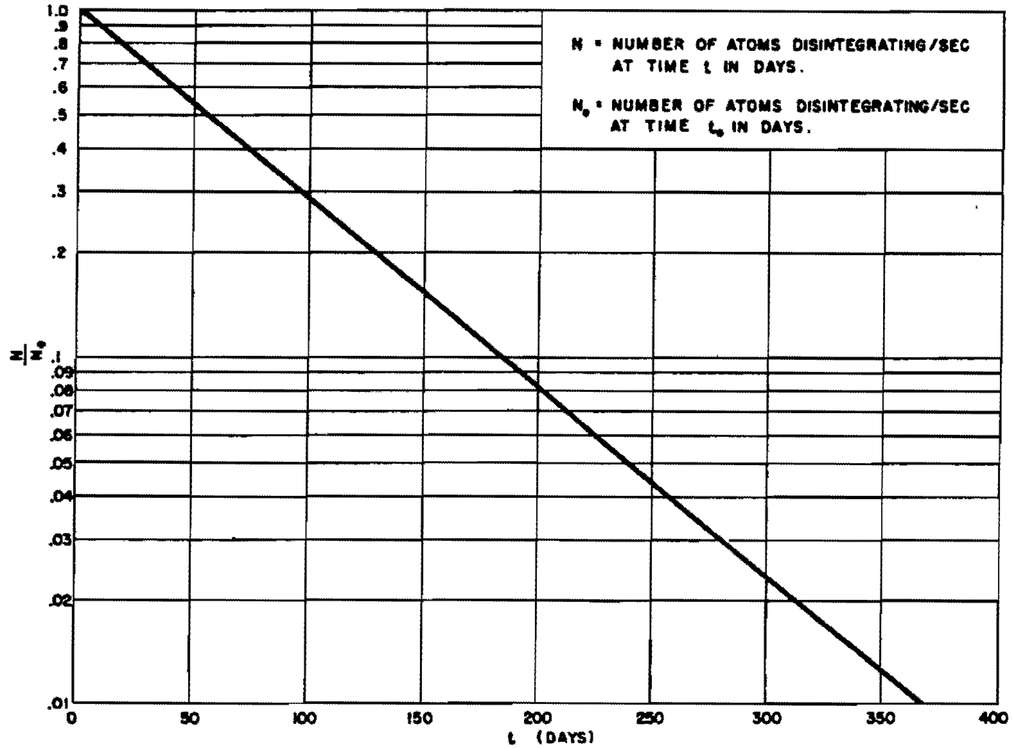
The ratio of N/N_o for strontium 89 is plotted against time in Fig. 24. The following time assumptions are made to determine the age of the strontium 89 for the satellite vehicle:

1. 100 days are required to cool and separate the strontium 89 after the fission of plutonium is completed.

2. 10 days are required to produce strontium cell, transport the cell to the launching location, install the cell in the auxiliary power plant, and launch the satellite.

3. Contact between the earth and the satellite is to be maintained for 14 days, and that 300 watts of useful electrical energy are desired at the end of the fourteenth day.

Therefore, the time for which a given energy release is desired is 100 + 10 + 14 = 124 days. If the overall plant efficiency (from the heat energy in the radioactive



RADIOACTIVE DECAY OF STRONTIUM 89

FIG. 24

cell to the energy produced by the electrical generator) is assumed to be 20%, this desired energy release is

$$\frac{300}{0.20} = 1500 \text{ watts.}$$

Using the conversion factor that one watt equals 6.29×10^{12} mev/sec, the energy release is

$$1500 \times 6.29 \times 10^{12} = 9.43 \times 10^{15} \text{ mev/sec.}$$

The ratio of N/N_0 for 124 days is obtained from Fig. 24 and is equal to 0.208. Therefore, if each atom of strontium 89 which disintegrates produces 1.52 mev, the number of atomic disintegration per second required at the beginning of the isotope's life is

$$N_0 = \left(\frac{9.43 \times 10^{15}}{1.52}\right) \frac{1}{.208} = 2.98 \times 10^{16} \text{ disintegrations per sec}$$

$$= 8.05 \times 10^{5} \text{ curies.}$$

The total number of atoms required is equal to the integral of Eq. (57) between $t = 0$ and $t = \infty$.

$$\Sigma N = \int_0^\infty N_0 \, e^{-\lambda t} \, dt$$

$$\Sigma N = \frac{N_0}{\lambda} \quad . \tag{59}$$

Therefore, the total number of strontium 89 atoms which will give the desired power is

$$\Sigma N = 2.98 \times 10^{15} \times \frac{55 \times 24 \times 3600}{.693} = 2.045 \times 10^{22} \text{ atoms.}$$

The required mass of strontium 89 in grams is

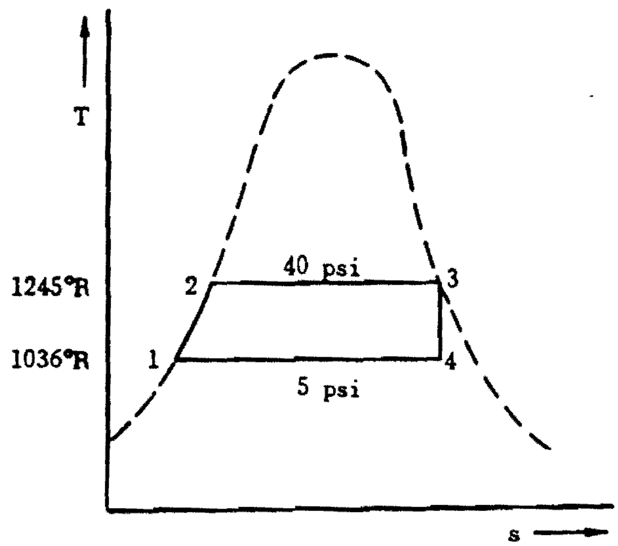
$$\frac{2.045 \times 10^{22} \times 89}{6.03 \times 10^{23}} = 3.02 \text{ grams.}$$

The exact concentration in which the strontium 89 may be obtained is unknown. The first step in the refining process is to separate chemically all the strontium isotopes from the other fission products. The most important separation, and also the most difficult one, is the removal of barium with its physiological harmful gamma rays. After the chemical separation, if it was desired to obtain pure strontium 89, it would be necessary to separate strontium 89 from the five or six other isotopes of strontium by using a mass spectrograph. However, it is believed that this difficult isotope separation will not be necessary because the weight of a sample of mixed strontium isotopes with 3 grams strontion 89 will be reasonably low (possible 5 to 50 grams). Furthermore, the complete chemical separation is not necessary since even a larger amount of dilution of the strontium isotope is desirable in order to make it more readily handled and assembled in connection with the auxiliary power plant.

APPENDIX XI

MERCURY-VAPOR SYSTEM OF AUXILIARY POWER

The proposed mercury-vapor cycle for the auxiliary power plant of the satellite rocket is shown in the temperature-entropy plane in Fig. 25.



MERCURY-VAPOR CYCLE ON TEMPERATURE-ENTROPY PLANE
FIG. 25

The temperature and pressure data is taken from Ref. 8. If the mercury is boiled at 40 psia to a saturated vapor condition and then expanded isentropically in a turbine to a pressure of 5 psia, the quality of the exhaust vapor is

$$x_4 = \frac{s_3 - s_1}{s_{v_4}} = \frac{.1308 - .0346}{.1200} = .801^{(\pm)} . \tag{60}$$

[±] For definition of symbols see pages 100 and 101.

The isentropic drop in heat content through the turbine is

$$\Delta h_t = h_3 - \left(h_1 + x_4 h_{v_4}\right) . \tag{61}$$

Therefore, in this system the isentropic drop in BTU/lb is

$$\Delta h_t = 147.8 - \left[20.3 + .801 \ (124.3)\right] = 27.9 \text{ BTU/lb.}$$

If an electrical generator is to put out 300 watts continuous output and has an efficiency of 80%, the power input to the generator must be

$$\frac{300}{0.80} = 375 \text{ watts} = 21.3 \text{ BTU/min.}$$

The efficiency of a turbine this small, even though carefully developed and constructed, will be low. Therefore, the combined adiabatic and mechanical efficiency of the turbine (not the cycle) is estimated to be 52 per cent. The required flow rate of mercury is therefore

$$w_{Hg} = \frac{21.3}{0.52} \times \frac{1}{27.9} = 1.47 \text{ lb/min.}$$

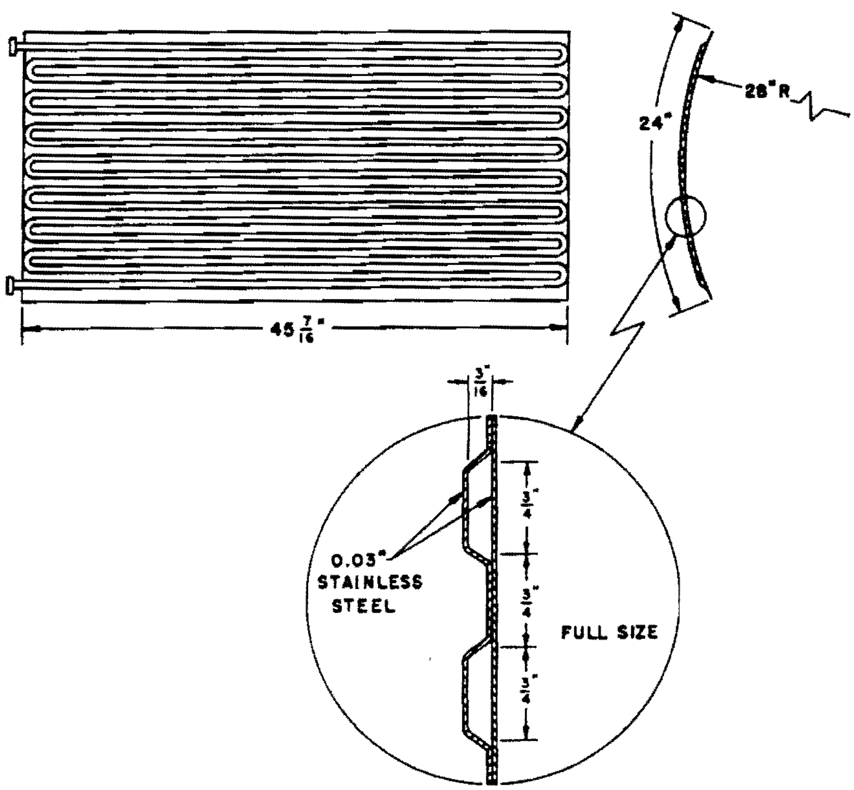
The quantity of heat which must be removed by the condenser per minute is

$$Q_c = w_{Hg} \left(h_1 + x_4 h_{v_4} - h_1\right) \tag{62}$$

$$= 1.47 \ (.801 \times 124.3) = 146.5 \text{ BTU/min.}$$

The length of condenser tubing necessary to condense the above mass flow of mercury vapor is now calculated. In the heat exchanger the heat source will be the *Hg* vapor, and the sink will be the outer surface or skin of the vehicle. It will be assumed that the condenser is on the side of the vehicle which in its orbit faces toward the earth[±], that eight lengths of condenser tubing will pass through one square foot of radiating area, and that the cross-sectional dimensions of the tubing are 3/4" × 3/16", as shown in Fig. 26.

[±] It is, of course, possible to orient the condenser so that it receives no radiation from either the earth or the sun. Calculations for these conditions indicate a reduction in condenser tubing length of about four per cent.



MERCURY VAPOR CONDENSER FOR SATELLITE ROCKET
FIG. 26

The basic mechanisms of heat transfer, viz., radiation, convection, and conduction, have been considered and the following conclusions reached:

1. Heat loss from the Hg molecules by radiation can be neglected because of symmetry properties, i.e., the quantum changes in the energy levels of rotation and interatomic vibration are small.

2. Heat transfer for condensing vapors in the absence of non-condensable gases combines the two mechanisms of convection and conduction and the heat transfer coefficient is large. However, because of the difficulties involved in estimating the gravitational factor[±] this method was not used.

3. In general, the forced convection heat transfer coefficient is smaller than that for a condensing vapor. Since the former is used in the following calculations the resulting size of condenser required is conservative.

In calculating the heat transfer by forced convection, certain simplifications have been made. The following flow and design characteristics are given: Q, thermal energy of the Hg vapor; T_G, the temperature of the Hg vapor; G, the mass flow; tubing dimensions 3/4" × 3/16", and tubing area, A, per square foot of vehicle surface area. With this data it is possible to calculate the length of tubing, L, needed to discharge Q BTU per hour. Assuming no loss of heat to the interior of the vehicle and no loss of heat by conduction in the surface skin of the vehicle, the transfer equation becomes

$$hA\,(T_G - T) = \sigma a T^4 - \sigma f_1 a T_1^{\,4} - \sigma f_1 S\left(\frac{R_S}{L}\right)^2 a\,T_2^{\,4} \tag{63}$$

where the term on the left represents the heat transferred from the Hg vapor to the wall, the first term on the right represents the heat lost to space by radiation from the vehicle surface, the second term on the right represents the heat gained due to radiation from the earth acting as a black radiator, and the third term on the right represents the heat gained due to the diffuse reflection of the sun's radiation from the earth. Calculations are made for a position in the orbit of maximum heat input, hence the calculated length will be the necessary maximum. Eq. (63) may be simplified to

$$\left(\frac{T}{100}\right)^4 + 761\left(\frac{T}{100}\right)h = 7860h + 1180 \tag{64}$$

[±] While attaining its orbit the vehicle will be under a variable-g force and while in the orbit will be gravity-free.

where T is the equilibrium temperature of the tubing wall (or vehicle surface) and h is the heat transfer coefficient. The equation used in calculating the heat transfer coefficient is

$$h = \frac{.023}{Pr} \times \frac{C_p G}{(DG/\mu_f)^{.2}} \tag{65}$$

where

$$Pr = \frac{C_p \mu}{k} \tag{66}$$

is Prandtl's number. Knowing T and h from Eqs. (64) and (65) it is then possible to calculate the condenser tubing length from the equation

$$Q = hPL\ (T_G - T) \tag{67}$$

where P is the tubing perimeter and L is the length.

The results are given in the following table for a condenser with 8 tubes per square foot:

	Units	Value
Q	BTU hr^{-1}	8,800
G	lb hr^{-1} ft^{-2}	90,500
T_{HG}	°R	1,032
T	°R	939
Re		11,980
Pr		.856
h	BTU hr^{-1} ft^{-2} $°F^{-1}$	9.23
L	ft	60.5
A	sq ft	7.57

The condenser consists of a corrugated 0.03-inch steel plate welded to a rectangular plate as shown in Fig. 26. The approximate weight of the sheet metal in this condenser is

$$3.78 \times 12 \times 0.03\ (24 + 24 + 16 \times 2 \times 3/16)\ 0.284 = 20.8 \text{ pounds.}$$

With the addition of 19.2 pounds for fittings, turning elbows, welds, and support bracket, the total weight of the condenser becomes 40 pounds. A liberal estimate of the total weight of mercury required is made by calculating the weight of liquid mercury which would completely fill the condenser (actually liquid mercury will only be found in the last part of the condenser, in the lines to the boiler, in the feed pump, and in first section of the strontium cell boiler). The weight estimate of mercury is therefore

$$60.5 \times 12 \times 3/16 \times 3/4 \times 0.491 = 50 \text{ pounds.}$$

The weight of the turbine and generator was based on the same assumptions given in Appendix IX, therefore, the weights are the same as in that appendix. The boiler feed pump and motor drive was based on minimum size and long duration considerations and was estimated at 5 pounds. Nine pounds of solid carbon dioxide are required to cool the condenser during the ascending trajectory, so the weight of the carbon dioxide plus its spring feeding mechanism is estimated to be 15 pounds. If the solution in the radioactive cell contains 0.1% strontium 89, the weight of the solution for 300 watts would be about 6.8 pounds. If the boiler tubes, cell casing, insulation, and fittings weigh 13.2 pounds, the total weight of the boiler would be 20 pounds. A summary of the unit weights of the radioactive boiler-mercury vapor system which will produce 300 watts for two weeks is given below:

Power Plant Unit	Weight in pounds
Mercury condenser	40
Mercury (working fluid)	50
Mercury turbine	15
Electrical generator	12
Boiler feed pump and motor drive	5
Carbon dioxide cake and feed mechanism	15
Radioactive boiler	20
Total Weight	157 pounds

SYMBOLS USED IN ANALYSIS

s_1 = entropy of saturated liquid mercury at 5 psia

s_3 = entropy of saturated mercury vapor at 40 psia

s_{v_4} = entropy of vaporization of mercury at 5 psia

h_1 = enthalpy of saturated liquid mercury at 5 psia, BTU lb^{-1}

h_3 = enthalpy of saturated mercury vapor at 40 psia, BTU lb^{-1}

h_{v_4} = enthalpy of vaporization of mercury at 5 psia, BTU lb^{-1}

σ = Stefan-Boltzmann constant = $.173 \times 10^{-8}$ BTU ft^{-2} hr^{-1} $°F^{-4}$

f_1 = geometrical factor = .863

S = albedo of earth = .43

R_S = radius of sun

L = earth-sun distance

T_1 = temperature of earth = 420°R

T_2 = temperature of sun

D = hydraulic diameter of condenser tube

C_p = specific heat of mercury

k = thermal conductivity at temperature T

μ = viscosity at temperature $T_{ave} = \dfrac{T + T_C}{2}$

μ_f = viscosity at film temperature T

APPENDIX XII

THERMOPILE SYSTEM FOR AUXILIARY POWER

In a thermopile system used for converting heat energy into electrical energy in the satellite rocket, the two important criteria would be the weight of the pile and the power loss due to heat flow through the wires. The relations used for determining these criteria for various material combinations are developed in this appendix.

NOMENCLATURE

n = number of pairs of thermocouples

r = resistance per thermocouple wire, ohms

E = voltage across load, volts

I = current through load, amperes

P = power in load, watts

e = voltage generated per thermocouple, volts

L = total length of thermopile circuit, cm

d = length of single thermocouple, cm

ρ = average electrical resistivity, ohm-cm

a = cross-sectional area per wire, cm^2

ω = average specific weight of wires, gm cm^{-3}

Q = thermoelectric power of each thermocouple, volts $°C^{-1}$

ΔT = temperature difference between hot and cold junctions, °C

A = total cross-sectional area of thermopile wires, cm^2

h = average thermal conductivity, cal sec^{-1} cm^{-1} $°C^{-1}$

APPENDIX XIII

CONSTANT ACCELERATION ROCKET MOTOR

Late in the study of the satellite rocket power plant, it became apparent that a constant acceleration rocket motor might be made for stages II and III of the vehicle by using a variable combustion chamber pressure.

This follows from a consideration of the thrust equation of a rocket motor:

$$F = \lambda \sqrt{\frac{2\gamma^2}{(\gamma - 1)} \left(\frac{2}{\gamma + 1}\right)^{\frac{\gamma+1}{\gamma-1}} \left[1 - \frac{p_e}{p_c}^{\frac{\gamma-1}{\gamma}}\right]}\; p_c f_t + (p_e - p_0)\, f_e ,$$

where:

F = thrust

λ = $1/2 + 1/2 \cos \alpha$ (where: α = nozzle half angle)

γ = ratio of specific heats = C_p/C_v

p_e = rocket motor exhaust pressure

p_c = rocket motor chamber pressure

f_t = rocket motor throat area

p_0 = atmospheric pressure at altitude of operation

f_e = rocket motor exhaust area .

For operation in a vacuum the thrust equation becomes

$$F = \lambda \sqrt{\left(\frac{2\gamma^2}{\gamma - 1}\right)\left(\frac{2}{\gamma + 1}\right)^{\frac{\gamma+1}{\gamma-1}} \left[1 - \left(\frac{p_e}{p_c}\right)^{\frac{\gamma-1}{\gamma}}\right]}\; p_c f_t + p_e f_e .$$

It can be shown that

$$\frac{f_e}{f_t} = \text{a function of } \frac{p_e}{p_c} .$$

Therefore, for a rocket nozzle of a given area ratio f_e/f_t the ratio p_e/p_c is a constant, regardless of the value of p_c, if separation of the gas stream from the nozzle walls does not occur. This condition will always be satisfied when the nozzle is operated in a vacuum.

Hence, if γ is considered constant the thrust equation for operation in a vacuum can be written as

$$\begin{aligned} F &= K_1 p_c + K_2 p_e \\ &= (K_1 + K_2\, p_e/p_c)\, p_c \\ &= K_3 p_c \end{aligned}$$

where K_1, K_2, and K_3 are constants.

Thus, the thrust is seen to be a linear function of combustion chamber pressure. However, in an actual rocket motor this linear variation of thrust with chamber pressure will have a lower limit defined by the lowest pressure at which smooth combustion can be obtained with the given fuels. For the case of hydrazine and oxygen, this pressure limit has not been determined. The lower pressure limit in conjunction with ν, the ratio of fuel weight to gross weight, determines the initial combustion chamber pressure, that is, the pressure when full thrust is required. As thrust is a linear function of pressure, and fuel flow rate is a linear function of pressure, and fuel flow rate is a linear function of thrust, the fuel flow rate is also determined. The problems expected in the design of a variable thrust motor of this sort are summarized below.

The reduction of pressure in the combustion chamber, with a throat of a given size, is accomplished by restricting the amount of fuel entering the chamber. If this is done by reducing the propellant feed pressure, difficulties from injection atomization, and fuel mixing, are likely. The result is poor combustion. Alternatively, if the propellant flow is restricted by mechanically varying the number of injectors being used at a given time the above difficulties are not likely to manifest themselves.

One major difficulty anticipated with varying thrust in this manner is in the cooling of a motor capable of large thrust when it is operating at low thrust. This is due to the low rate of propellant flow at low thrust.

a. If regenerative cooling is used, the coolant velocity through the coolant coil is low at low thrust with a resulting low rate of heat transfer between the motor wall and the coolant. Hence, the temperature of the combustion side of the motor wall will increase. Regenerative cooling at low thrust is further complicated by the small amount of propellant (coolant) flowing. This means that the amount of heat which can be absorbed by the coolant before it boils is small.

b. If sweat cooling is used, the amount of sweat coolant needed will likely be constant at whatever thrust the motor is operated because the combustion temperature is not greatly affected by pressure.

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General Motors Corporation Allison Division Indianapolis, Indiana Attn: Mr. Ronald Hazen	Bureau of Aeronautics Rep. General Motors Corporation Allison Division Indianapolis, Indiana	BUAER
G. M. Giannini & Co., Inc. 286 W. Colorado St. Pasadena, California		AAF
Hercules Powder Co. Port Ewen, N.Y.	Inspector of Naval Material 90 Church Street New York 7, New York	BUORD
Marquardt Aircraft Company Venice, California Attn: Dr. R. E. Marquardt	Bureau of Aeronautics Rep. 15 South Raymond Street Pasadena, California	AAF BUAER
Menasco Manufacturing Co. 805 E. San Fernando Blvd. Burbank, California Attn: Robert R. Miller Exec. Vice-Pres.		AAF
New York University Applied Mathematics Center New York, New York Attn: Dr. Richard Courant	Inspector of Naval Material 90 Church Street New York 7, New York	BUAER
Office of Chief of Ordnance Ordnance Research & Development Div. Rocket Branch Pentagon, Washington 25, D.C.		ORD DEPT
Polytechnic Institute of Brooklyn Brooklyn, New York Attn: Mr. R.P. Harrington	Inspector of Naval Material 90 Church Street New York 7, New York	BUAER
Purdue University Lafayette, Indiana Attn: Mr. G. S. Meikel	Inspector of Naval Material 141 W. Jackson Blvd. Chicago 4, Illinois	
Reaction Motors, Inc. Lake Denmark Dover, New Jersey	Bureau of Aeronautics Resident Representative Reaction Motors, Inc. Naval Ammunition Depot Lake Denmark, Dover, N.J.	BUAER

D. COMPONENT CONTRACTORS (Cont'd)

(3) PROPULSION

CONTRACTOR	TRANSMITTED VIA	COGNIZANT AGENCY
Rensselaer Polytechnic Institute Troy, New York Attn: Instructor of Naval Science		BUORD
Solar Aircraft Company San Diego 12, California Attn: Dr. M.A. Williamson		ORD DEPT
Standard Oil Company Esso Laboratories Elizabeth, New Jersey	Development Contract Officer Standard Oil Company Esso Laboratories, Box 243 Elizabeth, New Jersey	BUORD
University of Virginia Physics Department Charlottesville, Virginia Attn: Dr. J. W. Beams	Development Contract Officer University of Virginia Charlottesville, Virginia	BUORD
University of Wisconsin Madison, Wisconsin Attn: Dr. J.O. Hirschfelder	Inspector of Naval Material, 141 W. Jackson Blvd. Chicago 4, Illinois	BUORD
Westinghouse Electric Co. Essington, Pennsylvania	Bureau of Aeronautics Resident Representative Westinghouse Electric Corp. Essington, Pennsylvania	BUAER
Wright Aeronautical Corp. Woodridge, New Jersey	Bureau of Aeronautics Rep. Wright Aeronautical Corp. Woodridge, New Jersey	BUAER
Bethlehem Steel Corp. Shipbuilding Division Quincy 69, Mass. Attn: Mr. B. Fox	Supervisor of Shipbuilding, USN Quincy, Mass.	BUAER