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STRUCTURAL AND WEIGHT STUDIES OF A SATELLITE ROCKET

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DOUGLAS AIRCRAFT COMPANY, INC.

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PROJECT RAND

(AAF PROJECT MX-791)

STRUCTURAL AND WEIGHT STUDIES OF A SATELLITE ROCKET

G. H. CLEMENT

RA-15026

February 1, 1947

DOUGLAS AIRCRAFT COMPANY, INC.

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*This initial external distribution list includes the distribution of all related technical reports on the satellite vehicle.

LIST OF FIGURES

LIST OF SYMBOLS

a = axial acceleration

A = surface area

C = weight of payload compartment shell

d = maximum diameter of the rocket body

F = thrust of the rocket motor

g_s = the standard acceleration of gravity at sea level = 32.174 ft/sec^2 = the conversion factor between slugs mass and pounds weight

h = orbital altitude

I = specific impulse

$\bar{I}$ = mean specific impulse

k = a constant, may or may not be dimentionless

ℓ = length of divergent section of the satellite rocket body

ℓ_m = length of rocket motor

ℓ_o = overall length of the satellite rocket body

ℓ_T = propellant tank length

m = instantaneous mass

m = length of satellite rocket payload compartment

M = mass, not variable with time

n = the maximum value of the applied axial load factor, defined by the equation $n = (\sin\theta + a/g_s)$ max

N = total number of stages

p = pressure

p = pressure difference across the propellant pump

p_c = rocket motor combustion pressure

Q = applied structural load

r_b = maximum radius of the rocket body

r_c = body radius at the forward end of the initial stage motor compartment

r_d = base radius of the rocket body

r_T = propellant tank radius

s = length of motor compartment

t = time

t = thickness of tank wall

t_b = duration of burning

U = ratio $\tan\phi$ / $\tan\psi$

V = volume

W = the initial gross weight of a satellite rocket consisting of N stages

W_A = the weight of items in a given stage whose mass is a function of the mass rate of propellant flow

W_B = the basic weight of a given stage defined by the equation $W_B = W - W_{p*} - W_{L*}$

W_C = the weight of items in a given stage whose mass is a function of the degree of control required

W_I = the auto pilot weight

W_j = the initial gross weight of a particular stage, including the succeeding stage as payload

W_k = total pumping work required in a given stage

W_L = 'satellite payload'

$W_{L'}$ = payload of the final stage

W_{L*} = payload of a particular stage

W_M = miscellaneous weight of a given stage

W_P = propellant weight for a given stage

W_{P*} = weight of all items for a given stage whose mass is a function of time

W_p = weight of auxiliary (turbine) fuel for a given stage

W_Q = weight of the orbital power plant and its fuel

W_{au} = weight of the motor group auxiliary units (turbine, pumps, steam generator)

γ = specific weight

δ = length of tank compartment

ϵ = conversion factor between W_P and W_{P*}

η = efficiency

θ = angle between the tangent to the trajectory and the normal to the earth's radius vector

λ = overall length of a given stage

ν = propellant weight to gross weight ratio for a particular stage

ν^* = propellant and auxiliary fuel weight to gross weight ratio for a particular stage

ρ = radius of rocket body at the forward end of a particular stage

ρ = density

σ = unit stress

2ϕ = angle of divergence from the nose to the point of maximum body diameter

2ψ = angle of convergence from point of maximum diameter to the aft end of the rocket body

Subscripts

$(\ )_f$ = a final value

$(\ )_i$ = an initial value

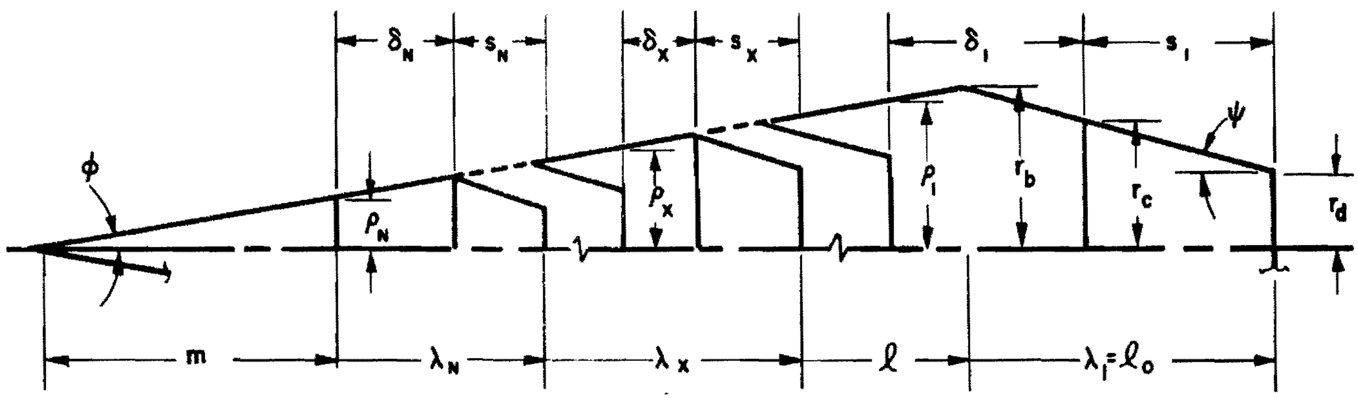
$(\ )_j$ = a particular stage

$(\ )_N$ = the final stage

$(\ )_X$ = an intermediate stage

$(\ )_1$ = the initial stage

$(\ )_1$; $(\ )_2$; $(\ )_3$; . . . . = a specific stage, where the stages are numbered in the order of firing.



GEOMETRY NOTATION

FIG. 1

SUMMARY

In this report consideration is given to problems which enter into the structural design and estimation of weights of a satellite rocket. Methods are shown whereby the structure, power plant and other basic weight items are taken into account in seeking a vehicle of minimum gross weight to achieve the task of placing a given payload on an orbit a few hundred miles above the earth. Many of the procedures as developed should be applicable to all long range rockets because of the apparent similarity of the satellite to rocket missiles.

A design study of a typical satellite rocket is presented as an illustration of the analytical methods and, to some extent, to furnish a check on the calculations.

STRUCTURAL AND WEIGHT STUDIES OF A SATELLITE ROCKET

I. INTRODUCTION

The most important factors affecting the performance of a rocket powered vehicle are the exhaust velocity of the gases generated in the rocket motor, the ratio of propellant weight to gross weight, and the ratio of the basic weight to gross weight. The purpose of this study is to arrive at a realistic engineering appraisal of the effect of factors which influence the basic weight to gross weight ratio, and consequently the gross weight of the satellite rocket, to develop reliable methods by which the gross weight and the basic weight to gross weight ratio may be estimated, and to determine information fundamental to the design of a typical satellite rocket.

The basic weight (W_B) of a satellite rocket is defined as the gross weight of the rocket less the weight of the payload less the weight of propellants and fuels. The importance of the basic weight to gross weight ratio as a parameter of gross weight may readily be seen from an examination of equation (1), which is obtained from the definition of basic weight as given above

$$W = \frac{W_{L'}}{\left(1 - \frac{\Sigma W_{P*}}{W} - \frac{\Sigma W_B}{W}\right)}, \tag{1}$$

where W represents the gross weight of a satellite rocket, $W_{L'}$ the payload, ΣW_{P*} the weight of all propellants and fuels, and ΣW_B the total basic weight.

From Eq. (1) it follows that for fixed values for $W_{L'}$ and $\Sigma W_{P*}/W$, large values of $\Sigma W_B/W$ will give large gross weights, while smaller values of $\Sigma W_B/W$ will give smaller gross weights. It is also instructive to note that Eq. (1) defines the upper limit of the basic weight to gross weight ratio. This may be expressed as

$$\lim\left(\frac{\Sigma W_B}{W}\right) = 1 - \frac{\Sigma W_{P*}}{W} . \tag{2}$$

It is quite obvious that the gross weight would tend to infinity even for very small finite 'payloads' as the ratio $\Sigma W_B/W$ approaches its limiting value. The 'practical' limit of the basic weight to gross weight ratio would, of course, be considerably smaller than that defined by Eq. (2).

Previous studies[1] of the influence of size on the basic weight of rockets indicate that it is a rather dangerous procedure to formulate the estimation of the basic weight of a rocket by the simple process of a geometric enlargement or reduction of a given prototype. It is pointed out that an unlimited geometrical enlargement of a prototype will ultimately bring a penalty in weight, due primarily to the fact that the inertia load imposed upon the structure will increase as the cube of the linear

For references see page 107

scale dimension, while the structure's ability to resist that load will, in certain cases, increase only as the square of the linear scale dimension. Further, when an attempt is made to reduce the size of the prototype it is found that some of the components of the prototype cannot be reduced proportionally and others cannot be reduced at all. Therefore it would appear that the method of estimating the basic weight of a satellite rocket based upon a geometric scaling of a given prototype can only be valid when the variation in size is small. Since the hydrostatic pressures due to inertia (volume) forces increase with the geometric enlargement of a tank, and further since the propellant weight constitutes approximately 70% to 90% of the gross weight of the satellite rocket, the proportions of the propellant tanks may have a considerable influence on the basic weight. In view of these conditions it may be advantageous from a weight standpoint to increase the diameter of the propellant tank and keep the height constant, thereby maintaining the hydrostatic pressure at a constant value. The weight of the propellant tanks and supporting structure would then increase only in proportion to the geometric enlargement of the rocket. The resultant fattening of the rocket would, however, tend to cause an increase in aerodynamic drag, which in turn would cause an increase in the propellant weight to gross weight ratio, thereby reducing to some extent the gain realized by keeping the ratio of basic weight to gross weight constant. It has also been shown that the propellant tank weight is proportional to the reciprocal of the propellant density for geometrically similar tanks. This indicates a serious weight disadvantage stemming from the use of low density fuels such as liquid hydrogen, when considered from the standpoint of the effect of propellant density on the basic weight to gross weight ratio.

The information presented above, summarizing previous work on scale effects as related to the overall satellite rocket design problem, has been most helpful in the formulation of the weight estimation methods used in this report. The problem of estimating the weight of a multi-stage satellite rocket, however, involves a multitude of details, each of which must receive adequate attention. Since the German A-4 (V-II) is the only long range rocket that has been successfully operated to date, it has been used as a base from which to work. The design of a multi-staged satellite rocket presents many problems, such as separation of stages during flight, and the automatic starting of the propulsive system during flight, which did not appear in the A-4, and for this reason it has been necessary to make several reasonable appearing assumptions in the development of the methods used. Another complicating factor is that, while the propellant weight to gross weight ratio is primarily a function of the rocket trajectory, it is related to the basic weight to gross weight ratio in such a manner that it is not possible to consider the effect of a given weight parameter on the basic weight to gross weight ratio independently of the effect of the same parameter on the propellant to gross weight ratio. This fact is clearly pointed out in the discussion given above of the effect of the propellant tank proportions upon the gross weight.

Since the parameters upon which the gross weight determinations were based have undergone continuous change and refinement during the study as a result of the findings from trajectory calculations which were carried on simultaneously, it will be found in the succeeding presentation that there will not be complete agreement of the values for gross weight derived in the different stages of the analysis. However, as the study progressed better values for the parameters became available and the final values presented here for the gross weights may be accepted with considerable confidence.

The following chapters in this report present the results of investigations made to determine the effect of some important weight parameters on the gross weight and the basic weight to gross weight ratio of a satellite rocket. Also, as a check on the analytical methods developed during this study and to verify the conclusions reached, a design study of a typical satellite rocket is presented.

II. WEIGHT STUDIES

1. Method of Analysis

In the quantitative analysis of any problem it is first necessary to have a convenient yardstick which may be used to determine an index of the relative worth of any given set of conditions when compared to several other sets of conditions. For the case of the design of a satellite rocket the gross weight represents such a convenient yardstick, and this unit of measure along with qualitative considerations of such items as cost, reliability, complexity of design, etc., has formed the basis of comparison. Further, having selected a unit of measure which enables the establishment of the relative worth of several proposals, it is necessary to have a method of determining with good precision information basic to the design of the selected satellite rocket. For these reasons, careful attention has been given to the development of a set of expressions from which the gross weight of the satellite rocket may be predicted. The derivation of these equations is given in Appendix I, and they are summarized for convenience in Table 1.

To compute the gross weight of a multi-staged satellite rocket, the gross weight of the final stage is first determined, then the weight of the preceding stage and so on until the weight of the initial stage is determined. The weight of the initial stage then gives the gross weight of the entire multi-staged satellite rocket. It should be noted that since the basic weight to gross weight ratio, and the propellant weight to gross weight ratio are related in such a manner that their variation with a given parameter of gross weight cannot be determined independently of one another, the following method of analysis must be used. The effect of the variation of a particular parameter of gross weight on the propellant weight to gross weight ratio (ν) is first determined from a consideration of the flight mechanics of the satellite rocket. Having determined the variation of the propellant weight to gross weight ratio with the parameter under consideration, consistent values of both ν and the particular parameter are entered in the equations summarized in Table 1. Since it has been found that the various parameters of gross weight, with the exception of the propellant weight to gross weight ratio, and the basic weight to gross weight ratio, are mutually independent to a certain degree, the equations are then solved holding all parameters except the ones being considered constant. This gives the variation in gross weight of the satellite rocket with the particular parameter being investigated. From this information the effect of the parameter on the basic weight to gross weight ratio may be deduced; furthermore, the optimum value, i.e., that value corresponding to the least gross weight, of the parameter may be isolated. Unfortunately, the solution of the equations giving the gross weight of the satellite rocket is an iteration, or trial and error process. No explicit solution has been discovered to date to eliminate the iteration requirement. Although the labor involved is considerable the computational work may be greatly facilitated by the use of a suitable computation form.

Table 1

SUMMARY OF WEIGHT EQUATIONS

Equations Governing the Weight of the Satellite Rocket

$$W = \frac{W_{L'}}{1 - \frac{\Sigma W_{P*}}{W} - \frac{\Sigma W_B}{W}} \qquad (1)$$

$$\lim \frac{W_B}{W} = 1 - \frac{W_{P*}}{W} \qquad (2)$$

$$W_j = (W_T + W_M + W_A + W_C + W_{P*} + W_{L*})_j \qquad (41)$$

$$(W_T)_N = .000155 \left[W_P \, n(\ell_T + \frac{r_T}{2})\right]_N + C + .05 \, W_{L'} \qquad (74)$$

$$(W_T)_N = .06322 \, [r_T \, (\ell_T + 2r_T)]_N + C + .05 \, W_{L'} \quad \text{(minimum gages)} \qquad (73)$$

$$(W_T)_x = .000155 \left[W_P n(\ell_T + \frac{r_T}{2})\right]_x \qquad (72)$$

$$(W_T)_1 = .000186 \left[W_P n_i(\ell_T + \frac{r_T}{2})\right]_1 \qquad (71)$$

$$(W_P)_j = \nu_j W_j \qquad (97)$$

$$(\ell_T)_N = \delta_N \qquad (125)$$

$$(\ell_T)_x = 1.90\delta_x \qquad (127)$$

$$(\ell_T)_1 = 1.90\delta_1 \qquad (127)$$

$$(r_T)_N = .564 \left(\frac{W_P}{\gamma_P \ell_T}\right)_N \qquad (123)$$

$$(r_T)_x = .437 \left[\frac{W_P}{\gamma_P \delta}\right]^{1/3} \qquad (124)$$

$$(r_T)_1 = .437 \left[\frac{W_P}{\gamma_P \delta}\right]^{1/3} \qquad (124)$$

$$(n_i)_j = (n_f)_j \, (1-\nu_j^*) \, (I_i/I_f)_j \qquad (81)$$

$$(W_M)_N = 0$$

$$(W_M)_x = .01\ W_x \qquad (82)$$

$$(W_M)_1 = .01\ W_1 \qquad (82)$$

$$(W_A)_N = \left[\frac{\bar{I}}{80} + \left(\frac{.000116}{\gamma_p} + .000812\right) p\right]_N \left(\frac{W_p}{t_b}\right)_N + 60 \qquad (92)$$

$$(W_A)_x = \left[\frac{\bar{I}}{80} + \left(\frac{.000116}{\gamma_p} + .000541\right) p\right]_x \left(\frac{W_p}{t_b}\right)_x + 55 \qquad (91)$$

$$(W_A)_1 = \left[\frac{\bar{I}}{80} + \left(\frac{.000116}{\gamma_p} + .000541\right) p\right]_1 \left(\frac{W_p}{t_b}\right)_1 + 55 \qquad (91)$$

$$(W_C)_N = 385 \left(\frac{W}{27,300}\right)_N^{7/6} \qquad (95)$$

$$(W_C)_x = 385 \left(\frac{W}{27,300}\right)_x^{7/6} \qquad (95)$$

$$(W_C)_1 = 1090 \left(\frac{W}{27,300}\right)_1^{7/6} \qquad (96)$$

$$(W_{p^*})_j = \nu_j^*\ W_j \qquad (102)$$

$$\nu_j^* = \epsilon_j \nu_j = \left(1.01 + \frac{.64p}{\gamma_p} \times 10^{-6}\right)_j \nu_j \qquad (101)$$

$$(W_{L^*})_N = W_{L'} = W_L + W_I + W_Q \qquad (104)$$

$$(W_{L^*})_x = (W)_{x+1} \qquad (103)$$

$$(W_{L^*})_1 = (W)_2 \qquad (103)$$

$$\left(\frac{W_p}{t_b}\right)_j = \left[\frac{n_f\ W(1-\nu^*)}{I_f}\right]_j$$

$$(W_B)_j = (W_T + W_M + W_A + W_C)_j \qquad (8)$$

Equations Governing the Satellite Rocket Geometry

$$m = 38.6541\ (\cot\phi)^{2/3} \qquad (106)$$

$$\rho_N = m \tan\phi \qquad (107)$$

$$\delta_N = \left[\left(.955\,\frac{W_P \tan\phi}{\gamma_P} + \rho_N^{\,3}\right) - \rho_N\right]_N \cot\phi \tag{108}$$

$$\lambda_N = m + \delta_N + s_N \tag{104}$$

$$\rho_x = \lambda_{x+1} \tan\phi \tag{113}$$

$$\delta_x = \left[\left(.573\,\frac{W_P \tan\phi}{\gamma_P} + \rho_x^{\,3}\right) - \rho_x\right]_x \cot\phi \tag{111}$$

$$\lambda_x = \lambda_{x+1} + \delta_x + s_x \tag{112}$$

$$\rho_1 = \lambda_2 \tan\phi \tag{113}$$

Implicit Equations giving λ_1

$$\left(.573\frac{W_P}{\gamma_P}\tan\phi\right)_1 = r_b^{\,3}\,(1 - U) - \rho_1^{\,3} - r_c^{\,3}\,U \tag{114}$$

$$U = \frac{\tan\phi}{\tan\psi} \tag{115}$$

$$\tan\psi = \frac{r_c - r_d}{s_1} \tag{116}$$

$$r_b = .5\,\lambda_1\frac{d}{\ell_o} \tag{117}$$

$$r_c = \frac{s_1(r_b - r_d)}{\lambda_1(1 - \ell/\ell_o)} + r_d \tag{118}$$

$$\delta_1 = \lambda_1 - \lambda_2 - s_1 \tag{119}$$

$$s_j = c + .0624\left(\frac{W_P}{P}\,p\right)_j^{1/3} + \left(2530 + .00653\,\frac{p}{\gamma_P}\,\frac{W_P}{t_b}\right)_j^{1/3} + (\ell_m)_j \tag{122}$$

2. Propellant Systems

The general expression for the gross weight of a satellite rocket is given by Eq. (1) as

$$W = \frac{W_L'}{(1 - \frac{\Sigma W_{P*}}{W} - \frac{\Sigma W_B}{W})}$$

where W is the gross weight of the satellite rocket, W_L, is the weight of the payload, where the term payload is defined as the weight of those items which are to be established on the satellite rocket's orbit and which do not contribute directly to the accomplishment of this object, ΣW_{p*} is the total weight of all propellants and fuels contained in the satellite rocket, and ΣW_B is the basic weight of the satellite rocket, where the term basic weight is defined as the weight of all items that are neither propellants, fuels nor payload. Since it has been shown that different propellant systems have inherently different specific impulses[2] and since the propellant weight to gross weight ratio ν is a function of the specific impulse[3] I (higher values of specific impulse giving lower values of the propellant weight to gross weight ratio), and also since the ratio $\Sigma W_{p*}/W$ is directly proportional to ν, it follows from Eq. (1) that, considering the effect of the specific impulse alone, different propellant systems will yield satellite rockets of different gross weights. Also it is apparent that the propellant system possessing the highest specific impulse (all other conditions being equal) will yield the satellite rocket of the least gross weight.

Now consider the effect of varying the characteristics of the propellant system on the basic weight to gross weight ratio. The two components of the basic weight to gross weight ratio that are affected by the two principal propellant characteristics, specific impulse I and propellant density ρ_p, are the ratio of the weight of the primary structure to gross weight, (W_T/W), and the ratio of the weight of the thrust producing equipment to gross weight (W_A/W). Neglecting small terms, the ratio W_T/W is given by an equation of the form

$$\frac{W_T}{W} = k\,\nu\,n\,(\ell_T + \frac{r_T}{2}) \ , \tag{3}$$

where ν is the propellant weight to gross weight ratio, n is the applied axial load factor, ℓ_T is the length of the propellant tank and r_T is the radius of the propellant tank. The ratio of the weight of the thrust producing equipment to the gross weight is given by an equation of the form

$$\frac{W_A}{W} = \left[k_1 I + \left(\frac{k_2}{\gamma_P} + k_3\right) p\right]\frac{\nu}{t_b} + k_4 \ , \tag{4}$$

where I is the specific impulse, γ_P is the specific weight of the propellants, p is the propellant pump discharge pressure, ν is the propellant weight to gross weight ratio and t_b is the duration of burning.

From Eqs. (3) and (4) it can be seen that

$$\frac{W_B}{W} = \phi\,(\nu,\ n,\ \ell_T,\ r_T,\ I,\ 1/\gamma_P,\ p,\ \text{and}\ \nu\,/t_b). \tag{5}$$

Now examine each of parameters of the ratio W_B/W listed in Eq. (5) with the purpose of determining how they are influenced by the propellant characteristics specific impulse I, and propellant density ρ_p. It has been previously stated that ν is a function of I. The axial load factor n, is an arbitrarily selected value for a given satellite rocket and is independent of I and ρ_p for the purpose of this argument.

Both ℓ_T and r_T are functions of the propellant tank volume which is in turn a function of the reciprocal of the propellant density; therefore ℓ_T and r_T are functions of the reciprocal of the propellant density ρ_p. The specific weight γ_p of the propellants is proportional to the propellant density; therefore, since the ratio W_B/W is a function of the reciprocal of the specific weight of the propellants, it is in turn a function of the reciprocal of the propellant density. The propellant pump discharge pressure p is a function of the rocket motor combustion pressure and the pressure losses in the propellant lines, and is a uniquely determined value for a given satellite rocket, hence not a function of I or ρ_p for the purpose of this discussion. The duration of burning t_b is a function of ν, I, and n. Since ν is a function of I, and n is independent of I and ρ_p for the purpose of this investigation, we may say that t_b is a function of I. Therefore from a consideration of Eq. (5) it may be said that the basic weight to gross weight ratio W_B/W is a function of the principal propellant characteristic's specific impulse, I, and propellant density, ρ_p, in the following manner.

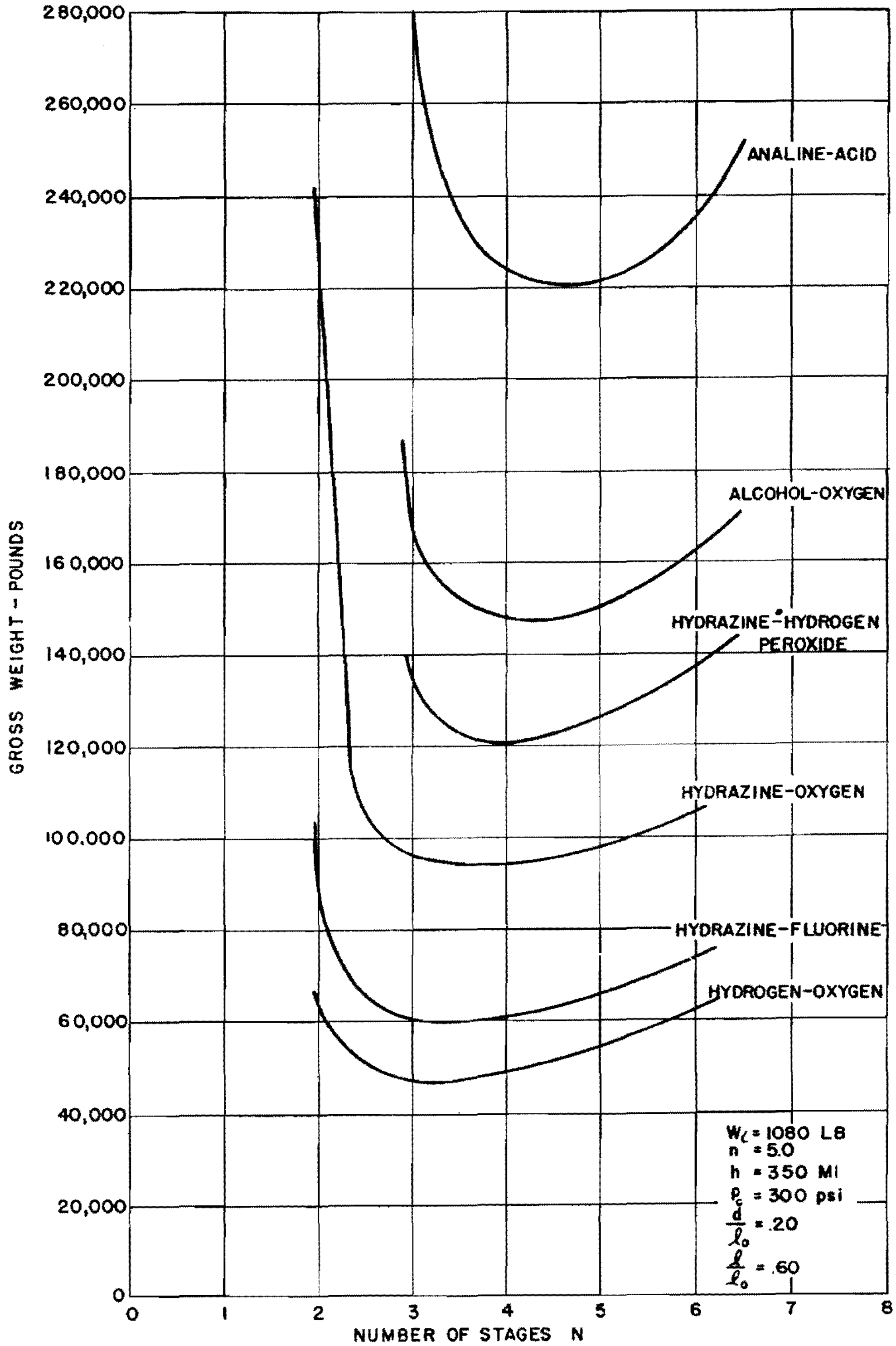
$$\frac{W_B}{W} = \phi\left(I, \frac{1}{\rho_p}\right) \tag{6}$$

The exact manner in which the gross weight and size of the satellite rocket will vary from one propellant system to another cannot be stated in simple general terms. However, it would appear, since the specific impulse has a greater effect on those items which tend to reduce the gross weight than on the items that tend to increase the gross weight with increasing specific impulse, and the propellant density affects items that tend to reduce the gross weight with increasing propellant density, that both high specific impulses and high propellant densities are to be desired. Fig. 2 presents graphically the variation of the gross weight with the number of stages for satellite rockets employing several different propellant systems. Here it may be seen that the propellant systems having generally higher specific impulses yield comparatively smaller gross weights. It is interesting to note that the hydrogen-oxygen system which has the highest specific impulse of all the propellant systems investigated also has the least propellant density. In spite of its low density this system appears to be the best from the gross weight standpoint; however, it possesses several undesirable characteristics which tend to eliminate the hydrogen-oxygen system from the list of possible 'practical' satellite rocket propellants.

3. Staging

Consider now the fundamental task of the satellite rocket, namely to accelerate a given mass of material called the payload to a velocity such that the payload will enter an orbit about the earth, that is, will become a satellite of the earth.

The satellite rocket consists essentially of three parts: the payload, the propellants, and those items which contain the payload and propellants and also make the chemical energy of the propellants available in a form most useful for accelerating the payload to its orbital velocity. Now, assume that the satellite rocket has the most efficient devices available today for the conversion of the chemical energy of the propellants to their most mechanically useful form. Furthermore, assume that structure of the satellite rocket is as efficient as present technology permits. From an examination of the process involved in establishing the payload on its orbit it may be seen that not only the payload is being accelerated to its orbital velocity,



GROSS WEIGHT vs NUMBERS OF STAGES

FIG. 2

but also the entire structural mass of the satellite rocket as well. It is apparent at once that if the mass of the satellite rocket's structure that must be accelerated along with the payload were reduced in some manner, more of the energy available in each unit mass of the propellants consumed would be available to accomplish the primary purpose of accelerating the payload to the orbital velocity and less would be wasted on the useless auxiliary process of accelerating the rocket structure. This introduces the concept of the staged satellite rocket where weight may be discarded once it has served its purpose and is no longer necessary. To illustrate this idea consider a two stage satellite rocket. The payload, that mass which is to be established on an orbit about the earth, will be carried in the second, smaller, stage of the two stage device; while the larger, initial, stage will carry the second stage as its 'payload'. The operation of a two stage satellite rocket may be described as follows: the initial stage will accelerate the second stage until all of its propellants are exhausted. At this point the initial stage will be discarded and the second stage will then add its velocity increment to that already given it by the first stage. By means of this device the total mass that must be accelerated to the orbital velocity is less than that which it would have been had the satellite rocket consisted of but a single stage.

The manners in which staging may be accomplished are manifold. Two systems seem to merit mention at this time. The first, arbitrarily called independent staging, assumes that each stage of the multi-staged satellite rocket shall be a unit within itself and not depend upon the operation of another stage to complete its function. Each stage will have its own propellant system, its own tanks and structure and its own rocket motors which operate independently of those in the other stages, the entire unit being jettisoned after it has served its purpose. The second type of staging, arbitrarily called dependent staging, assumes that each stage of a multi-staged satellite rocket operates as an ancillary of the final stage. Each stage will again have its own propellant system, propellant tanks and rocket motors which may be jettisoned when their usefulness has been fulfilled. However, the rocket motors of a dependently staged satellite rocket do not operate in an independent fashion. To illustrate the idea of dependent staging, again consider the case of the two stage satellite rocket. The rocket motors of the initial stage of this dependently staged satellite rocket are arranged in such a manner that they may be operated simultaneously with the rocket motors of the final stage. During the operation of the initial stage both sets of motors are utilized to supply the necessary thrust. At the termination of burning for the initial stage, the first stage equipment including the rocket motors associated with this stage are discarded, while the final stage rocket motors continue in operation. Since the weight of a rocket motor is a function of the thrust it produces, it follows that in the case of a dependently staged satellite rocket some saving may be made in the weight of the initial stage rocket motors, as they only supply a part of the thrust required, while in the case of the independently staged satellite the first stage rocket motors supply all of the thrust required by that stage.

A more detailed consideration of the mechanism required by a dependently staged satellite rocket indicates, however, that the weight saved in the initial stage rocket motors will be at least balanced by the increased mechanical complexity of the system. For this reason all investigations of the effects of staging have been made for an independently staged satellite rocket only.

Table 2

BASIC WEIGHT TO GROSS WEIGHT RATIO FOR SEVERAL SATELLITE ROCKETS HAVING DIFFERENT PROPELLANT SYSTEMS

Propellant System	Number of Stages: 2	3	4	5	6	Initial Conditions	
Hydrogen-Oxygen	.1728	.2081	.2414	.2789		W_L'	1080#
Alcohol-Oxygen		.1524	.1756	.1952	.2193	n	5.0
Hydrazine-Hydrogen Peroxide		.1529	.1667	.1905	.2155	h	350 mi
Analine-Acid		.1480	.1632	.1830	.2047	p_c	300 psi
Hydrazine-Fluorine	.1360	.1485	.1721	.2000		d/ℓ_o	.20
Hydrazine-Oxygen	.1322	.1458	.1698	.1896		ℓ/ℓ_o	.65

Fig. 2 and Table 2 give the variation of gross weight and the basic weight to gross weight ratio with the number of stages for several propellant systems. It will be seen in Fig. 2 that it is not possible to design a single stage satellite rocket having a payload of 500 pounds, an orbital altitude of 350 miles, and utilizing any of the propellant systems investigated. Further, it is apparent that there is an optimum number of stages, that is, a number of stages which will give the least gross weight for a satellite rocket utilizing any particular propellant system. Table 2 shows that in general the basic weight to gross weight ratio of the satellite rocket increases as the number of stages increases. This does not contradict the argument for applying the concept of staging to the satellite rocket. As was pointed out in the discussion above, the purpose of staging is to provide a device by which a greater portion of the available chemical energy of the propellants may be utilized to accomplish the ultimate purpose of the satellite rocket, that is, to establish a given mass of material, the payload, on an orbit at a given altitude above the earth's surface. The fact that staging does accomplish this very purpose is illustrated graphically by Fig. 2.

4. Orbit Altitude

Let W_B represent the basic weight of a multi-staged satellite rocket of any number of stages, where basic weight is defined as the weight of everything that is neither propellants, fuels, nor the payload of the final stage, then from Eq. (1)

$$W = \frac{W_L'}{\left(1 - \frac{\Sigma W_{P*}}{W} - \frac{\Sigma W_B}{W}\right)} ,$$

where W represents the gross weight of a multi-staged satellite rocket of any number of stages, W_L', the payload of the final stage, ΣW_{P*} the weight of all propellants and fuels contained in the satellite rocket, and ΣW_B the basic weight of the satellite rocket.

It has been found from studies of the flight mechanics of the satellite rocket that as the orbital altitude is increased, while the payload of the final stage is held constant, the ratio of the propellant weight required to the gross weight of the satellite rocket ν will also increase.[3]

The ratio of the primary structure weight to the gross weight of the satellite rocket, neglecting small terms, is given by an equation of the form

$$\frac{W_T}{W} = k\ \nu\ n\ (\ell_T + \frac{r_T}{2}) \tag{7}$$

where W_T/W represents the ratio of primary structure weight to gross weight, ν the ratio of propellant weight to gross weight, n the applied axial load factor, ℓ_T, the length of the propellant tank, and r_T the radius of the propellant tank. Since the ratio of the primary structure weight to gross weight constitutes a portion of the basic weight to gross weight ratio, it is obvious from Eqs. (1) and (7) that as ν increases so also will $\Sigma W_B/W$ increase.

The quantity $(1 - \Sigma W_{P*}/W - \Sigma W_B/W)$ in Eq. (1) cannot become negative as Eq. (1) would then yield negative gross weights for the satellite rocket, a physical absurdity. Therefore the limiting value of the quantity $1 - \Sigma W_{P*}/W - \Sigma W_B/W$ is zero, and this condition defines the limiting value of $\Sigma W_B/W$ which is given by Eq. (2) as

$$\lim\left(\frac{\Sigma W_P}{W}\right) = 1 - \frac{\Sigma W_{P*}}{W}\ .$$

Now it follows that as the ratio $\Sigma W_B/W$ approaches its limiting value of $1 - \Sigma W_{P*}/W$, the quantity $1 - \Sigma W_{P*}/W - \Sigma W_B/W$ in Eq. (1) will approach its limiting value of zero, and the gross weight of the satellite rocket will tend to infinity.

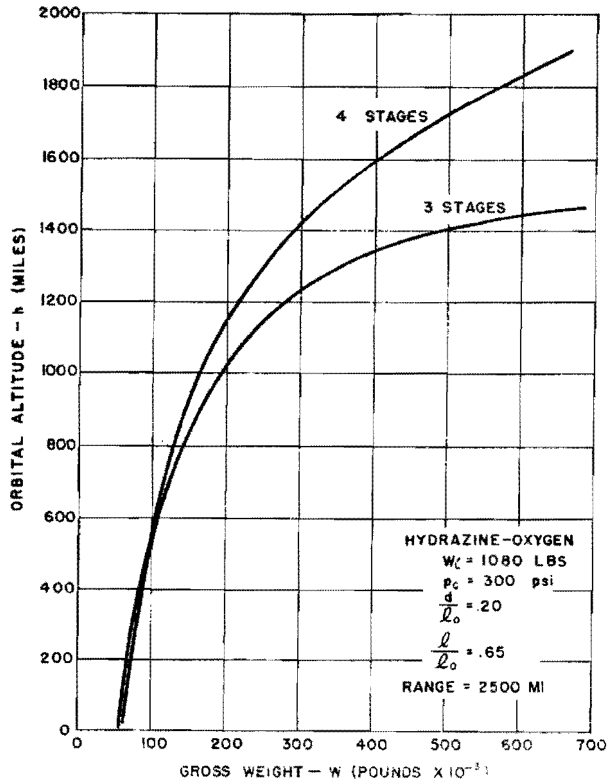
The above facts lead to the following conclusions. As the orbital altitude is increased for a fixed payload in the final stage of a given satellite rocket, the gross weight of the satellite rocket will also increase. Furthermore, for a fixed payload in the final stage, and a given number of stages, there is an altitude beyond which it is impossible to establish the satellite rocket on an orbit. Fig. 3 illustrates these conclusions for both a three and four stage hydrazine-oxygen satellite rocket. The limiting orbital altitude for a three stage hydrazine-oxygen rocket having a range of 2500 miles is given by Fig. 3 as being approximately 1500 miles. The term range may be defined as the distance along the surface of the earth from the launching site to the point where the satellite enters its orbital path.

5. Axial Load Factor

The ratio of the weight of the primary structure to gross weight for a given stage of a particular satellite rocket, when minimum structural material gage considerations do not apply, is given by Eq. (7) as

$$\left(\frac{W_T}{W}\right)_j = \left[k\ \nu\ n(\ell_T + \frac{r_T}{2})\right]_j\ ,$$

where W_T/W is the primary structure weight to gross weight ratio, ν is the propellant weight to gross weight ratio, n is the applied axial load factor, ℓ_T is the length of the propellant tank, and r_T is the radius of the propellant tank.



ORBITAL ALTITUDE VS GROSS WEIGHT

FIG. 3

The gross weight of a given stage is given by

$$W_j = \left(\frac{W_{L^*}}{1 - \frac{W_{P^*}}{W} - \frac{W_B}{W}} \right)_j \text{ , (from Eq. [1])}$$

where W is the gross weight, W_{L^*} the payload of the given stage, W_{P^*} the weight of propellants and fuels, and W_B is the basic weight.

The basic weight, W_B, is in turn defined as

$$(W_B)_j = (W_T + W_M + W_A + W_C)_j \tag{8}$$

where W_T is the primary structure weight, W_M the weight of miscellaneous structure, W_A the weight of the thrust producing equipment, and W_C the weight of control equipment.

From a consideration of foregoing equations it is clear that since W_T/W is a function of n and since the value of W_B/W and hence the gross weight W depends upon W_T/W, the gross weight W of a given stage of the satellite rocket is a function of the applied axial load factor, n.

Further, for a given set of conditions, that is, for a given stage of a particular satellite rocket, both the size of the propellant tanks and the propellant weight to gross weight ratio will be fixed. Therefore, the weight of the primary structure may be expressed as

$$(W_T)_j = (k\ Wn)_j \tag{9}$$

where the product nW represents the load applied to the primary structure. Also, the gross weight of the stage may be expressed by Eq. (41) as

$$(W_j) = (W_T + W_M + W_A + W_C + W_{P^*} + W_{L^*})_j \text{ ,}$$

where it may be seen that the gross weight of the stage is equal to the sum of the weight of the primary structure and other terms which are not directly a function of the applied load. Therefore, since from Eq. (9) the weight of the primary structure increases as the applied load increases, it follows, from Eq. (41), that the gross weight of the stage also increases as the load applied to the primary structure increases.

The problem of selecting the optimum value of the axial load factor n for a given stage involves a number of considerations. First, the programming of the acceleration in the given stage must be such that the maximum load applied to the structure will be as small as possible during the acceleration period. This follows from the discussion given above where it was shown that the weight of the primary structure and hence the gross weight of the stage increases as the applied load increases. Second, having determined the best manner in which to program the acceleration of a given stage, it then becomes necessary to find the proper maximum value of the axial load factor to be applied to that stage.

Consider first the programming of the acceleration for a particular stage. As has been pointed out above, the gross weight W of the stage is a function of the load applied to the primary structure which may be expressed as the product of the gross weight of the stage W and the applied axial load factor n.

Now consider the various manners in which both the gross weight of the stage W and the load factor n may be varied with time. The two cases that are of primary interest are, first, a constant mass rate of propellant flow, representing the simplest condition from the standpoint of rocket motor design, since throttling is not involved; second, constant load factor, which has been shown to have certain desirable characteristics from a consideration of the flight mechanics of the satellite rocket.

Considering now the case of constant mass rate of propellant consumption, the general expression for load factor[3] is given by

$$n_t g_s = \frac{\frac{dm_p}{dt} g_s I}{M_0 - \int_0^t \epsilon \frac{dm_p}{dt} dt}, \tag{10}$$

where n_t is the applied load factor at time t, g_s is the standard acceleration of gravity at sea level, I the specific impulse, m_p the mass of propellants discharged at time t, dm_p/dt the time rate of propellant flow, $\epsilon\ dm_p/dt$ the time rate of change of mass of the stage, and M_0 the initial mass of the stage.

For constant mass rate of propellant flow, the time rate of propellant flow may be written as

$$\frac{dm_p}{dt} = \frac{M_p}{t_b}, \tag{11}$$

where M_p is the initial mass of the propellants and t_b is the length of the burning period in seconds. Therefore,

$$n_t = \frac{(M_p/t_b)\ I}{M_0 - \int_0^t \epsilon \frac{M_p}{t_b} dt} = \frac{(M_p/t_b)\ I}{M_0 - \epsilon \frac{M_p}{t_b} t} \tag{12}$$

The ratio of the instantaneous load factor n_t to the final load factor n_f $(t = t_b)$ then may be expressed as

$$\frac{n_t}{n_f} = \frac{M_0 - \epsilon M_p}{M_0 - \epsilon M_p \left(\frac{t}{t_b}\right)} . \tag{13}$$

The ratio of the instantaneous gross weight of the stage W_t, to the initial gross weight W, may be written as

$$\frac{W_t}{W} = \frac{W - \epsilon(W_p/t_b)\ t}{W} . \tag{14}$$

The applied load Q is then given by

$$Q_t = \frac{n_t}{n_f} \cdot \frac{W_t}{W} (n_f W) . \tag{15}$$

Fig. 4 gives the variation of n_t/n_f, W_t/W and $Q_t/n_f W$ with t/t_b for the case of constant mass rate of propellant consumption.

Now consider the case of constant load factor burning. Again the general expression for load factor is

$$ng_s = \frac{\frac{dm_p}{dt} g_s \quad I}{M_0 - \int_0^t \epsilon \frac{dm_p}{dt} dt} ,$$

which may be rearranged to read

$$\frac{n}{I} = \frac{\frac{1}{M_0}\frac{dm_p}{dt}}{1 - \epsilon \int_0^t \frac{1}{M_0}\frac{dm_p}{dt}\, dt} = k, \text{ a constant.} \tag{16}$$

Let $x = \frac{1}{M_0}\frac{dm_p}{dt}$, then

$$k = \frac{x}{1 - \epsilon \int_0^t x\, dt} ; \tag{17}$$

and let $y = \int_0^t x\, dt$, then from Eq. (17)

$$k\ dt = \frac{dy}{1-\epsilon y} , \tag{18}$$

$$-\frac{\ln(1-\epsilon y)}{\epsilon} = kt ,$$

and

$$1 - \epsilon y = e^{-\epsilon k t} . \tag{19}$$

Therefore, since $y = \int_0^t x \, dt$, $x = \frac{1}{M_0}\frac{dm_p}{dt}$ and $k = \frac{n}{I}$,

$$\frac{M_0 - \epsilon m_p}{M_0} = exp. \frac{-\epsilon n}{I} t \quad ; \tag{20}$$

and since $M_t = M_0 - \epsilon m_p$, the ratio of the instantaneous gross weight W_t to the initial gross weight, W, may be written as

$$\frac{W_t}{W} = exp. \frac{-\epsilon n}{I} t \quad . \tag{21}$$

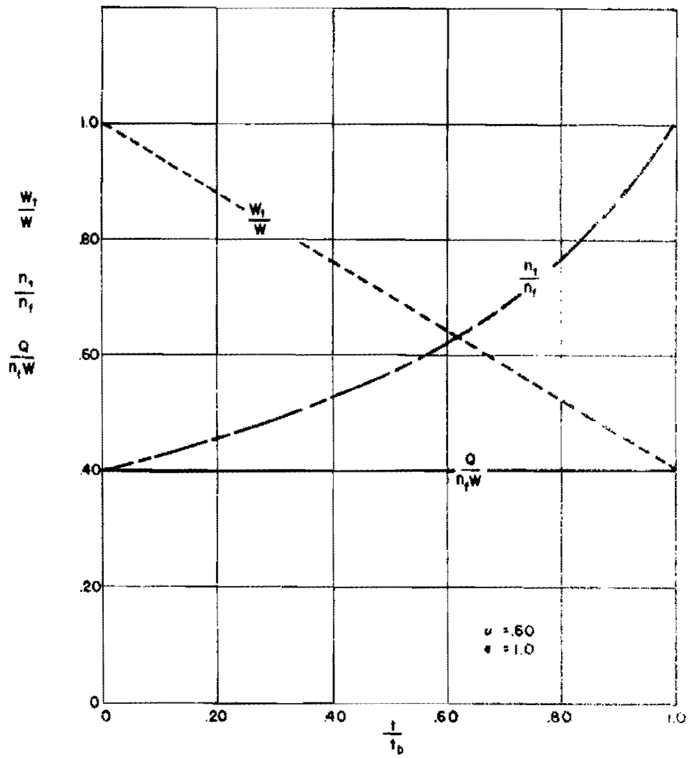
The applied load is again given by Eq. (14) as

$$Q_t = \frac{n_t}{n_f} \cdot \frac{W_t}{W} (n_f W) .$$

Fig. 5 gives the variation of n_t/n_f, W_t/W and $Q_t/n_f W$ with t/t_b for the case of constant load factor.

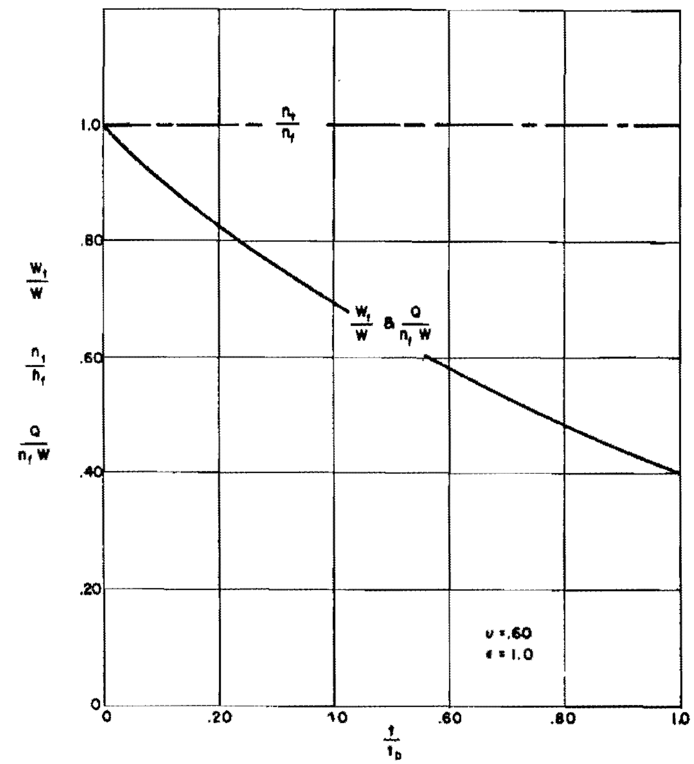
Since Eq. (9) states that the weight of the primary structure is a linear function of the applied load, the weight of primary structure W_t increases as the applied load increases, and since the initial gross weight W of the stage increases as the weight of the primary structure increases, it is obvious from a comparison of Figs. 4 and 5 that the condition of constant mass rate of propellant consumption is the most efficient; that is, this condition will correspond to a stage of the least gross weight.

Now consider the problem of selecting the proper maximum load factor for a particular stage of a multi-staged satellite rocket. The optimum value of n for a given stage of the satellite rocket will depend principally upon the relative importance of the variations of two parameters with n, namely ν (propellant weight to gross weight ratio) and W_B/W (basic weight to gross weight ratio). In the initial stage both ν and W_B/W are very sensitive to small changes in n, ν decreasing rapidly with n and W_B/W increasing rapidly with n. For intermediate stages, however, the variation of both ν and W_B/W has been found to be less than the corresponding variation in the initial stage. Furthermore the decrease in ν with increasing n is much less significant than the corresponding increase of W_B/W with increases in n. This indicates that the optimum value of n for the intermediate stage will be less than the optimum value of n for the initial stage. This follows from the fact that the optimum value of n (that value of n corresponding to the least gross weight of the stage) will be that at which the sum of the propellant weight to gross weight ratio ν, and the basic weight to gross weight ratio W_B/W is a minimum. From a consideration of the weight of the structure required to carry the applied load, the reasoning applied to the intermediate stages would also apply to the final stage. However, it has been found



LOAD FACTOR, GROSS WEIGHT, AND APPLIED LOAD VS TIME FOR CONSTANT MASS RATE OF PROPELLANT CONSUMPTION

FIG. 4

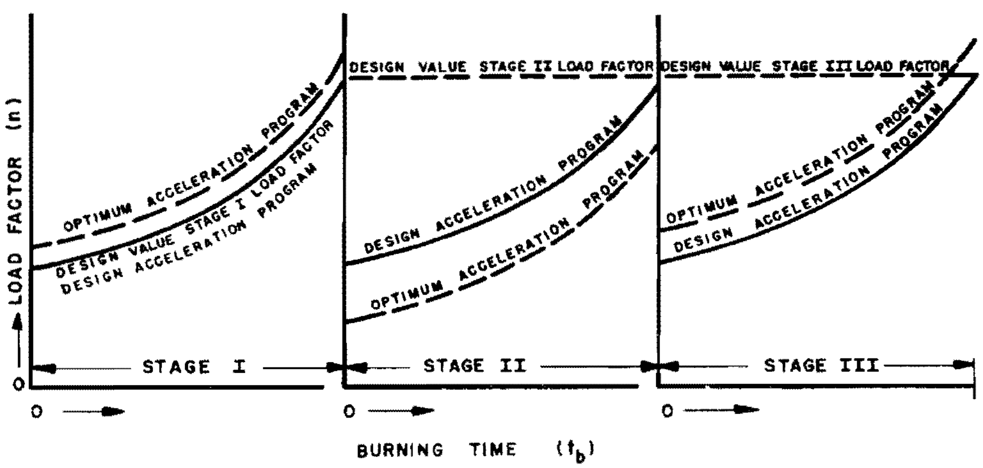


LOAD FACTOR, GROSS WEIGHT, AND APPLIED LOAD VS TIME FOR CONSTANT LOAD FACTOR

FIG. 5

that the weight of the final stage is influenced more by minimum gage considerations than it is by the applied load factor. Hence the load factor applied to the final stage could easily be greater than that applied to the initial and intermediate stages.

Since the intermediate stages of the satellite rocket must be designed to resist the maximum load factor applied by the initial stage, it is impractical, of course, to program the acceleration of the satellite rocket in such a manner that both the initial and intermediate stages operate at their respective optimum load factors. It would be possible though to operate the final stage at its optimum load factor. However, the variation of ν with load factor in the final stage is much less significant than it is in the initial or intermediate stages. Therefore, for the purpose of simplicity, the final stage, the intermediate stages, and the initial stage have all been designed to operate at the same maximum axial load factor. Fig. 6 gives a schematic representation of the optimum and design axial load factors for a three stage satellite rocket.

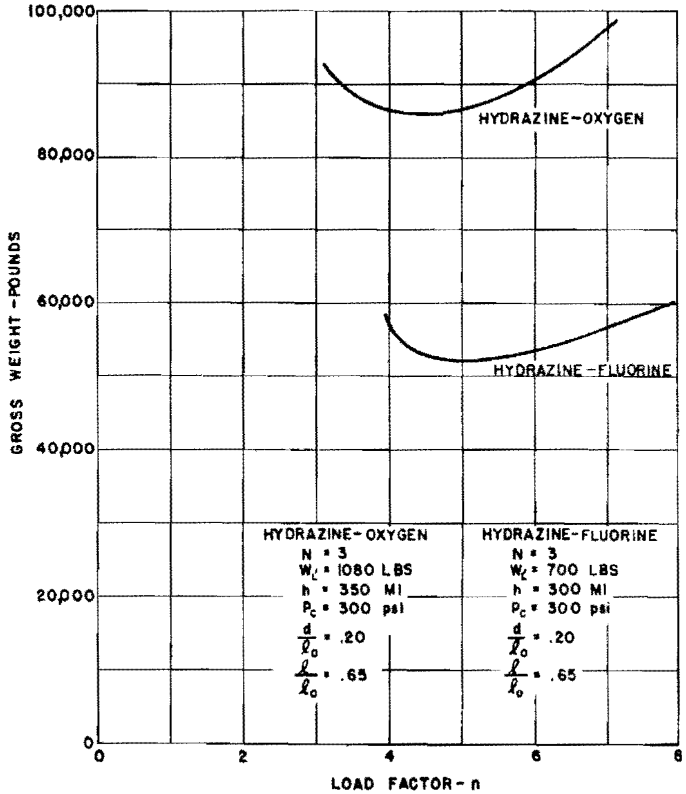


SCHEMATIC DIAGRAM

GIVING VARATION OF DESIGN ACCELERATION PROGRAM, OPTIMUM ACCELERATION PROGRAM AND DESIGN VALUE OF LOAD FACTOR WITH TIME FOR EACH STAGE OF A THREE STAGE SATELLITE ROCKET

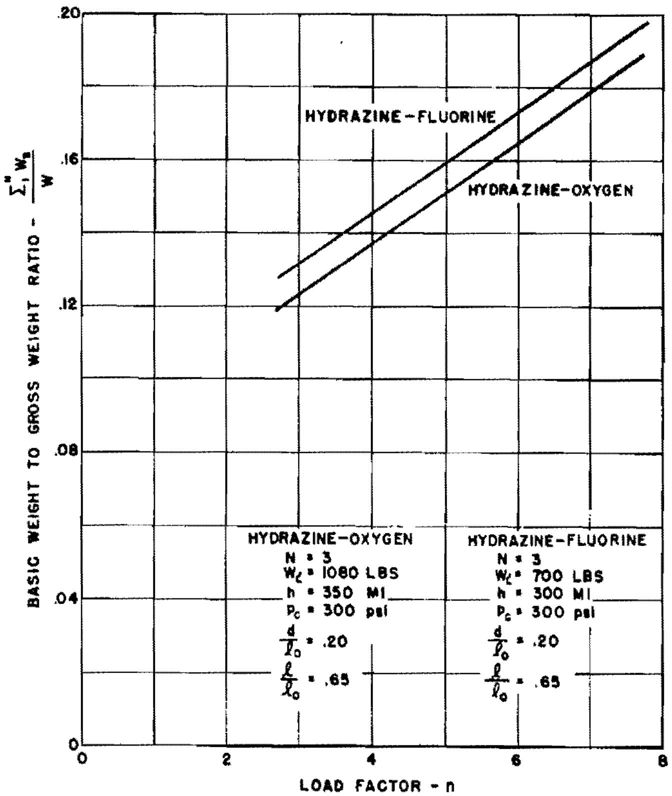
FIG. 6

It may be seen from an examination of Fig. 6 that the design acceleration program for the satellite rocket as a whole must be a compromise between the higher values of n desired for stages I and III and the lower value of n desired for stage II. The variation of gross weight with axial load factor n is given in Fig. 7 for two three stage satellite rockets having different propellant systems and different orbital altitudes. This figure indicates that the optimum axial load factor (that corresponding to the least gross weight) for a three stage hydrazine-fluorine satellite rocket having an orbital altitude of 300 miles and a final stage payload W_L' of 700 pounds is about 5.0. Also the optimum axial load factor for a three stage hydrazine-oxygen satellite rocket having an orbital altitude of 350 miles and a final stage payload W_L' of 1080 pounds is shown by Fig. 7 to be approximately 4.5. It is interesting to note that these accelerations are well within the limit which a man can withstand, and are also about 20% lower than the value of 6.5 given in ref. 1.



GROSS WEIGHT VS AXIAL LOAD FACTOR

FIG. 7



BASIC WEIGHT TO GROSS WEIGHT RATIO VS AXIAL LOAD FACTOR

FIG. 8

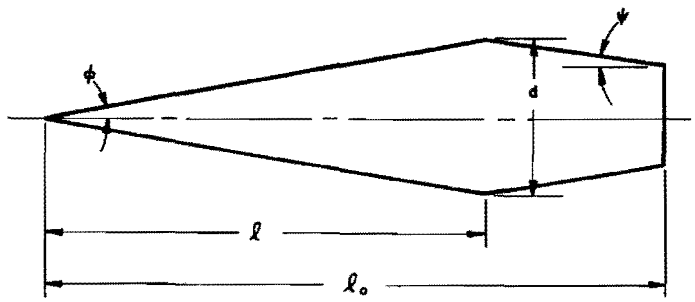
Fig. 8 shows that the basic weight to gross weight ratio is a linear function of axial load factor, the basic weight to gross weight ratio increasing as the axial load factor increases.

6. Body Shape Parameters

It was pointed out in the introductory remarks that previous investigations[1] have indicated that the gross weight of the satellite rocket is greatly influenced by its shape. Aerodynamic considerations dictate the shape of the satellite rocket body to be long and slender, thereby minimizing the aerodynamic drag. Structural considerations, however, require that the satellite rocket body should be short and fat, thereby reducing the structural load due to inertia (volumetric) forces which for a constant volume increase as the length of the body increases. When two or more conflicting sets of requirements occur in the same problem, there exists a single particular combination of these requirements which will represent the optimum condition. The optimum condition in this case is that corresponding to the satellite rocket of the least gross weight, and the purpose of this investigation is to find the satellite rocket configuration which will result in the least gross weight. In an earlier investigation of this problem[16] an approximation of the variation of the gross weight of the satellite rocket with two simplified shape parameters was determined. The shape parameters used in that study were ϕ, the half angle subtended at the nose of the satellite rocket by lines drawn to the extremeties of its maximum diameter, and ℓ/ℓ_0, the ratio of the length of this section between the nose and the maximum diameter to the overall length. A later, more precise, determination of the effect of aerodynamic drag on the propellant weight to gross weight ratio[4] indicated that the simplified shape parameters would be more convenient to use if they were expressed as d/ℓ_0, the ratio of the maximum satellite rocket diameter to the overall length, and ℓ/ℓ_0 the ratio of the length of the section between the nose and the maximum diameter to the overall length of the satellite rocket body. Fig. 9 illustrates the simplified shape parameters used in the current study to approximate the best shape of the satellite rocket body at supersonic speeds. The final shape of the satellite rocket will of course represent a compromise between the most desirable shapes at supersonic and subsonic speeds, and can only be determined by a development program. It is felt, however, that the final shape will not differ greatly from that defined by this simplified study. In general, the shape of the satellite rocket body may be stated to be roughly that of a right circular cone increasing to some maximum diameter, and then decreasing in diameter to a value at the aft end such that the base drag will be a minimum.

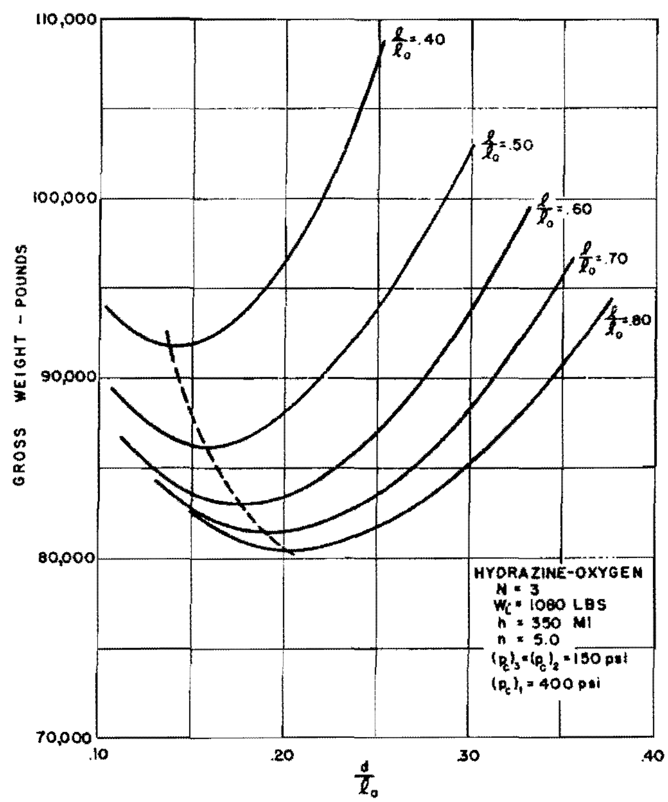
Fig. 10 gives the variation of the gross weight of the three stage hydrazine-oxygen satellite rocket with the parameter d/ℓ_0, the ratio of the maximum diameter to the overall length for various values of ℓ/ℓ_0, the ratio of the length of the section between the nose and the maximum diameter to the overall length.

To isolate the effect of the ratios d/ℓ_0 and ℓ/ℓ_0 on the gross weight, the variation of gross weight with ℓ/ℓ_0 for the optimum combination of d/ℓ_0 and ℓ/ℓ_0 is given in Fig. 11, and the variation of gross weight with ℓ/ℓ_0 for the optimum combination of d/ℓ_0 and ℓ/ℓ_0 is given in Fig. 12. From an examination of these two figures it may be seen that within the range of variables investigated the gross weight will generally decrease for increasing values of both d/ℓ_0 and ℓ/ℓ_0. Also, for the range of the variables investigated, Figs. 11 and 12 indicate the best value of d/ℓ_0 to be about .20 and the best value of ℓ/ℓ_0 to be about .80. Since the value

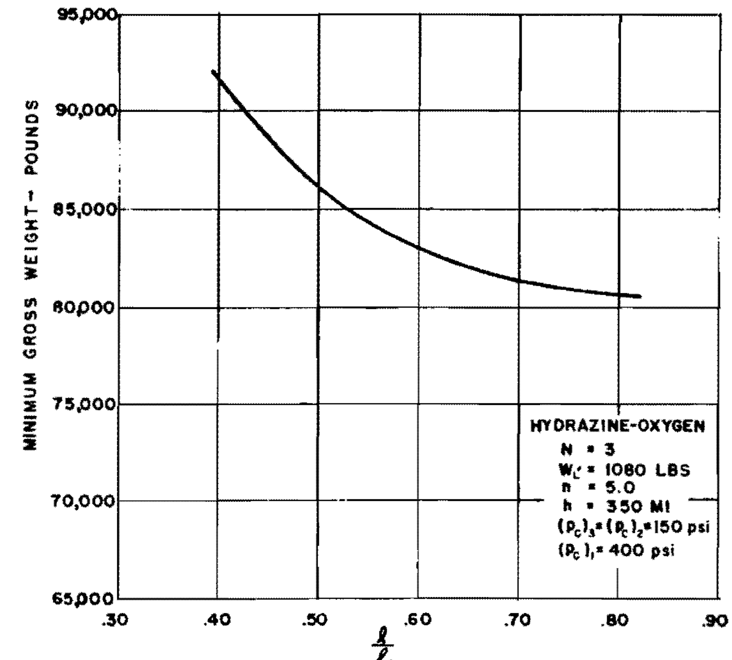


2ϕ = ANGLE OF DIVERGENCY
2ψ = ANGLE OF CONVERGENCY
ℓ = LENGTH OF DIVERGENT SECTION
d = MAXIMUM DIAMETER
ℓ_o = OVERALL LENGTH

SIMPLIFIED SHAPE PARAMETERS
(FOR SATELLITE ROCKET BODY)
FIG. 9

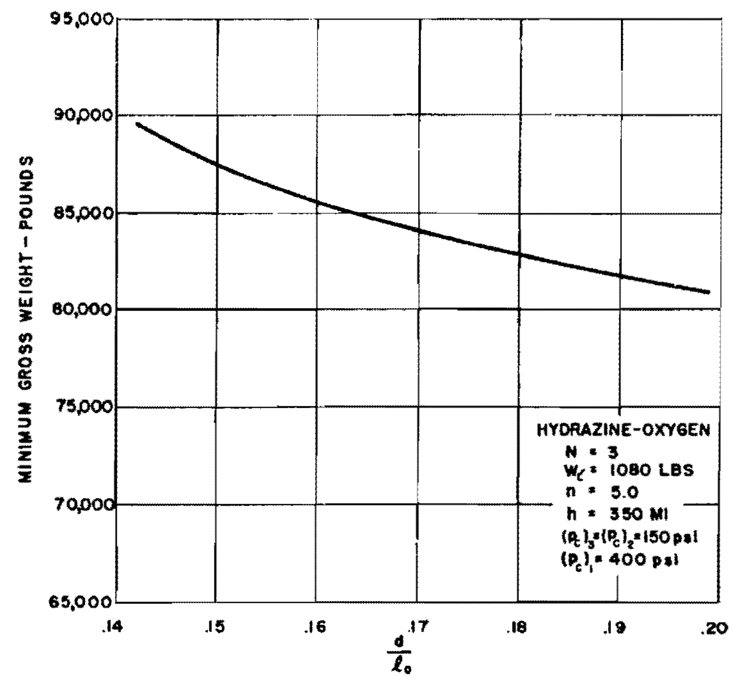


GROSS WEIGHT VS $\frac{\ell}{\ell_o}$ AND $\frac{d}{\ell_o}$
FIG. 10



MINIMUM GROSS WEIGHT vs $\frac{\ell}{\ell_o}$

FIG. 11



MINIMUM GROSS WEIGHT vs $\frac{d}{\ell_o}$

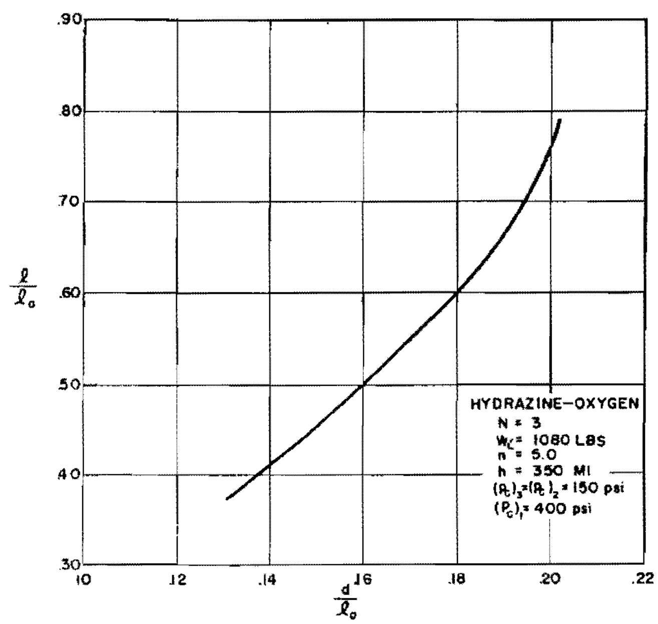
FIG. 12

ℓ/ℓ_0 = .80 is the upper limit of the range of investigation of this variable, the implication exists that perhaps higher values of the ratio ℓ/ℓ_0 may yield lower gross weights. It should be pointed out, however, that the aerodynamic effect of the angle of convergency, 2ψ, on the ratio ℓ/ℓ_0 has not been specifically included in this study. This angle cannot be made arbitrarily large, but is limited to a value such that the flow past the vehicle body will not separate from the vehicle body in the convergent section. This in turn limits the maximum value that ℓ/ℓ_0 may assume, and it is believed at this time to be about .80. For this reason a reasonable value of ℓ/ℓ_0 has been assumed in the design study appearing in a later section of this report.

Fig. 13 gives the relation between ℓ/ℓ_0 and d/ℓ_0 which will yield a satellite rocket of minimum gross weight.

From this curve, for given values of either d/ℓ_0, or ℓ/ℓ_0, a corresponding value of ℓ/ℓ_0 or d/ℓ_0 may be selected.

The selected value of either d/ℓ_0 or ℓ/ℓ_0 then represents the optimum combination of these two variables for the given condition, and the combination of d/ℓ_0 and ℓ/ℓ_0 determined in this manner will yield a satellite rocket of the least gross weight for the imposed condition.

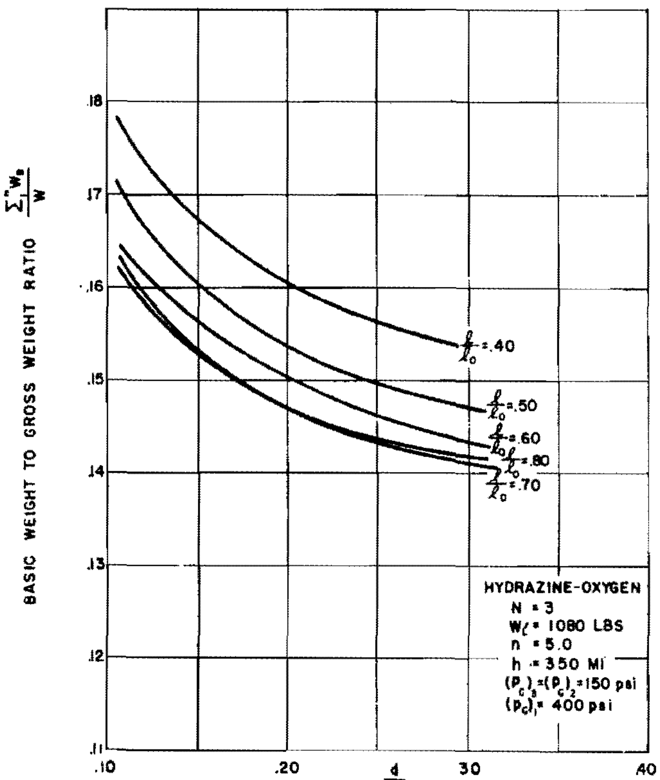


RELATION BETWEEN $\frac{\ell}{\ell_0}$ AND $\frac{d}{\ell_0}$ TO GIVE A SATELLITE ROCKET OF MINIMUM GROSS WEIGHT

FIG. 13

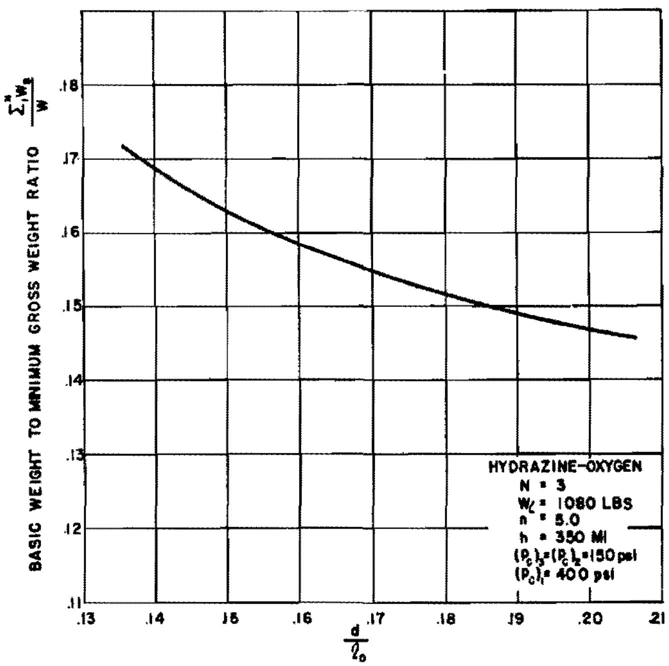
Fig. 14 gives the variation of the basic weight to gross weight ratio with d/ℓ_0 and d/ℓ_0 for the three stage hydrazine-oxygen rocket. Here it can be seen that the basic weight to gross weight ratio in general decreases with increasing value of both d/ℓ_0 and ℓ/ℓ_0. It can be seen, however, that this does not hold true when the value ℓ/ℓ_0 approaches 70% to 80%. For smaller values of d/ℓ_0, that is below about 19.5%, the basic weight ratio continues to decrease with increasing values of both ℓ/ℓ_0 and d/ℓ_0; however, above $d/\ell_0 \approx$ 19.5% it may be seen from Fig. 14 that ℓ/ℓ_0 of .80 gives higher values of the basic weight to gross weight ratio than does ℓ/ℓ_0 = .70.

Again, to isolate the effect of the ratios ℓ/ℓ_0 and d/ℓ_0, the variations of the basic weight to the minimum gross weight ratio with d/ℓ_0 is given in Fig. 15 and the variation of the basic weight to the minimum gross weight ratio with ℓ/ℓ_0 is given in Fig. 16. From Fig. 15 it can be seen that as the ratio d/ℓ_0 increases, the basic weight to minimum gross weight ratio decreases, although at a decreasing rate. Fig. 16 shows that as the ratio ℓ/ℓ_0 increases, the basic weight to minimum gross weight ratio generally decreases, however, only very small gains can be made by increasing the ratio ℓ/ℓ_0 above .70.



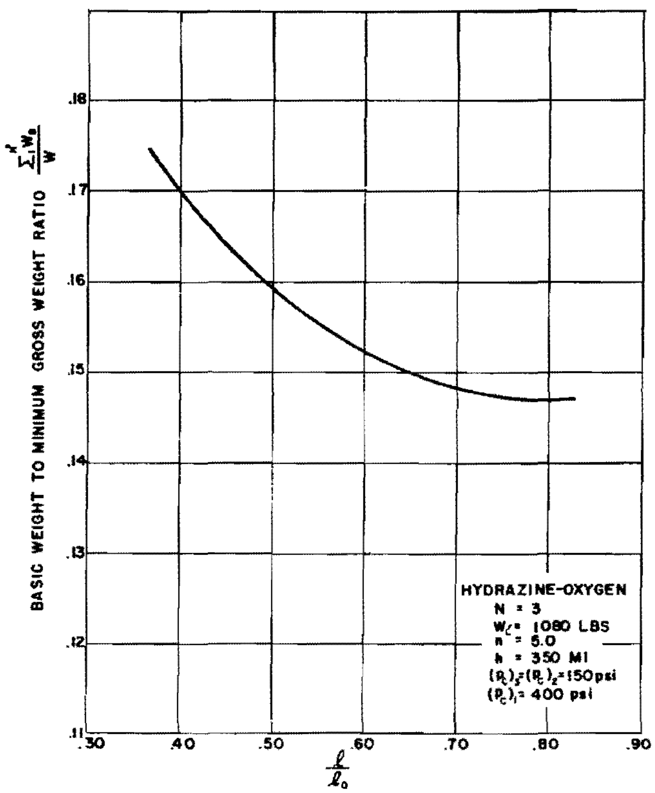
BASIC WEIGHT TO GROSS WEIGHT RATIO
$\frac{\ell}{\ell_o}$ AND $\frac{d}{\ell_o}$

FIG. 14



BASIC WEIGHT TO MINIMUM GROSS WEIGHT
RATIO VS $\frac{d}{\ell_o}$

FIG. 15



BASIC WEIGHT TO MINIMUM GROSS WEIGHT RATIO VS $\frac{\ell}{\ell_0}$

FIG. 16

7. Rocket Motor Combustion Pressure

Studies of the characteristics of several liquid propellant systems[2] for the satellite rocket indicate that significant gains in specific impulse can be realized, while not exceeding reasonable temperature limits, if increased rocket motor combustion pressures are utilized. Since the propellant weight to gross weight ratio ν is a function of the specific impulse[3] (higher specific impulses I yield lower values of ν) it follows that some gains in the satellite rocket gross weight may be realized by going to higher combustion pressures. This follows directly from the general gross weight equation

$$W = \frac{W_L'}{\left(1 - \frac{\Sigma W_{P*}}{W} - \frac{\Sigma W_B}{W}\right)} \quad ,$$

where W is the gross weight of the satellite rocket, W_L' is the weight of the payload in the final stage, $\Sigma W_{P*}/W$ is the ratio of the total propellant and fuel weight to gross weight, and $\Sigma W_B/W$ is the ratio of the basic weight to gross weight.

Since $\Sigma W_{P*}/W$ is a function of ν, any reduction in ν which did not affect $\Sigma W_B/W$ adversely would result in a reduction of gross weight. The ratio of basic weight to gross weight $\Sigma W_B/W$, however, is a function of two variables in this case which tend to oppose one another. Neglecting small terms, the weight of the primary structure is given by an equation of the form

$$W_T = kW_P n(\ell_T - \frac{r_T}{2}) \qquad \text{(eq. [3])}$$

where W_T is the weight of the primary structure, W_P is the weight of the propellants, n is the applied axial load factor, ℓ_T is the length of the propellant tank and r_T is the radius of the propellant tank. Since W_P is a linear function of ν it follows that for smaller values of ν, W_T will be smaller. The weight of the thrust producing equipment, however, is given by an equation of the form

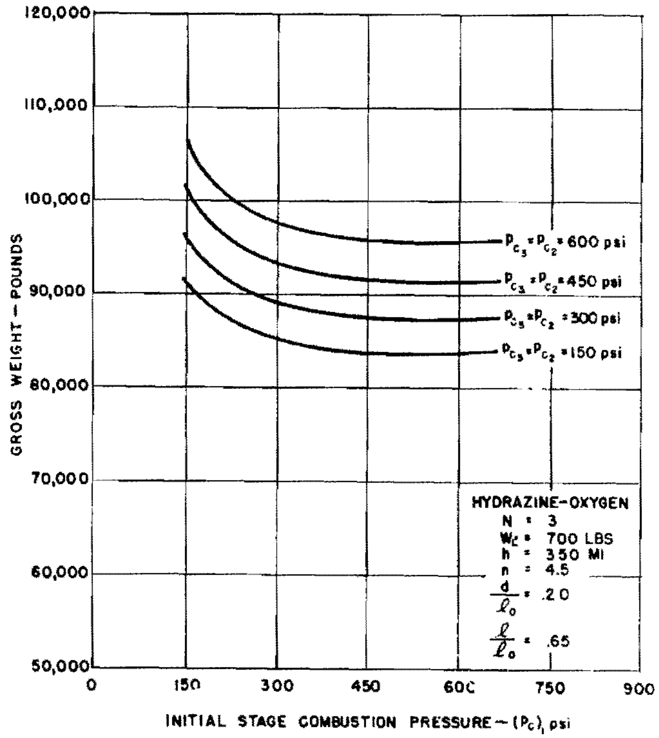
$$W_A = (k_1 I + k_2 p) \frac{W_P}{t_b} + k_3 , \qquad (22)$$

where W_A is the weight of the thrust producing equipment, I is the specific impulse, p is the propellant pump discharge pressure, W_P is the weight of propellants, and t_b is the duration of the burning period. Since the propellant pump discharge pressure is a function of the combustion pressure p_c, higher values of the combustion pressure give higher propellant pump discharge pressures. The reduction in the propellant weight W_P, due to an increase in the specific impulse I, by increasing combustion pressure may be more than offset by the increase in the weight of the thrust producing equipment W_A, due to an increased propellant pump discharge pressure. Since the basic weight W_B is a function of the sum of the primary structure weight and the weight of the thrust producing equipment, it is clear that the ratio $\Sigma W_B/W$ is not independent of the combustion pressure. It is difficult to predict the effect of increasing the combustion pressure on the basic weight to gross weight ratio, since the effect depends upon the relative magnitude of the increase in specific impulse realized due to a corresponding increase in combustion pressure.

Figs. 17 and 18 give the results of an investigation of the effect of varying the combustion pressure on the gross weight and the basic weight to gross weight ratio for a three stage hydrazine-oxygen satellite rocket. It may be seen from a study of Fig. 17 that optimum combustion pressures (those corresponding to the minimum satellite rocket gross weight) for the three stage hydrazine-oxygen satellite rocket are a comparatively low combustion pressure, 150 psi, in the second and third stages, and a comparatively high combustion pressure, 600 psi, in the initial stage. Fig. 18 indicates that the basic weight to gross weight ratio generally tends to increase as the combustion pressure increases, for the range of values investigated. It appears, however, that the rate of increase of the basic weight to gross weight ratio diminishes as the combustion pressure increases.

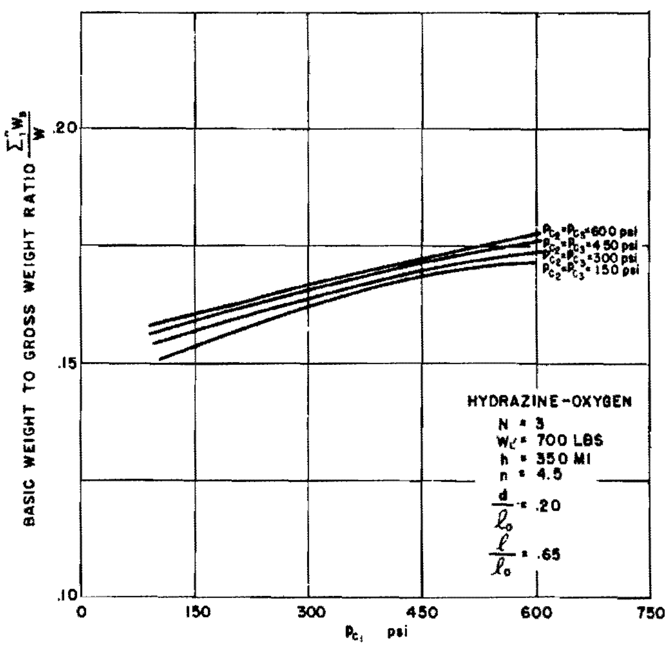
This would indicate that the decreasing propellant weight to gross weight ratio ν, due to higher combustion pressure, has an increasingly greater effect on the basic weight to gross weight ratio W_B/W, than does the increase in specific impulse and propellant pump discharge pressure as the combustion pressure increases.

Significant gains in the reduction of gross weight can be realized by utilizing the proper combustion pressure in each stage. In the light of this information perhaps a more desirable propellant, from the standpoint of danger to operating personnel.



GROSS WEIGHT vs COMBUSTION PRESSURE

FIG. 17



BASIC WEIGHT TO GROSS WEIGHT RATIO
vs COMBUSTION PRESSURE

FIG. 18

than the hydrazine-fluorine combination could be utilized for the proposed satellite rocket without increasing the gross weight beyond reasonable limits.

For this reason the effect of varying the combustion pressure on a satellite rocket employing three different propellant systems was investigated. Figs. 19 and 20 present the results of this investigation, Fig. 19 giving the variation of gross weight with combustion pressure for a three stage satellite rocket employing, first, a hydrazine-fluorine propellant system in all three stages, second, a hydrazine-oxygen propellant system in the first stage and hydrazine-fluorine in the latter two stages, and third, a hydrazine-oxygen propellant system in all three stages. Fig. 20 gives the variation of the basic weight to gross weight ratio with gross weight for the above conditions. Fig. 19 indicates that the use of a hydrazine-oxygen system in all three stages in preference to a hydrazine-fluorine system will increase the gross weight of the satellite rocket by about 30%. However, it is felt that the degree by which the operational hazards would be reduced by the use of oxygen in preference to fluorine as an oxidizer more than offset this increase in gross weight.

8. Protection Against Meteorite Collision

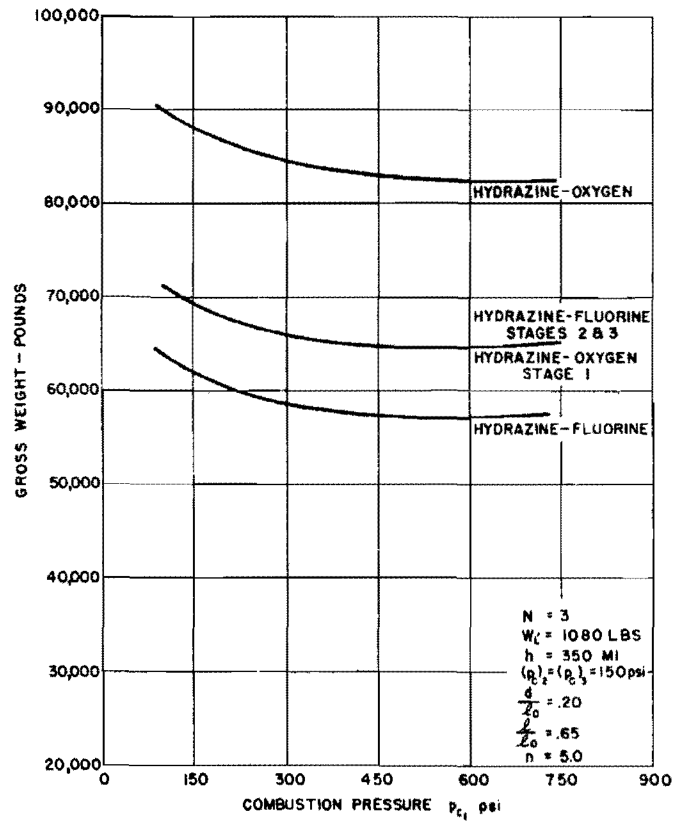
The importance of resolving the problem of the necessity of protecting the satellite rocket against possible damage due to meteorite collision is at once apparent when it is considered that the weight of the armor required may substantially alter the performance of the satellite rocket. The problem may be considered to consist of three parts: first, a study of the probability of a collision between the satellite rocket and a meteorite of a given magnitude; second, an investigation of the size and velocities of meteorites as a function of meteorite magnitude; and third, a study of the mechanism of penetration of metal plates by high speed particles.

a. Probability of Satellite Rocket-Meteorite Collision

For the purpose of investigating the probability of a satellite rocket-meteorite collision, the path followed by the satellite rocket may be considered to consist of two parts. The first portion of the path is the ascending trajectory from the surface of the earth to the point where the satellite rocket enters its orbit about the earth. The second portion of the path is the orbit itself. The duration of the ascending trajectory is sufficiently short, 15 minutes or less[3], to reduce the probability of a collision during this period to a negligible value. The satellite rocket, however, in order to accomplish its function must remain in its orbit about the earth for at least several days. Here the probability of a satellite rocket-meteorite collision may assume considerable importance. The probability which best represents the critical condition for the case of a satellite rocket-meteorite collision is that which gives the chance of the event occurring *at least once* in the given time interval. Furthermore, it is of greater interest to know the probability of a collision with a meteorite of a given size *or larger* than to restrict the event to a collision with a meteorite of a single given magnitude.

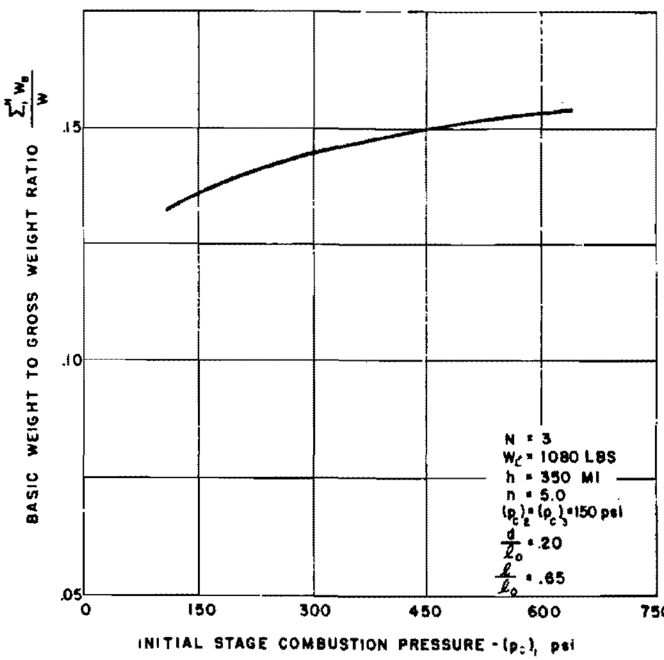
The probability p_r that an event will occur exactly r times in n independent trials is expressed by the Binomial Law[5]

$$p_r = \frac{n!}{r!\,(n-r)!}\,(p)^r\,(1-p)^{n-r}, \tag{23}$$



GROSS WEIGHT VS p_{c_1}

FIG. 19



BASIC WEIGHT TO GROSS WEIGHT RATIO VS COMBUSTION PRESSURE

FIG. 20

where p is the probability of the occurrence of the event in a single trial. This expression, while exact, is cumbersome to use since, in the cases under consideration, it involves the factorials of very large numbers.

When r is small compared to n, a good approximation to the Binomial Law is given by the Poisson formula[5]

$$p_r = \frac{(np)^r}{r!} e^{-np} , \tag{24}$$

where p_r is the probability that the event will occur exactly r times in n independent trials, and p is the probability of the event occurring in a single trial.

From Eq. (24) the probability p_0 that the event will fail to occur ($r = 0$) is

$$p_0 = e^{-np} . \tag{25}$$

Therefore, letting $p_{\geq 1}$ denote the probability of the event occurring at least once we have

$$p_{\geq 1} = 1 - p_0, \text{ or}$$

$$p_{\geq 1} = 1 - e^{-np} . \tag{26}$$

It is assumed that the meteorites entering the earth's atmosphere will have a random distribution over the surface of the earth's atmospheric shell and also a random distribution as regards their occurrence with time. It is also assumed that the meteorites travel through the atmosphere along the earth's radius vector and that the satellite rocket planform area is normal to the earth's radius vector.

If S_M represents the total number of meteorites from visual magnitude -3 up to and including magnitude M entering the earth's atmosphere in a 24 hour period, A_b represents the planform area of the satellite rocket payload compartment, A_e represents the surface area of the earth's atmospheric shell at a given altitude and T represents the time interval in hours for which the probability is desired, then (ref. 1)

$$np = \frac{S_M A_b}{24 A_e} T . \tag{27}$$

Therefore from Eq. (25)

$$p_{>1} = 1 - e^{-\frac{S_M A_b}{24 A_e} T} . \tag{28}$$

It should be noted that this probability does not exclude the possibility of the satellite rocket-meteorite collision occurring more than once in the given time interval T. in fact it specifically allows for the occurrence of more than one collision. Further, in interpreting the results of the application of Eq. (28), a probability of 1 means that the event is certain to occur, while a probability of 0 means that the event is certain not to occur. Table 3 gives the probability of a collision between the satellite rocket and a meteorite of a given magnitude or larger occurring at least once for various time intervals for a typical satellite rocket traversing an orbit 300 miles above the surface of the earth. The values used for S_M are given in Table 4 which is based essentially on observed meteors at heights of the order of 100 miles. Although it is recognized that these numbers would be slightly larger at higher altitudes (about 10% at 300 miles), they are undoubtedly in error by an amount greater than this, and therefore have been used as they stand.

Table 3

PROBABILITY OF AT LEAST ONE COLLISION BETWEEN THE SATELLITE ROCKET PAYLOAD COMPARTMENT AND A METEORITE OF MAGNITUDE M OR LARGER IN THE TIME T AT 300 MILES ALTITUDE

Visual Magnitude	Number of Meteorites	Probability of at least one collision			
		T = 120 Hrs	T = 240 Hrs	T = 480 Hrs	T = 720 Hrs
M	S_M	$P_{\geq 1}$	$P_{\geq 1}$	$P_{\geq 1}$	$P_{\geq 1}$
5	7.47×10^{7}		1.0×10^{-6}	2.0×10^{-6}	4.0×10^{-6}
6	1.88×10^{8}	2.0×10^{-6}	3.0×10^{-6}	7.0×10^{-6}	1.1×10^{-5}
7	4.72×10^{8}	4.0×10^{-6}	9.0×10^{-6}	1.8×10^{-5}	2.7×10^{-5}
8	1.18×10^{9}	1.1×10^{-5}	2.2×10^{-5}	4.5×10^{-5}	6.7×10^{-5}
9	2.98×10^{9}	2.8×10^{-5}	5.6×10^{-5}	1.0×10^{-4}	2.0×10^{-4}
10	7.47×10^{9}	7.1×10^{-5}	1.0×10^{-4}	3.0×10^{-3}	4.0×10^{-4}
12	4.72×10^{10}	4.5×10^{-4}	1.0×10^{-3}	1.798×10^{-3}	2.697×10^{-3}
15	7.47×10^{11}	7.075×10^{-3}	1.41×10^{-2}	2.7902×10^{-2}	4.161×10^{-2}
20	7.47×10^{13}	5.0747×10^{-1}	7.5739×10^{-1}	9.50213×10^{-1}	9.81684×10^{-1}
25	7.47×10^{15}	9.99999×10^{-1}	9.99999×10^{-1}	9.99999×10^{-1}	9.99999×10^{-1}

$$P_{\geq 1} = 1 - e^{-\frac{S_M A_b}{24 A_e} T}$$

Where:

$P_{\geq 1}$ = Probability of at least one collision in the time T.

S_M = Total number of meteorites from visual magnitude - 3 up to and including magnitude M entering the earth's atmosphere in a 24 hour period.

A_b = Planform area of the satellite rocket payload compartment = 11.5 square feet.

A_e = Surface area of earth's atmospheric shell at 300 miles = 60.6×10^{14} square feet.

T = Time interval in hours.

b. Size and Velocities of Meteorites

The visual magnitude of a meteorite is given in terms of a scale where numerically large magnitudes represent faint bodies. Furthermore, two meteorites which differ by five magnitudes have a hundred-fold difference in brightness and, since the brightness is directly proportional to the mass, have a hundred-fold difference in mass. The number, size and mass of meteorites entering the earth's atmosphere each 24 hours is given in Table 4 which is from a table given by Watson, ref. 6, page 115.

Leonard[7] gives the speed of a meteorite encountering the earth head-on as 47.4 miles/second, while a meteorite overtaking the earth will catch up at a speed of 8.2 miles/second. The amount the meteorite will decelerate in traversing the earth's atmosphere from its outer limit to an altitude at which it is likely to encounter a satellite rocket has been shown to be negligible. Watson, ref. 6, page 104, states that this deceleration is small and in general a bit less than the amount by which the earth's attraction has speeded up the meteorite. Furthermore, Whipple[8] states that at all magnitudes brighter than the 20th the deceleration above 100 miles is negligible and should by all means be neglected in calculating the effects of meteorites upon a satellite rocket.

Table 4

THE NUMBER, MASS, AND SIZE OF METEORITES ENTERING THE ATMOSPHERE EACH DAY
(Based on Watson)

Visual Magnitude	Observed Number	True Number N	Mass Grams	Weight lbs w	Diameter of Equivalent Sphere, ft* d
-3	28,000	28,000	4.0	8.72×10^{-3}	$.427 \times 10^{-1}$
-2	71,000	71,000	1.6	3.53×10^{-3}	$.317 \times 10^{-1}$
-1	180,000	180,000	.630	1.39×10^{-3}	$.232 \times 10^{-1}$
0	450,000	450,000	.250	5.51×10^{-4}	$.1705 \times 10^{-1}$
1	1,100,000	1,100,000	.100	2.20×10^{-4}	1.257×10^{-2}
2	2,800,000	2,800,000	.040	8.72×10^{-5}	$.922 \times 10^{-2}$
3	6,400,000	7,100,000	.016	3.53×10^{-5}	$.683 \times 10^{-2}$
4	9,000,000	18,000,000	.0063	1.39×10^{-5}	$.500 \times 10^{-2}$
5	3,600,000	45,000,000	.0025	5.51×10^{-6}	$.367 \times 10^{-2}$
6		110×10^{6}	.0010	2.20×10^{-6}	$.2705 \times 10^{-2}$
7		280×10^{6}	.00040	8.72×10^{-7}	$.1986 \times 10^{-2}$
8		710×10^{6}	.00016	3.53×10^{-7}	1.471×10^{-3}
9		18×10^{8}	.000063	1.39×10^{-7}	1.078×10^{-3}
10		45×10^{8}	.000025	5.51×10^{-8}	$.793 \times 10^{-3}$
15		45×10^{10}	2.5×10^{-7}	5.51×10^{-10}	1.705×10^{-4}
20		45×10^{12}	2.5×10^{-9}	5.51×10^{-12}	$.367 \times 10^{-4}$
25		45×10^{14}	2.5×10^{-11}	5.51×10^{-14}	$.793 \times 10^{-5}$
30		45×10^{16}	2.5×10^{-13}	5.51×10^{-16}	1.705×10^{-6}

lbs = grams $\times 2.205 \times 10^{-3}$

*Based on a specific gravity of 3.4.

c. Penetration of Metal Plates by High Speed Particles

Assuming normal impact, a high speed particle striking a solid medium will either be stopped by the wall or pass through the wall and emerge with a reduced velocity due to the resistance offered by the wall to the penetration of the particle. Two hypotheses have been presented for the penetration of metal plates by very small but high speed particles. The first hypothesis, that given by Grimminger[1], assumes that the particle will penetrate the plate as though the plate were perfectly deformable like a fluid, and will be referred to in this report as the non-shattering impact method. The second hypothesis, that given by Whipple[8], assumes that the shock wave set up by a fast moving particle entering a solid medium may be represented by a right circular cone having a total apex angle of 60°, and that the total energy of the particle is used either to fuse or to vaporize the material included in this cone. This method of approach to the problem will be referred to in this report as the shattering impact method.

The penetration equation as given by Grimminger for non-shattering impact applies only to aluminum, and is based on a drag coefficient for the particle of 2/3. It was later found that the drag coefficient should more properly be 1.0[9]. A correction to Grimminger's equation for a drag coefficient of 1 and for an 18-8 type stainless steel yields

$$\frac{T}{d} = .578 + .646 \lg_e \frac{V_m}{1.52} \quad , \tag{29}$$

where T represents the penetration distance in centimeters, d represents the diameter of the particle in centimeters and, V_m represents the velocity of the particle before impact in kilometers per second.

The critical condition is represented by the case where the meteorite has its maximum velocity with respect to the satellite rocket, that is, $V_m = 76.3$ km/sec (47.4 miles/sec). Therefore for the critical condition the expression for non-shattering impact reduces to

$$T = 4.60\, d \; . \tag{30}$$

The penetration formula as given by Whipple[8] for the case of shattering impact is

$$T = \left(\frac{12}{\pi \rho \zeta}\right)^{1/3} E_m^{\;1/3} \quad , \tag{31}$$

where T is the penetration distance in centimeters, ρ is the plate density in grams/cubic centimeter, ζ is the heat required to melt or vaporize the plate material in ergs per gram, and E_m is the total kinetic energy of the meteorite in ergs. For an 18-8 type stainless steel plate Eq. (31) reduces, for melting, to

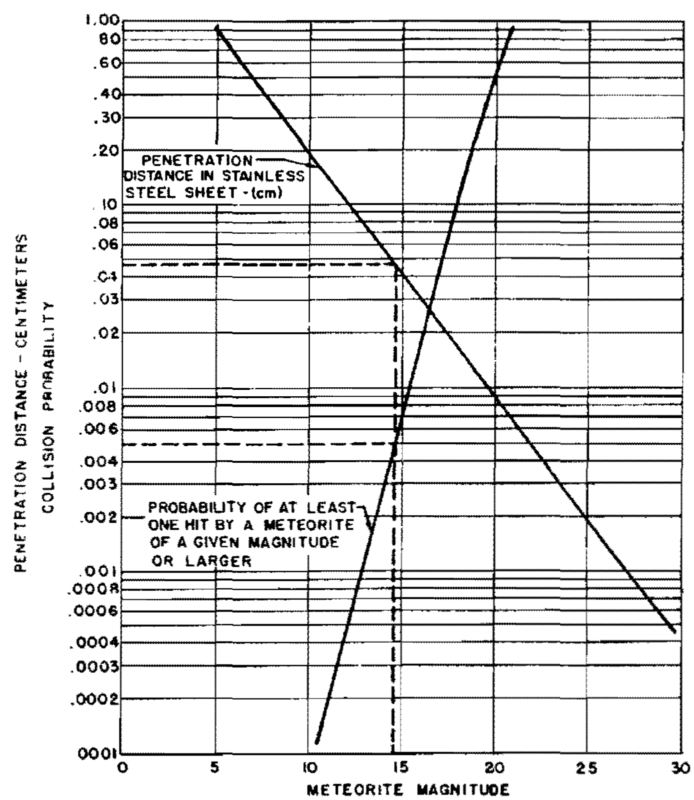
$$T = 3.43 \times 10^{-3} E_m^{\;1/3}, \tag{32}$$

and, for vaporizing, to

$$T = 1.80 \times 10^{-3} E_m^{1/3}. \quad (33)$$

Table 5 gives the penetration distance of meteorites from magnitude -3 to magnitude 30 in an 18-8 type stainless steel plate according to both the shattering impact and the non-shattering impact hypotheses.

It is felt that a plate of sufficient thickness to give 1 chance in 200 of a penetration in 5 days will adequately protect the contents of the satellite rocket payload compartment against possible damage due to meteorite collision. Fig. 21 gives the variation of the probability of at least one satellite rocket meteorite collision with a meteorite of a given magnitude or larger as well as the variation of the 'standard' penetration in centimeters of a meteorite striking a stainless steel plate, as a function of meteorite magnitude. The 'standard' penetration curve represents the mean penetration as given by the shattering impact, and the non-shattering impact methods for the critical conditions. From an examination of Fig. 21 it is apparent that the previously selected minimum gage of .020 inches gives adequate protection against possible damage to the contents of the payload compartment due to satellite rocket-meteorite collisions.



SATELLITE ROCKET—METEORITE COLLISION PROBABILITY AND METEORITE PENETRATION VS METEORITE MAGNITUDE

FIG. 21

Table 5

PENETRATION DISTANCE IN 18-8 TYPE STAINLESS STEEL SHEET BY A METEORITE OF A GIVEN MAGNITUDE

Visual Magnitude	Meteorite Mass	Meteorite Dia	Meteorite Velocity	Meteorite Kinetic Energy	Penetration Distance		
					Shattering Impact		Non-Shattering Impact
					Melting	Vaporizing	
M	m	d_m	V_m	E_m	T	T	T
	Grams	cm	km/sec	ergs	cm	cm	cm
-3	4.0	1.31	76.3	11.644×10^{13}	16.60	8.788	6.026
0	2.5×10^{-1}	5.18×10^{-1}	76.3	7.2775×10^{12}	6.589	3.488	2.383
5	2.5×10^{-3}	1.13×10^{-1}	76.3	7.2775×10^{10}	1.418	7.509×10^{-1}	5.198×10^{-1}
10	2.5×10^{-5}	2.40×10^{-2}	76.3	7.2775×10^{8}	3.058×10^{-1}	1.619×10^{-1}	1.104×10^{-1}
15	2.5×10^{-7}	5.18×10^{-3}	76.3	7.2775×10^{6}	6.589×10^{-2}	3.488×10^{-2}	2.383×10^{-1}
20	2.5×10^{-9}	1.13×10^{-3}	76.3	7.2775×10^{4}	1.418×10^{-2}	7.515×10^{-3}	5.198×10^{-3}
25	2.5×10^{-11}	2.40×10^{-4}	(76.3)	7.2775×10^{2}	3.058×10^{-3}	1.619×10^{-3}	1.104×10^{-3}
30	2.5×10^{-13}	5.18×10^{-5}	(76.3)	7.2775	6.589×10^{-4}	3.488×10^{-4}	2.383×10^{-4}

9. Propellant Distribution System

It has been frequently assumed that a gas pressurized propellant distribution system is the most economical from a weight standpoint for small rockets using liquid propellants, while for larger rockets a turbine-pump arrangement is the best. It was originally assumed that the satellite rocket would fall in the latter class. This assumption was later verified[10] by a study of the comparative weights of the two systems for a typical three stage satellite rocket. Furthermore, it is also shown in this study that a fully pump fed system would weigh less than a partially gas fed system.

Table 6

PROPELLANT DISTRIBUTION SYSTEM WEIGHT

Stage	Gross Wt	Propellant Distribution System Wt		Ratio $\frac{\text{Pump Fed}}{\text{Pressurized}}$
		Pressurized	Pump Fed	
III	4615	171	150	.875
II	18,630	682	370	.54
I	86,420	6910	1831	.265

From Table 6, which gives the results of this study, it is clearly indicated that as the gross weight increases the weight advantage of the pump fed propellant delivery system as compared to the gas pressurized system becomes increasingly significant. Also, that even for the smallest stage of the satellite rocket the use of a pump fed propellant delivery system in preference to the gas fed system results in a 12.5% saving in the weight of this item.

10. Body Shell Material

A study of the skin temperature likely to be realized during the ascending trajectory of the satellite rocket[4] indicates that a peak temperature of about 1500°F will be realized.

Since this temperature is well above the melting point of aluminum and its alloys, the use of such materials for the outer shell of the satellite rocket is eliminated.

Fortunately the duration of temperatures of this order will be very short; moreover the period during which thrust load is applied to the structure will be approximately 15 minutes or less.

In view of these conditions a stabilized 18-8 type stainless steel sheet in the 1/2 hard condition has been selected as a possible material for the shell of the satellite rocket. This selection was made since the mechanical properties of this austenitic steel will not vary significantly with the values of temperature and time interval at that temperature which may be realized in the satellite rocket.

Data on the short-time, high-temperature mechanical properties of sheet materials which may be suitable for the satellite rocket shell are particularly meager and further investigations in this field will perhaps disclose other materials better adapted to the particular conditions for which the satellite rocket shell must be designed.

11. Primary Structure

Two fundamentally different configurations have been investigated as possible structural types which may be employed in the design of the satellite rocket. The first type consists of a thin unstiffened shell and employs the use of internal pressure to maintain structural stability. The second type consists of a grid framework covered by a thin skin, designed in accordance with present airframe practice.

For the purpose of evaluating the relative merits of the two types of construction, studies were undertaken to develop two satellite rockets which differed only in the type of primary structure used in the initial stage. The work on the pressurized satellite rocket was done at the Contractor's El Segundo plant and is included as Appendix III of this report.

The weight of the equivalent 'conventional' satellite rocket was estimated from methods similar to those developed in Appendix I.

Table 7

STAGE I WEIGHT BREAKDOWN OF A 'PRESSURIZED' AND A 'CONVENTIONAL' SATELLITE ROCKET

	Items	'Pressurized'	'Conventional'
W_T	Primary Structure	1015 lbs	2350 lbs
W_M	Miscellaneous Structure	720	865
W_N	Thrust Producing Equipment	1263	1500
W_C	Surfaces & Controls	1016	1260
W_B	Basic Weight	4014	5975
W_P*	Propellants & Fuels	37,435	45,000
W_L*	Payload	6540	6540
W	Gross Weight	47,989	57,515
W_B/W	Basic weight to gross weight ratio	.0836	.104

As may be seen from Table 7, which presents the results of these two studies, the use of the 'pressurized' type structure in this case would seem to result in a 20% reduction in the basic weight to gross weight ratio for the initial stage of a two stage satellite rocket. The 'pressurized' study does not include, however, any allowance for the equipment necessary to maintain the required pressure in the satellite rocket, or for the structure necessary to attach the stabilizing fins to the satellite rocket body, while these items or their equivalent are included in the 'conventional' study. In view of these facts, the 'conventional' primary structure has been arbitrarily selected for the proposed satellite rocket in order to be conservative. Further investigation will be necessary, however, before a final choice can be made.

III. STRUCTURAL DESIGN STUDIES

In order to integrate the results of the many investigations which have been made in the various fields of satellite rocket performance and design, a design study of a typical satellite rocket has been undertaken. Furthermore, this study provides a means for the consideration in greater detail of many of the manifold 'practical' problems encountered in the design of a satellite rocket which have up to this point been given only a general treatment, and also will serve as a verification of the methods used to estimate the gross weight of the satellite rocket, and as a check on some of the assumptions made in the development of this method. The design of the satellite rocket as presented at this time represents only the skeleton of a workable device as it exists at this early stage in its development and should by no means be taken to represent the best possible configuration for the chosen conditions.

1. General Design Conditions

This design study is for a satellite rocket employing the optimum values of the various design parameters as found in previous sections of this report and is summarized briefly below.

The satellite rocket will utilize a hydrazine-liquid oxygen propellant system in all stages, will have an orbital altitude of 350 miles, will have a primary structure designed in accordance with present airframe practice, will employ a pump fed propellant delivery system in all stages, and the propellants will be consumed at a constant mass rate in all stages.

The payload has been arbitrarily established at 500 pounds. This item, along with an allowance of 180 pounds for auto pilots and 400 pounds for the orbital electric power plant places the weight of fixed equipment, $W_L{}'$, in the final stage at 1080 pounds.

The number of stages for the above conditions which will give a satellite rocket of the least gross weight is found from Fig. 2 to be four. However, since there is only a 2% increase in gross weight by using a three stage satellite rocket, this number of stages has been selected as representing the optimum condition since the gain in reliability of operation of a three stage satellite rocket over a four stage rocket should more than offset the small increase in gross weight.

The optimum value of the maximum axial load factor for the proposed satellite rocket is indicated by Fig. 7 to be about 4.5. This information was not available at the time the selection was made and a value of $n = 5.0$ has been used for the design study. It is felt, however, that this will not appreciably affect the results of this study.

Fig. 17 and Fig. 19 show that the combustion pressure employed in the latter two stages should be relatively low, while the combustion pressure used in the initial stage should be comparatively high. A combustion pressure of 150 psi has been selected for the third stage. Due primarily to space limitations imposed on the rocket motors, a combustion pressure of 300 psi has been selected for the second stage. Fig. 19 indicates that the combustion pressure of the initial stage should be about 600 psi; however, this information was not available at the time the selection was made, and a value of 400 psi was chosen for the first stage.

The ratio of maximum body diameter to overall length d/ℓ_0 of about .20 has been selected as Fig. 10 shows this to be the optimum condition, while the ratio of the distance from the nose to the point of maximum diameter to the overall length ℓ/ℓ_0 has been established as .65 on the basis of the earlier study to maintain a reasonable angle of convergency for the satellite rocket body. Later studies on the satellite rocket body shape have indicated that a higher value of the ratio ℓ/ℓ_0 may yield a lower gross weight, however, this information was not available at the time the selection was made.

2. General Arrangement of the Satellite Rocket

The external configuration of the satellite rocket is shown in Fig. 22, and the general arrangement is given in Drawing No. RRL-4 which is included as Appendix V of this report. The shape is that of a typical projectile having a pointed nose and a contoured shell which increases to a maximum diameter at 65% of the overall length and then decreases to a diameter at the base which is compatible with the space requirements of the rocket motors of stage 1. Four stabilizing fins are attached to the after body in a cruciate array. The overall length of the satellite rocket body is 565 inches while its maximum diameter is 123 inches. To facilitate assembly in the field, transportation from the factory to the launching site, and handling in the factory, the satellite rocket is made up of a number of subassemblies as illustrated in Fig. 23. These subassemblies are a payload compartment, a tank compartment for each stage, a rocket motor support structure for each stage, and the initial stage stabilizing group.

a. Payload Compartment

The payload compartment constitutes the nose section of the satellite rocket and is in the shape of a right circular cone. It has a volume of 35 cubic feet and houses 500 pounds of instrumentation and communication equipment W_L and 180 pounds of control equipment, that is, automatic pilots W_I. The skin of the payload compartment is made of .020 inch thick stabilized 18-8 type stainless steel sheet to protect the contents against possible damage due to meteorite collision.

b. Tank Compartment

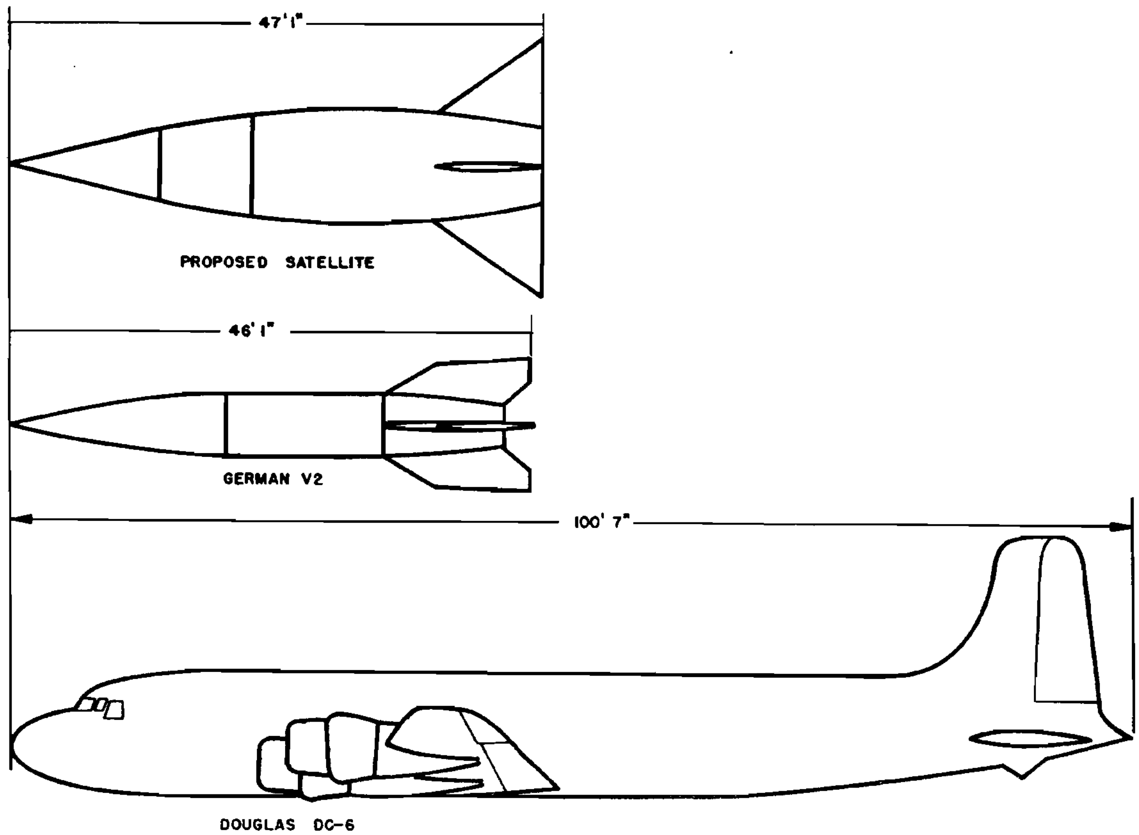
The tank compartment for a given stage houses the propellant tanks for that stage and has the external shape of a truncated right circular cone. Furthermore, the mechanism required for the separation of stages during flight is located in the forward portion of the tank compartment.

The external shell of the tank compartments for the various stages forms the principal load carrying structure for that portion of the satellite rocket, and is made of a stabilized 18-8 stainless steel skin over a grid of longitudinal and transverse stiffeners of the same material.

The hydrazine (anhydrous) tank is made of a suitable aluminum alloy, while the liquid oxygen tank is made of stainless steel.

c. Rocket Motor Support Structure

The supporting structure for the rocket motors and associated equipment in each stage is made of a tubular steel framework terminating at the forward end in a thrust ring which attaches to the aft end of the given stages' tank com-



SIZE COMPARISON

FIG 22

partment. The tubular framework carries the load due to the rocket motor thrust for that stage, while the thrust ring distributes this load uniformly to the shell of the tank compartment to which it attaches.

d. **Stabilizing Group**

The initial stage stabilizing group is made up of a shell structure in the form of a truncated cone, and four stabilizing fins which attach to the after portion of this shell structure. The shell structure attaches at its forward end to the aft end of the first stage tank compartment, forms a covering for the first stage rocket motor support structure, and has a diameter at the aft end which is compatible with the space requirements of the rocket motors. Furthermore, this shell supports the weight of the entire satellite rocket when it is erected on the launching platform.

f. **Propellant Delivery System**

The propellant delivery system for each stage is similar to that of the A-4 and consists of dual turbine driven pumps which deliver the fuel and the oxidizer from their respective tanks to the rocket motor through the necessary control and regulating valves. The turbine is driven by hot gases which are generated from the decomposition of hydrogen peroxide under the action of a suitable catalyst. The hydrogen peroxide is delivered to its reaction chamber from the storage tank by helium gas under high pressure. The plumbing connecting the various units is provided with expansion joints at suitable points to prevent breakage of the lines.

g. Rocket Motors

The rocket motors[10] conform generally to present conventional design practice for regeneratively cooled liquid propellant motors, except that there is a sharp change in contour between the convergent and divergent sections of the nozzle in contrast to the usual smooth transition.

h. Controls and Stabilizing Surfaces

Suitable devices are provided to adequately control the attitude and thrust of each stage of the satellite rocket.

Thrust control is provided by establishing a constant rate of propellant flow to the rocket motors. Attitude control on the ascending trajectory is provided by four movable control motors mounted symmetrically about the central fixed rocket motor. By means of this device an improvement in rocket motor performance is realized over the case where jet vanes are used. This increased performance is primarily due to the absence of jet vane drag, and further, the erosion problems encountered when jet vanes are used are avoided. Yaw and pitch control are achieved by a symmetrical deflection of the proper pair of control motors, while roll control is obtained by the differential deflection of the control motors in pairs.

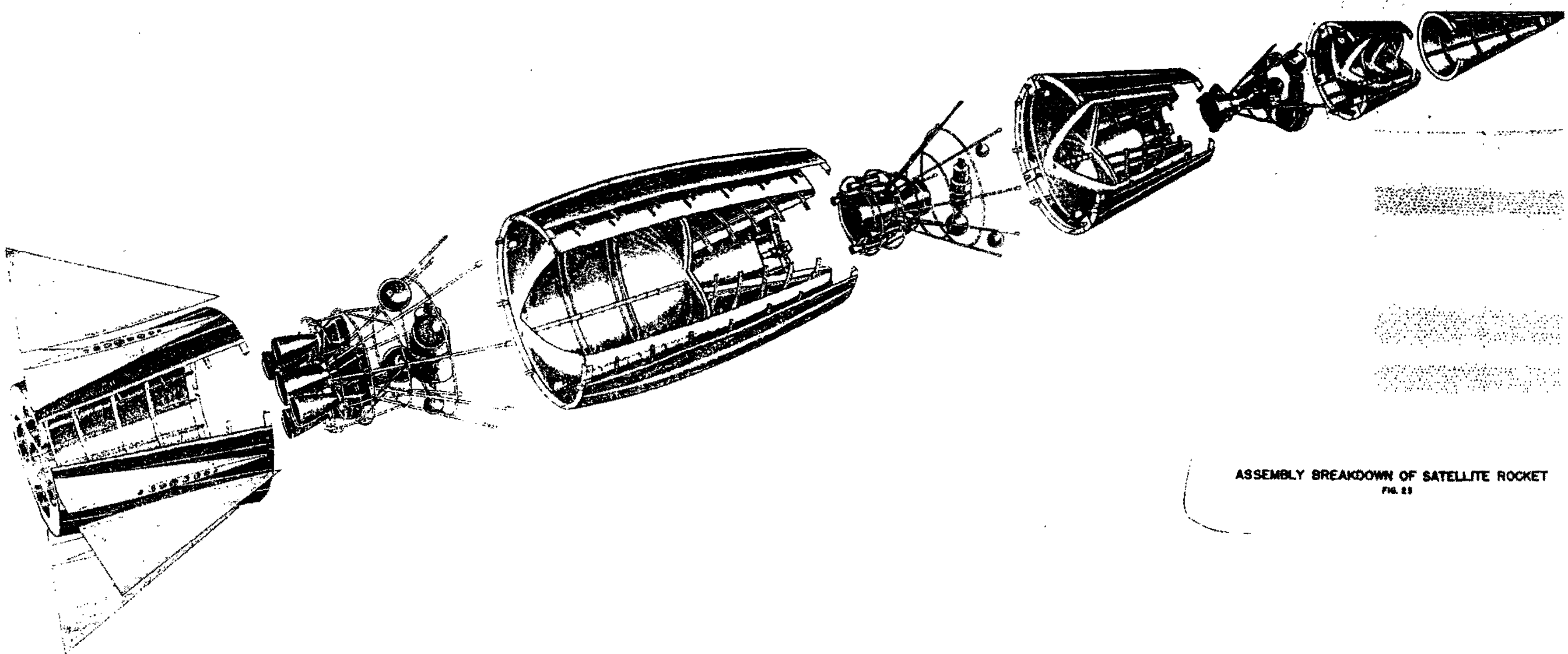
Orbital attitude control is achieved by means of three flywheels mounted on mutually perpendicular axes. Through changing the angular momentum of these flywheels in the proper combination the desired attitude may be maintained.

The control devices in all stages and the orbital controls are directed by a single regulating unit located in the payload compartment. This unit consists of essentially two parts: first, an automatic pilot providing trajectory control intelligence and the second, an automatic pilot providing orbital control intelligence.

The first stage stabilizing surfaces are mounted on the after portion of the satellite rocket body. They are arranged in the form of a cross and each has the shape of a 4% thick, modified double wedge delta wing having an aspect ratio of 2.31. These fins are completely fixed and have no movable surfaces, since all the required attitude control is provided by the control motors.

i. Auxiliary Power Sources

Two additional power sources are provided to furnish the electric power required to operate the communications equipment, instrumentation, regulating and control devices, and the servo-system. The first supplies the necessary power during the ascending trajectory and may be either a battery in each stage or a small generator driven by the propellant delivery turbine. The second is located in the final stage and supplies the power required during the orbital



ASSEMBLY BREAKDOWN OF SATELLITE ROCKET

FIG. 23

flight of the satellite rocket. This latter device is fully discussed in a separate report[10].

j. Stage Separation Device and Release Mechanism*

From the structure of the satellite rocket, a three stage vehicle with a considerable portion of each succeeding stage fitting inside the preceding stage, as shown in Fig. 23, it is apparent that the stage separation device must accomplish the following four requirements:

1. Orient the stages relative to each other with respect to pitch and yaw axes.

2. Absorb any tensile forces between stages imposed by aerodynamic or control forces manifested as a bending of the missile about its pitch or yaw axes.

3. Positive release at time required. A time lag in the operation of one release mechanism would cause the release to be eccentric and the succeeding stages to be tilted.

4. Cause stages to separate with a minimum of interference.

This can be accomplished by a device such as is illustrated in Fig. 25. Four of these mechanisms, each complete with helium tank, solenoid valve, and pressure regulating and reducing valve are used for each of the stage separations. Each set of four is equally spaced on the periphery of the stage separation plane and at angles of 45° to the yaw and pitch axes, as shown in Fig. 24.

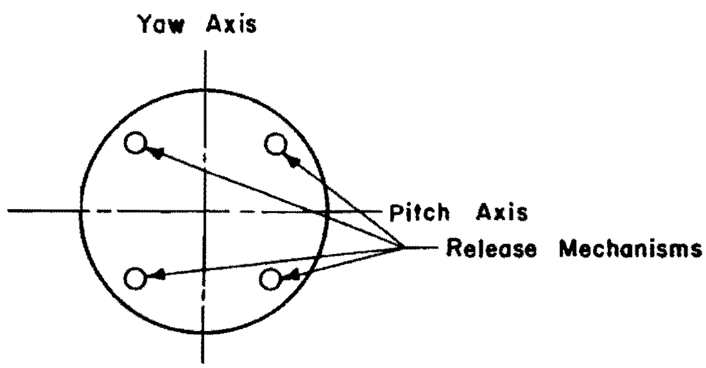
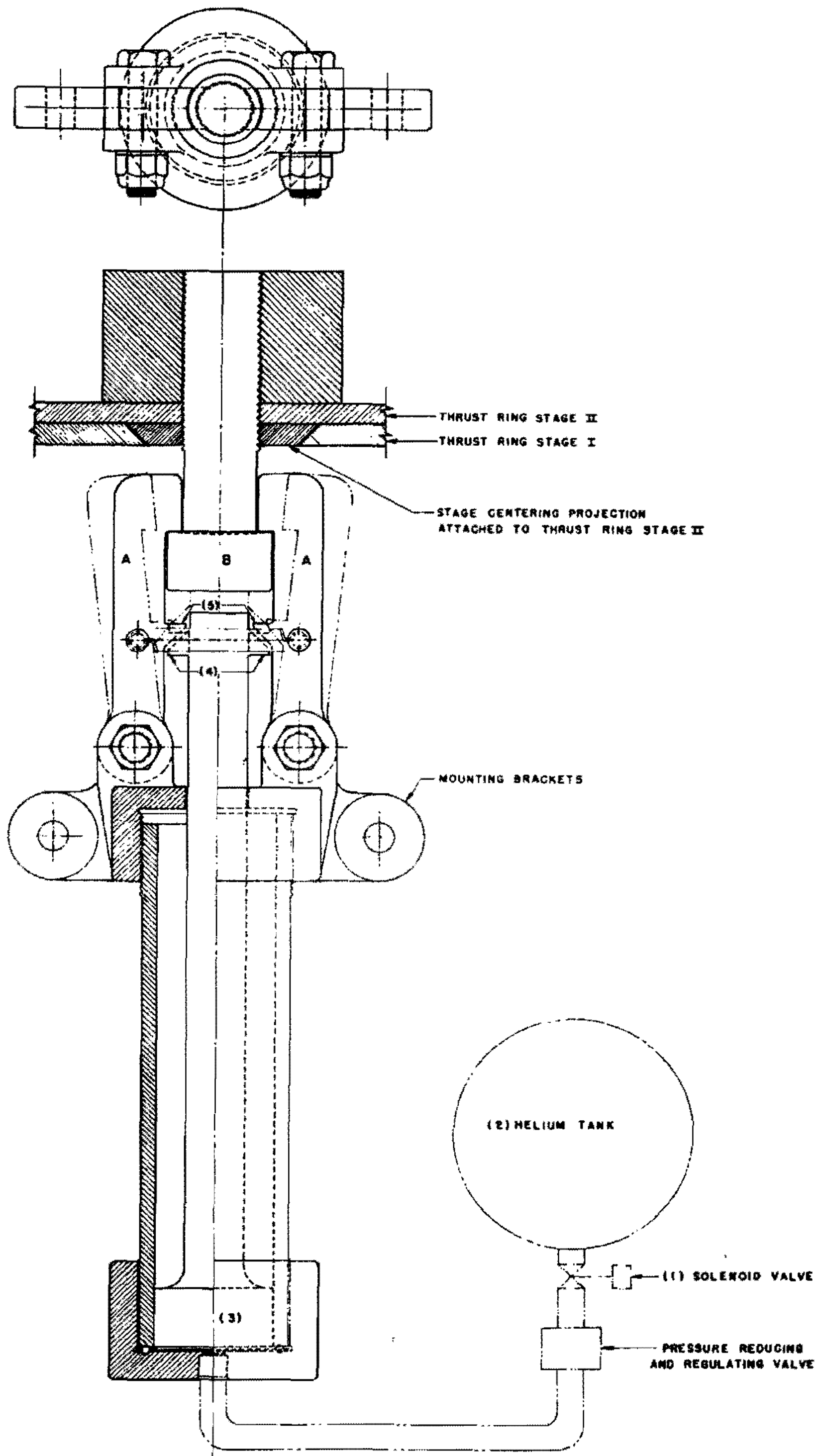


FIG. 24

As pointed out in requirement 3 above, to insure successful accomplishment of the stage separation, all of the actuating units must function simultaneously; or rather, if the operation of one or more of the units precedes or follows the operation of the remaining units the time lag involved must not exceed a certain permissible limit. Time has not allowed a complete consideration of this requirement in the device described below, which will, after further study, have to be modified to satisfy this condition.

* The following discussion of the stage separation device and release mechanism is due to J. O. Crum.



STAGE SEPARATION DEVICE

FIG. 25

The release mechanism itself is mounted in stage I for the separation of stage I and II, and in stage II for the separation of stages II and III. The connection between stages is solely by the action of the clamps A of the mechanism on the projecting lug B of the succeeding stage. In the no load condition, the clamps A are held in position by a spring.

The action of the mechanism is as follows (see Fig. 25). Consider only the separation of stages I and II and the action of one mechanism. The action of all of the other mechanisms is identical.

At the end of burning of stage I the solenoid (1) controlling the release of helium from the high pressure tank (2) is activated and thus admits helium to the cylinder of the release mechanism through a pressure reducing valve. As the high pressure gas must reach each of the four cylinders simultaneously to produce an axial separating force, the electric current supply for the four solenoids must be a common switch. As the piston (3) moves, the cams (4) on the piston rod engage the cams (5) on the clamps A to release the lugs B. After the accomplishment of this the piston travels further until the piston rod engages the lug (B) and pushes the two stages apart during the remainder of its stroke. This action imparts a relative velocity to the two stages sufficient to produce a separation of 30 feet in the allotted time without exceeding the allowable stresses in either stage.

Consider the separation of stages I and II, which is the first separation. If the satellite rocket can be made to fly at zero angle of attack relative to the air stream at the time of stage separation there will be no aerodynamic moment at separation to tilt stage II, although this is a condition of unstable equilibrium. Stage I is itself air stable. This separation occurs at an altitude of approximately 100,000 feet and at a velocity of approximately 6200 feet per second, hence the aerodynamic moment is a factor which must be considered as it has 1.22 seconds to act during which separation occurs and no control force is acting. This is particularly important as stage II is partially submerged in stage I. However, this effect becomes negligible at the separation of stages II and III which occurs at approximately 318,000 feet and 15,800 feet per second. This problem is discussed in a separate report[4].

The time necessary for the separation of stages is determined in the following manner.

Thrust of a lower stage just prior to stage separation is given by:

$$T = (m_1 + m_2)g \sin\theta + (m_1 + m_2)X g$$

$$= (m_1 + m_2)g\ (X + \sin\theta) \tag{34}$$

where

T = thrust

m_1 = mass of the lower stage empty

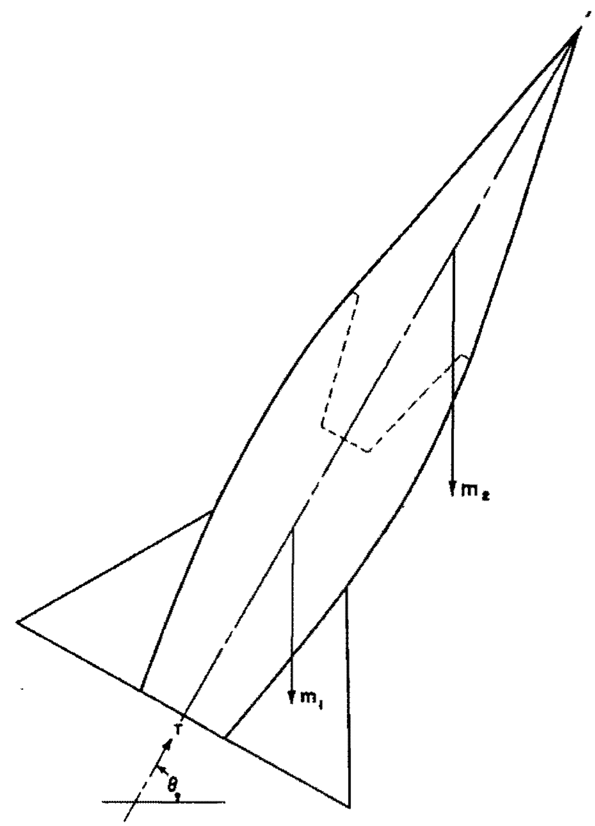


DIAGRAM - SATELLITE ROCKET LOADS AT SEPARATION

FIG. 26

m_2 = mass of higher stages, full

g = acceleration of gravity

X = acceleration of rocket along flight path

θ = angle of trajectory relative to the instantaneous horizontal,

but the thrust of the lower stage is capable of accelerating the satellite rocket in this condition at a maximum of $5g$.

Therefore

$$5 = X + \sin\theta \ . \tag{35}$$

For this reason the force which can be applied to the lower stage during separation without causing structural failure is

$$F = m_2 \, g(\sin\theta + X) \ , \tag{36}$$

where

$$F = \text{force exerted by stage separation mechanism.}$$

This force will by definition produce a 5g acceleration of the higher stage.

The reaction will produce on the lower stage an acceleration $\underline{a}$ g along the trajectory.

$$F + m_1 \sin\theta\, g = m_1\, a_1$$

$$a_1 = \frac{F}{m_1} + \sin\theta\, g\ . \qquad (37)$$

The relative acceleration between the lower stage and the higher stage will be

$$(a_1 + X)g = (\frac{F}{m_1} + \sin\theta\, g) + (\frac{F}{m_2} - \sin\theta\, g) = F(\frac{m_1 + m_2}{m_1\, m_2})\ . \qquad (38)$$

Due to the design of the separation mechanism this acceleration will act only during the first 8″ of travel.

The relative velocity between the lower stage and the higher stage at the end of acceleration is

$$v = \sqrt{2(a_1 + X)g\, S}\ , \qquad (39)$$

where

$$S = 8 \text{ inches.}$$

It has been arbitrarily assumed that the stages will be separated by 30 feet prior to the operation of the rocket motors in the upper stage. It is believed that at this distance an explosion of the fuel tanks of the lower stage due to the hot exhaust gases of the higher stage will not appreciably affect the motion of the higher stage.

The time for the separation d = 30 ft to occur with the stages moving at the above relative velocity is

$$d = v\, t$$

$$t = \frac{d}{v}. \qquad (40)$$

This is the time interval between the stopping of the thrust of the lower stage and the starting of the thrust of the higher.

For the separation of stages I and II this time is 1.22 seconds.

For the separation of stages II and III this time is 1.06 seconds.

3. Erection Procedure

The rocket may be shipped in the form of major subassemblies, such as tank section and power plant section for each stage. Final assembly can be made at the launching site, beginning with the stage 1 motor section and proceeding with erection in a vertical position.

4. Structural Design Criteria

Some of the principal loads to which the primary structure of the satellite rocket is subjected may be classified as an axial compression due to the rocket motor thrust, a pressure load on the satellite rocket shell due to the rocket's motion through the atmosphere, a bending load due to a stable pitched attitude (tilt) during the operation of the first stage, a bending load due to the curvature of the rocket's trajectory, a bending load due to instability in pitch or yaw or both, and a torsional load due to roll instability. Time has not allowed a complete study of all of the loading conditions to which a satellite rocket may be subjected, but the following may be said in a qualitative manner about those listed above.

The axial compression due to the rocket's thrust will be the principal design criteria for most of the satellite rocket's shell structure. The most notable exception to this is the shell of the first stage stabilizing group which must be designed to carry the dead weight of the rocket in its static launching attitude, and to carry the shear and bending due to the action of the stabilizing fins during flight. The dynamic pressure load on the rocket's shell will have, in general, a secondary effect when compared to the axial compression. The effect of those loads which cause bending of the rocket is also generally small when compared to the axial compression. Of these bending loads, that due to the programmed tilt of the rocket with respect to its line of motion during the operation of the initial stage appears to be the most important; while calculations of the angular accelerations which must be given to the rocket in order to keep it on its curved trajectory in stable motion indicate that these bending loads are of the least importance. The bending and torsion due to instability in pitch, yaw, and roll has not been accurately determined, but these loads appear to be secondary to that due to tilt.

Table 8 gives the structural design criteria used for the design study, and a preliminary structural analysis of the proposed satellite rocket is given in Appendix IV.

Drawing No. RRL-4, Appendix V of this report, presents a preliminary design of the satellite rocket as it exists at this stage of its development.

5. Comparative Weight Summary

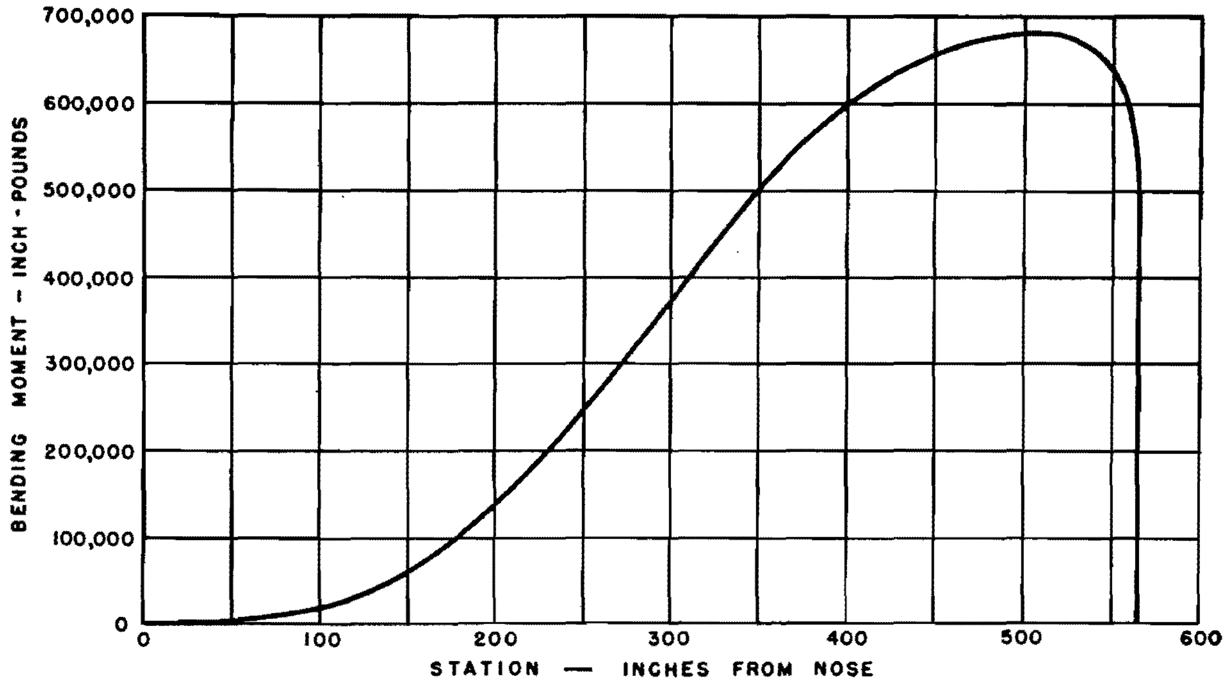
A comparative summary of the weight of the satellite rocket as estimated from the methods developed in Appendix I and as calculated from a consideration of the preliminary design study is given in Table 9. From an examination of this table it may be seen that the calculated gross weight of 86,463 pounds exceeds the estimated gross weight of 86,420 pounds by less than one per cent. It is felt that this close agreement verifies the general validity of the method developed to estimate the gross weight of the satellite rocket.

Due to the fact that the optimum value of a few of the design parameters had not been completely isolated at the time the design study was undertaken, reasonable

Table 8

STRUCTURAL DESIGN CRITERIA
for
Three Stage Satellite Rocket, Hydrazine-Liquid Oxygen Propellant System, Orbit Altitude of 350 Miles, Propellant to Gross Weight Ratio of .6532 Per Stage, and a Maximum Axial Load Factor of 5.0

Item	Design Load Factor n		Propellant Tanks
	Compression	Tension	Full or Empty
A. Axial Loads			
1. Shell Structure			
a. Payload Compartment	$(n_f)_1 = 5.00$	--	
b. Tank Compartment			
1. Stage 3	$(n_f)_1 = 5.00$	--	Full
2. Stage 2	$(n_f)_1 = 5.00$	--	Full
3. Stage 1	$(n_i)_1 = 1.44$	--	Full
c. Stabilizing Group	$(n)_0 = 1.00$	$(n_f)_1 = 5.00$	
2. Motor Support Structure			
a. Stage 3	$(n_i)_3 = 1.72$	$(n_f)_1 = 5.00$	Full
b. Stage 2	$(n_i)_2 = 1.72$	$(n_f)_1 = 5.00$	Full
c. Stage 1	$(n_i)_1 = 1.44$	$(n)_0 = 1.00$	Full
3. Propellant Tanks & Supports			
a. Stage 3	$(n_f)_1 = 5.00$		Full
b. Stage 2	$(n_f)_1 = 5.00$		Full
c. Stage 1	$(n_i)_1 = 1.44$		Full
4. Turbine Fuel Tanks & Supports			
a. Stage 3	$(n_f)_1 = 5.00$		--
b. Stage 2	$(n_f)_1 = 5.00$		--
c. Stage 1	$(n_i)_1 = 1.44$		--
5. Helium Tanks & Supports			
a. Stage 3	$(n_f)_1 = 5.00$		--
b. Stage 2	$(n_f)_1 = 5.00$		--
c. Stage 1	$(n_f)_1 = 5.00$		--
	Pressure		Application
B. Pressure Loads			
1. Shell Structure	.868 psi		External
2. Propellant Tanks			
a. Oxygen			
1. Stage 3	22.5 psi		Internal
2. Stage 2	32.0		Internal
3. Stage 1	19.6		Internal
b. Hydrazine			
1. Stage 3	19.0 psi		Internal
2. Stage 2	25.8		Internal
3. Stage 1	25.1		Internal
3. Turbine Fuel Tanks			
a. Stage 3	400 psi		Internal
b. Stage 2	400		Internal
c. Stage 1	400		Internal
4. Helium Tanks			
a. Stage 3	3000 psi		Internal
b. Stage 2	3000		Internal
c. Stage 1	3000		Internal
C. Bending Loads	See Figure 27		
D. Torsional Loads	Not Included		



BENDING MOMENT DIAGRAM

FIG. 27

values had to be assumed for the purpose of this design study. It was found after the optimum values of these parameters had been determined that the assumed values corresponded very closely to the optimum values. It is felt then that the general validity of this design study is not affected by the use of these assumed values.

The estimated gross weight of the satellite rocket for the given conditions and utilizing the optimum values of the design parameters will be less than that found in the design study. The estimated gross weight of this three stage hydrazine-oxygen satellite rocket having a final stage payload of 1080 pounds and an orbital altitude of 350 miles is 84,400 pounds, a reduction of 2.4 per cent over the value given in the design study.

IV. CONCLUSIONS

It has been shown in the studies presented that to establish a man-made satellite weighing 1550 pounds on an orbit 350 miles above the surface of the earth will require a staged rocket utilizing a hydrazine-oxygen propellant system and weighing only 85,000 pounds.

The propellant system hydrazine-oxygen was selected since it gave a reasonable gross weight and did not present the severe physical difficulties inherent in the higher performance systems, hydrogen-oxygen and hydrazine-fluorine. It was shown to be generally impractical to build a single stage satellite rocket utilizing any of the propellant systems investigated. The number of stages to give the least gross weight for the hydrazine-oxygen propellant system was shown to be four, however, a three stage rocket is proposed since the 2% increase in gross weight will be more than offset by the reduced complexity and the increased reliability of the three stage rocket as compared to the four stage rocket. A constant mass rate of propellant consumption was found to be the optimum method of burning the propellants in any stage since it results in the least applied structural load. The optimum value of the maximum applied load factor was found to be in the range of 4 to 5. Since this value is a transient condition and occurs only instantaneously, it places the satellite well within the realm of man-carrying devices.

The shape of the satellite rocket body was found to have considerable influence on the gross weight. For the three stage hydrazine-oxygen rocket, the optimum body shape is approximately that of a right circular cone increasing to a maximum diameter at 76% of the overall length and then decreasing to a diameter consistent with the requirements of the initial stage rocket motors. The optimum value of the maximum diameter was found to be about 20% of the overall length. It was shown to be desirable to operate the rocket motors in the latter stages at relatively low combustion pressures and the initial stage rocket motor at a comparatively high combustion pressure. For the three stage hydrazine-oxygen rocket the optimum combustion pressure in the initial stage was found to be 600 psi and the optimum combustion pressure in the second and third stages to be 150 psi. The minimum skin gage of .020 inches adequately protects the contents of the satellite payload compartment against possible damage due to meteorite collision, since with this gage there is only one chance in 200 that the payload compartment skin will be penetrated by a meteorite in five days. A pump fed propellant distribution system was found to be more economical of weight than either a fully gas pressurized or a partially pump fed and partially gas pressurized distribution system for rockets of the size of the proposed satellite.

Other studies have shown that the shell of the satellite rocket will reach a maximum temperature of the order of 1500°F during the ascending trajectory. In order to minimize the effect of this high temperature on the mechanical properties of the satellite rocket shell material, a stabilized 18-8 type stainless steel has been selected as a possible material for the shell of the satellite rocket. While the use of a thin shell type of construction using internal pressure to maintain structural stability offers some saving in gross weight, a conventional primary structure consisting of a grid framework of longitudinal and transverse stiffeners covered by a thin skin has been arbitrarily selected for the proposed satellite rocket in order to be conservative. Further study will be necessary before the final selection can be made.

The structural design study verifies the general validity of the method developed to estimate the gross weight of the satellite rocket, since the calculated gross weight of 86,463 pounds determined from the design study agrees very closely with the estimated gross weight of 86,420 pounds for the satellite rocket used in the design study.

Table 9

COMPARATIVE WEIGHT SUMMARY

Item	Stage I		Stage II		Stage III	
	Est.Wt Lbs	Cal.Wt Lbs	Est.Wt Lbs	Cal.Wt Lbs	Est.Wt Lbs	Cal.Wt Lbs
Group T	2365.0		644.0		232.5	
Payload Comp't Shell						98.3
Payload Comp't Internal Struct.						50.4
Tank Comp't Shell		779.3		205.3		98.6
Stabilizing Group Shell		401.0				
Hydrazine Tank		356.5		70.8		25.4
Oxygen Tank		261.4		118.8		19.5
Hydrogen Peroxide Tank		20.7		1.1		0.3
Helium Tank		107.6		29.5		14.3
Thrust Ring		252.0		135.0		83.3
Motor Mount		88.5		31.6		45.6
Group M	865.0		186.0		0.0	
Stage Separation Device		60.0		40.0		0.0
Miscellaneous Structure		805.0*		146.0*		
Group A	2800.0		580.0		192.3	
Rocket Motor		1700.0*		400.0*		99.0*
Turbine, Pumps, Plumbing		1100.0*		180.0*		93.3*
Group C	4184.0		247.0		48.4	
Internal Controls		1484.0*		247.0*		48.4*
External Controls		2700.0*		0.0		0.0
Group P*	57574.0		12360.0		3062.0	
Hydrazine		34290.0*		7390.0*		1830.0*
Oxygen		22740.0*		4900.0*		1215.0*
Hydrogen Peroxide		544.0*		70.0*		17.0*
Group L*	18632.0		4615.2		1080.0	
Payload		18773.5		4818.4		1080.0
Total (Gross Weight)	86420.0	86463.2	18632	18773.5	4615.2	4818.4

*Estimated value

APPENDIX I.

DERIVATION OF WEIGHT ESTIMATION METHODS

The weight of the satellite rocket will be most conveniently determined if it is considered to be composed of a number of families of items so selected that the weight of the members of each family will be a function of the same parameters and will vary in a similar manner. The weight of a given stage of a staged rocket may therefore be expressed as the sum of the weights of certain functional groups [1]

$$(W)_j = [\, W_T + W_M + W_A + W_C + W_P{}^* + W_L{}^* \,]_j \tag{41}$$

where

W = the gross weight of a given stage, including the succeeding stage as payload.

W_T = the weight of items whose mass is determined by the maximum load they must carry, and by their geometrical configuration. Includes such items as tankage and primary structure.

W_M = miscellaneous weight. Includes an allowance for stage separation devices.

W_A = the weight of items whose mass is a function of the mass rate of propellant flow. Includes rocket motors, turbine, pumps, and plumbing.

W_C = the weight of items whose mass is a function of the degree of control required. Includes servo motors, jet vanes, fins, etc.

$W_P{}^* = W_P + W_p$ = the weight of all items whose mass is a function of time. Includes the weight of the propellants W_P and the weight of the auxiliary fuels W_p.

$W_L{}^*$ = the payload.

As a starting point in the development of a method to be used in estimating the weight of the satellite rocket, a weight breakdown of the German A-4, modified to yield a mass ratio (gross weight ÷ gross weight less propellant and auxiliary fuel weight) of 4, is presented in Table 10[11,12].

Table 10

WEIGHT BREAKDOWN OF THE A-4

ITEM	MASS RATIO 3.25		MASS RATIO 4	
	Wt in Pounds	Sub Total	Wt in Pounds	Sub Total
A. Basic Weight				
1. Instrument Compartment		875		720
Radio Equipment	155		125	
Structure	325		270	
Electrical Equipment	315		260	
Nitrogen Supply	80		65	
2. Tank Compartment		1,400		1,165
Shell	790		650	
Oxygen Tank	375		315	
Alcohol Tank	235		200	
3. Motor Compartment		2,572		2,105
Shell	410		340	
Plumbing	70		60	
Rocket Motor	1,025		835	
Motor Mount	260		210	
Auxiliary Equipment	(807)		(660)	
Turbine	170		140	
Pumps	220		180	
Gas Generator	65		55	
Compressed Air and Bottles	240		195	
H_2O_2 Tank	92		75	
Permanganate	20		15	
4. Flight Controls		1,335		1,090
External Fins	750		610	
Jet Vanes and Drive	470		385	
External Vanes and Drive	115		95	
B. Propellants and Fuels		18,960		20,480
Alcohol	7,650		8,260	
Oxygen	10,930		11,810	
Hydrogen Peroxide	380		410	
C. Warhead		2,150		1,750
D. Gross Weight		27,300		27,300

1. EQUATIONS GOVERNING SATELLITE ROCKET WEIGHT

Having defined the functional components of a satellite rocket, it is now necessary to investigate the properties of each of these groups and to develop an expression which will represent its weight.

a. Group T - Tanks and Primary Structure

The weight of Group T is a function of the geometric size and shape of the structure and also of the maximum structural load, except when minimum gage considerations apply. Since the propellants constitute a large part (60% to 90%) of the gross weight of the satellite rocket, it is to be expected that the overall structural weight will be influenced to a large extent by the propellant tank weight. For simplicity the tank shape shall be assumed to be a right circular cylinder with an effective radius and length consistent with the satellite rocket geometry, the location of the tank in the rocket, and the amount of propellent to be contained. The weight breakdown of the components of the A-4 that properly fall in this group is presented in Table 11.

Table 11

WEIGHT BREAKDOWN OF GROUP T FOR THE A-4

Mass Ratio = 4

ITEM	WEIGHT IN POUNDS	SUB TOTAL
1. Instrument Compartment		335
Structure	270	
Nitrogen Supply	65	
2. Tank Compartment		1165
Shell	650	
Oxygen Tank	315	
Alcohol Tank	200	
3. Motor Compartment		820
Motor Mount	210	
Shell	340	
Compressed Air and Tank	195	
H_2O_2 Tank	75	
4. Total		2320

It has been frequently assumed that for small rockets using liquid propellants a gas pressurized fuel system is the most economical from a weight standpoint, while for larger vehicles a turbine pump arrangement is the best. It was first assumed, and later demonstrated[10], that the satellite rocket would fall in the latter class. For this reason a pump fed propellant distribution system has been contemplated for the satellite rocket, and a turbine fuel tank is included in Group T.

Considering first the case of larger structures where the effects of minimum gages need not be considered, we may develop the expression for the weight of Group T in the following manner.

The pressure acting on the tank walls is given by

$$p = k \gamma_p n \ell_T , \tag{42}$$

where p = the pressure in psi, k = a constant,

γ_p = the specific weight of the propellants in lbs/in.^{3},

n = the applied load factor (the number of g's acceleration), and

ℓ_T = the tank length in inches.

The applied tensile stress in the tank is given by

$$\sigma_T = \frac{p\, r_T}{t} \tag{43}$$

where σ_T is the applied tensile stress in psi, r_T the effective tank radius in inches, and t the wall thickness of the tank in inches. Equating the applied tensile stress (σ_T) to the allowable tensile stress which is constant for any given material, Eq.(43) may be written as

$$t = k_2 p r_T = k_3 \gamma_p n \ell_T r_T . \tag{44}$$

The area of the side walls of the tank (A_w) is

$$A_w = 2\pi \ell_T r_T \tag{45}$$

while the area of the tank bottom, A_b, is $A_b = \pi r_T^2$. (46)

The total surface area of the tank, A_T, then becomes

$$A_T = A_w + A_b = \pi(2\ell_T r_T + r_T^2). \tag{47}$$

The volume of the tank material, V_T, may then be expressed as

$$V_T = t \times A_T = k_3 \gamma_p n \pi (2\ell_T^2 r_T^2 + \ell_T r_T^3) , \tag{48}$$

and the weight of the wall material is

$$W_w = \gamma_T V_T , \tag{49}$$

where γ_T is the specific weight of the wall material, a constant for a given material.

Since the weight of Group T has been assumed to vary as the weight of the propellant tank it may be expressed as

$$W_T = k_4 \gamma_T V_T = k_6 \gamma_P n\pi (2\ell_T^2 r_T^2 + \ell_T r_T^3). \tag{50}$$

The propellant weight may be written as

$$W_P = \gamma_P \pi r_T^2 \ell_T ; \tag{51}$$

therefore the ratio of the weight of Group T to the propellant weight is

$$\frac{W_T}{W_P} = k_5 n(2\ell + r) = k_8 n(\ell + \frac{r}{2}). \tag{52}$$

The value of the constant k_8 may be obtained from a consideration of the weight of the components of the A-4 that properly apply for any given case.

The weight of Group T for the initial stage of the proposed satellite rocket would include all of the items that appear in the A-4 with the exception of the instrument compartment structure. Also, since a separate allowance will be made for the weight of the hydrogen peroxide (turbine fuel) tank in any given stage we have the following from Table 10 and Table 11 for the A-4:

$$W_T^* = 1975 \text{ lbs} \qquad W_P = 20,480 \text{ lbs}$$

$$n = 2.0 \qquad \ell_T = 243 \text{ in.} \qquad r_T = 31.5 \text{ in.}$$

$$\frac{W_T^*}{W_P} = \frac{1975}{20480} = k_8\, n(\ell_T + \frac{r_T}{2}) = k_8(2)(243 + \frac{31.5}{2})$$

$$k_8 = .000186.$$

Therefore for the initial stage of a multi-staged satellite rocket

$$W_T^* = .000186\ W_P\, n(\ell_T + \frac{r_T}{2}) . \tag{53}$$

Another term must be added to Eq. (53) representing the weight of the turbine fuel tank required in this stage. For the development of this term it will first be necessary to consider the amount of turbine fuel required. It is contemplated that the satellite rocket will have a pump fed propellant distribution system, the centrifugal pumps being powered by a turbine driven by gases generated by the decomposition of hydrogen peroxide under the action of a suitable catalyst.

The total amount of work required to pump the propellants for a given stage is given by

$$W_k = V_P' p' , \tag{54}$$

where W_k is the work required in ft-lbs, V_p' is the total volume of the propellants to be pumped in cubic feet, and p' is the pressure difference across the pump in lbs/sq ft.

The following efficiencies are assumed as being representative of those that may be reasonably expected:

$$\eta_{\text{turbine}} = 25\%$$

$$\eta_{\text{pump}} = 65\%$$

$$\eta_{\text{combustion}} = 90\%$$

Therefore the total energy E_k required by the turbine fuel is

$$E_k = \frac{V_p'p'}{(.25)(.65)(.90)} \tag{55}$$

$$= 6.85\ V_p'p' \quad \text{ft-lbs}$$

$$= .00874\ V_p'p' \quad \text{BTU}$$

Assuming that a 90% (by weight) solution of hydrogen peroxide is used as turbine fuel, a specific enthalpy of 1137 BTU/lb may be obtained from the turbine fuel under typical operating conditions[13]. The weight of turbine fuel required therefore is

$$W_p = \frac{.00874\ V_p'p'}{1137}$$

$$W_p = 7.69\ V_p'p' \times 10^{-6}\ \text{lbs.} \tag{56}$$

Since the specific gravity of a 90% solution of H_2O_2 is 1.393, the volume of turbine fuel required is

$$V_p' = \frac{7.69 \times 10^{-6}}{(1.393)(62.4)} V_p'p'$$

$$= 8.85\ V_p'p' \times 10^{-7} \quad (\text{ft}^3)$$

$$V_p = 1.274 \frac{W_p}{\gamma_p} p \times 10^{-4} \quad (\text{in.}^3) \tag{57}$$

where W_p is the weight of propellants in pounds, γ_p is the specific weight of propellants in lbs/in.^{3}, and p is the pressure difference across the pump in psi.

If the turbine fuel tank is assumed to be spherical, then

$$V_p = \frac{4}{3}\pi r^3 = 1.274 \frac{W_p}{\gamma_p} \times 10^{-4}$$

$$r^3 = 3.04 \frac{W_p}{\gamma_p} p \times 10^{-5} \quad (\text{in.}^3) \tag{58}$$

$$d_p = .0624 \left(\frac{W_p}{\gamma_p} p\right)^{1/3} \quad (\text{in.}) \tag{59}$$

The weight of the turbine fuel tank will then be

$$W_e = t \times A \times \gamma_e \ , \tag{60}$$

where $t = p_e r/2\sigma$ = the thickness of the tank wall in inches, p_e the pressure in psi, r the radius of the tank in inches, σ the allowable tensile stress in the tank wall in psi (for stainless steel σ = 93,500 psi), $A = 4\pi r^2$ the surface area of the tank wall in square inches, and γ_e the specific weight of the tank wall material in lbs/in.^{3} (for stainless steel γ_e = .284 (lbs/in.^{3}).

The maximum pressure in the turbine fuel tank may be expressed as

$$p_e = n d \gamma_p + p_s \quad (\text{psi}), \tag{61}$$

where p_s is the static pressure in psi.

Therefore the weight of the turbine fuel tank (for stainless steel) becomes

$$W_e = \frac{(n d_p \gamma_p + p_s) \ r}{2 \ (93,500)} (4\pi r^2)(.284)$$

$$W_e = 5.801 \frac{W_p}{\gamma_p} p \left[.0624 \ n \ \gamma_p \left(\frac{W_p}{\gamma_p} p\right)^{1/3} + p_s \right] (10)^{-10} . \tag{62}$$

Allowing 15% of the tank shell weight for fittings, doublers, etc., the total turbine fuel tank weight becomes

$$W_e = 6.671 \frac{W_p}{\gamma_p} p \left[.0624 \ n\gamma_p \left(\frac{W_p}{\gamma_p} p\right)^{1/3} + p_s \right] (10)^{-10} . \tag{63}$$

The total weight of Group T for the initial stage of a multi-staged vehicle will now be the sum of Eq. (53) and Eq. (63):

$$W_T = .000186 \ W_p \ n \left(\ell_T + \frac{r_T}{2}\right) + 6.671 \frac{W_p}{\gamma_p} p \left[.0624 \ n\gamma_p \left(\frac{W_p}{\gamma_p} p\right)^{1/3} + p_s \right] (10)^{-10} . \tag{64}$$

The weight of Group T for an intermediate stage of a multi-staged satellite rocket would include all of the items that compose this group in the A-4, with the exception of the instrument compartment structure, which appears only in the final stage, the shell that surrounds the motor compartment, and the fixed aerodynamic control surfaces (fins). It is felt that the intermediate stages of the satellite will be operating at altitudes sufficiently high (atmospheric density sufficiently low) that the fins would be ineffectual as a stabilization device and that the increase in aerodynamic drag due to the omission of the shell around the motor compartment would be negligible. Again, a separate allowance will be made for the weight of the hydrogen peroxide (turbine fuel) tank, therefore we have from Tables 10 and 11 for the A-4:

$$W_T^* = 1645 \text{ lbs} \qquad W_P = 20{,}480 \text{ lbs}$$

$$n = 2.0 \qquad \ell_T = 243 \text{ in.} \qquad r_T = 31.5 \text{ in.}$$

Substituting the above value in Eq. (52) and solving for k_8 gives

$$k_8 = .000155$$

$$\therefore \; W_T^* = .000155 \; W_P \; n \left(\ell_T + \frac{r_T}{2}\right), \tag{65}$$

and from Eqs. (63) and (65)

$$W_T = .000155 \; W_P \; n \left(\ell_T + \frac{r_T}{2}\right) + 6.671 \frac{W_P}{\gamma_P} p \left[.0624 \; n \gamma_p \left(\frac{W_P}{\gamma_P} p\right)^{1/3} + p_s\right] 10^{-10}. \tag{66}$$

The weight of Group T for the final stage of a multi-staged satellite rocket may be again developed, where minimum gage considerations do not apply, from a consideration of the weights of the respective components of the A-4. The expression for the weight of the final stage will be identical to that for the weight of the intermediate stages with the exception that a separate term must now be added to express the weight of the payload (instrument) compartment. An allowance of 35 cubic feet is made for the volume of the payload compartment and the gage of the sheet used to cover this compartment is determined from a consideration of the amount of protection required against meteorite penetration. A stainless steel skin .020 inches thick was found to be adequate for this purpose. Furthermore, an allowance of 5% of the weight of the equipment to be installed in the payload compartment has been made to account for the necessary supporting structure. Therefore the weight of the payload compartment may be expressed as

$$W = C + .05 \; W_L' \; , \tag{67}$$

where the value of C, the weight of the payload compartment shell, is a function of the vehicle geometry. The weight of Group T then becomes

$$W_T = .000155 \; W_P \; n \left(\ell_T + \frac{r_T}{2}\right) + 6.671 \frac{W_P}{\gamma_P} p \left[.0624 \; n \gamma_p \left(\frac{W_P}{\gamma_P} p\right)^{1/3} + p_s\right] 10^{-10} + C + .05 \; W_L' \; . \tag{68}$$

It has been found, however, that the size of the final stage is so small that the weight of Group T is no longer a function of the load applied to the structure, but rather is determined by minimum gage considerations. A minimum gage of .020 inches has been established as being a reasonable value for the sheet material from which the satellite rocket shell is fabricated. It is again assumed that the weight of the motor mount will vary in direct proportion to the weight of the propellant tanks. The ratio of the weight of the motor mount to the weight of the tanks for the A-4 is, from Table 11, 210/1165. Another factor to be taken into consideration in the estimation of the weight of the final stage is that, from a consideration of the optimum trajectory characteristics, it has been found desirable to insert a period of coasting in the final stage followed by a final application of thrust. This means that the propellant tanks must be compartmented by the installation of suitable diaphrams to separate the pressurizing gas from the liquid contents, thus preventing a mixing or emulsifying effect during the acceleration free coasting period. Taking the above items into consideration, and allowing an additional 50% of the wall material weight for the stiffening members gives

$$\begin{aligned} W_T^* &= \left[1.5\ (2\pi r_T \ell_T) + 6\pi r_T^2\right](.020)(.284)\left(1 + \frac{210}{1160}\right) \\ &= .06322\ r_T\ (\ell_T + 2\ r_T)\ , \end{aligned} \tag{69}$$

and from Eqs. (69), (67) and (63)

$$\begin{aligned} W_T = {} & .06322\ r_T\ (\ell_T + 2\ r_T) + 6.671\,\frac{W_P}{\gamma_P}\,p\left[.0624\ n\gamma_p\left(\frac{W_P}{\gamma_P}\,p\right)^{1/3}\right. \\ & \left. + \ p_s\right](10^{-10}) + C + .05\ W_L'\ . \end{aligned} \tag{70}$$

It has been found that the weight of the hydrogen peroxide (turbine fuel) tank is generally very small when compared to the other items in Group T. For this reason the weight of the turbine fuel tank may be neglected, in most cases, when estimating the weight of this group. Simplifying the equation giving W_T by neglecting the weight of the hydrogen peroxide tank gives:

for the initial stage

$$(W_T)_1 = .000186\ W_P\ n\ \left(\ell_T + \frac{r_T}{2}\right)\ ; \tag{71}$$

for an intermediate stage

$$(W_T)_x = .000155\ W_P\ n\ \left(\ell_T + \frac{r_T}{2}\right)\ ; \tag{72}$$

for the final stage where minimum gages apply

$$(W_T)_N = .06322\ r_T\ (\ell_T + 2\ r_T) + C + .05\ W_L'\ ; \tag{73}$$

and for the final stage where minimum gages do not apply

$$(W_T)_N = .000155\ W_p\ n\left(\ell_T + \frac{r_T}{2}\right) + C + .05\ W_L' \ ; \tag{74}$$

In the use of the equations giving W_T, it should be pointed out that the value of n used must be consistent with the value of W_p used. For the initial stage the design load factor n is that occurring at the beginning of operation of that stage (minimum acceleration, maximum propellant weight), while for the intermediate and final stages the design load factor n is the maximum value of n that occurs during the initial stage (maximum acceleration, maximum propellant weight). The applied load factor n is defined by [8].

$$n_t g_s = \frac{\dfrac{dm_p}{dt}\, g_s\, I}{M_0 - \displaystyle\int_0^t \epsilon \frac{dm_p}{dt} dt} \ , \tag{75}$$

where n_t is the applied load factor at time t, g_s is the standard acceleration of gravity at sea level, I the specific impulse, m_p the mass of propellants discharged at time t, $\epsilon\ (dm_p/dt)$ the time rate of change of mass of the stage, dm_p/dt the time rate of propellant flow, and M_0 the initial mass of the stage.

For a constant mass rate of propellant flow the time rate of propellant flow may be written as

$$\frac{dm_p}{dt} = \frac{M_p}{t_b} \tag{76}$$

where M_p is the initial mass of the propellants and t_b is the duration of burning for the stage.

Therefore

$$n_t = \frac{\dfrac{M_p}{t_b} I}{M_0 - \displaystyle\int_0^t \epsilon \frac{M_p}{t_b} dt} \ , \tag{77}$$

since $M_p/M_0 = \nu$ and $\epsilon\nu = \nu^*$ [Eq. (101)]

$$n_t = \frac{\dfrac{\nu}{t_b} I}{1 - \dfrac{\nu^*}{t_b} t} . \tag{78}$$

Eq. (78) gives the instantaneous value of the applied load factor n_t at any time t. The values of n_i, the initial (minimum) applied load factor, and n_f, the final (maximum) applied load factor may be determined readily from Eq. (78), and are

$$n_i = \frac{\nu}{t_b} I_i \tag{79}$$

$$n_f = \frac{\nu}{t_b} I_f \frac{1}{1 - \nu^*} \tag{80}$$

and from Eqs. (79) and (80)

$$n_i = n_f (1 - \nu^*) \frac{I_i}{I_f} \ . \tag{81}$$

b. Group M - Miscellaneous Weight

The weight of Group M constitutes an allowance for the stage separation devices required in a multi-staged satellite rocket. Since the purpose of staging is to reduce to a minimum the mass which must be accelerated to the orbital velocity, it follows that the stage separation devices should be part of that portion of the rocket which is to be discarded, rather than a part of that portion of the rocket which is to go on after the separation is accomplished. For this reason no allowance is made for the weight of Group M in the final stage, while in the intermediate and initial stages the weight of Group M has been assumed to be 1% of the gross weight of the stage

$$W_M = .01\, W \ . \tag{82}$$

c. Group A - Thrust Producing Equipment

The weight of Group A, the thrust producing equipment, includes the weights of all items whose mass is a function of the mass rate of propellant flow. The weights of the components of the A-4 which properly fall in this classification are given in Table 12.

Table 12

WEIGHT BREAKDOWN OF GROUP A FOR THE A-4

Mass Ratio = 4

ITEM	WEIGHT IN POUNDS
1. Turbine	140
2. Pumps	180
3. Gas Generator	55
4. Permanganate	15
5. Plumbing	60
6. Rocket Motor	835
TOTAL	1285

Weight of the Rocket Motor

The weight of a rocket motor may be expressed as a fraction of the thrust F that the motor produces. On the basis of past experience in rocket motor design it will be assumed that the weight of the rocket motor is given by

$$W_m = \frac{F}{80}, \tag{83}$$

since

$$F = \frac{I\, W_p}{t_b}, \tag{84}$$

$$W_m = \frac{I}{80}\frac{W_p}{t_b}. \tag{85}$$

Weight of the Plumbing

The weight of the plumbing is a function of the pressure in the system, and the mass rate of propellant flow. An analysis of the performance of the A-4 shows the pressure loss in the lines from the discharge side of the pump to the motor to be

Alcohol lines = 6 atm loss

Oxygen line = 3 atm loss

∴ Average loss = 4.5 atm = 67.5 psi.

The A-4 combustion pressure is 15 atm, therefore the mean total pressure at the discharge side of the pumps is 292.5 psi. At full thrust the A-4 propellant rate is given as[12] 316 lbs/sec.

Assuming, for the initial and intermediate stages of the satellite rocket, the minimum weight of the plumbing to be 10 pounds, the plumbing weight, W_h, may then be expressed as

$$W_h = 10 + k\,p\frac{W_p}{t_b}, \tag{86}$$

where p is the pressure difference across the propellant pump, and W_p/t_b is the weight rate of propellant flow.

Substituting the values given above in Eq. (86) and solving for k

$$k = .000541$$

$$W_h = 10 + .000541\,p\frac{W_p}{t_b}. \tag{87}$$

The weight of the plumbing in the final stage will be greater than the weight of this item in the initial and intermediate stages, due to the additional complexity of the system necessitated by the period of coasting that occurs in this stage. An increase of 50% in the weight of the plumbing in the final stage is made to allow for this added complexity. Therefore, for the final stage

$$(W_h)_N = 15 + .000812\, p \frac{W_p}{t_b} . \tag{88}$$

Weight of Auxiliary Equipment

The weight of the auxiliary equipment, consisting of the turbine, the pumps, and the turbine gas generating chamber with its catalytic lining, will vary with the power required to accomplish the pumping of the propellants. The minimum weight of this group has been assumed to be 45 pounds. If W_{au} represents the weight of the auxiliary units, γ_p the specific weight of the propellants, p the pressure difference across the pumps, W_p the weight of the propellants, and t_b the burning time, then the weight of this group may be expressed as

$$W_{au} = 45 + k \frac{1}{\gamma_p} p \frac{W_p}{t_b} . \tag{89}$$

From the A-4

$$W_{au} = 390 \text{ lbs}$$

$$\gamma_p = 60.3 \text{ lbs/ft}^3 = .0349 \text{ lbs/in.}^3$$

$$p = 292.5 \text{ psi}$$

$$\frac{W_p}{t_b} = 316 \text{ lbs/sec} .$$

Allowing a 10% decrease in the weight of this group for expected improvements in design, and substituting the above values in Eq. (89), the value of k can be determined:

$$k = .000116$$

$$W_{au} = 45 + .000116 \frac{p}{\gamma_p} \frac{W_p}{t_b} . \tag{90}$$

The weight of Group N will be the sum of the weights of the rocket motor, the plumbing, and the auxiliary equipment. Therefore from the above, for the initial and intermediate stages of a multi-staged satellite rocket

$$W_A = \left[\frac{I}{80} + \left(\frac{.000116}{\gamma_p} + .000541 \right) p \right] \frac{W_p}{t_b} + 55 , \tag{91}$$

and for the final stage

$$\left(W_A\right)_N = \left[\frac{I}{80} + \left(\frac{.000116}{\gamma_P} + .000812\right)P\right]\frac{W_P}{t_b} + 60 \ . \qquad (92)$$

d. **Group C - Controls**

The weight of Group C includes the weight of all items whose mass is a function of the degree of control required. From past aircraft experience the ratio of the weight of the control system to the gross weight of the unit has been found to vary as the square root of the linear scale dimension L for similar systems. Therefore

$$W_C = k\, L^{½}\, W \ , \qquad (93)$$

but L varies as $W^{1/3}$;

$$W_C = k\, W^{7/6} \ . \qquad (94)$$

The components of the A-4 that properly fall in this group are given in Table 13.

Table 13

WEIGHT BREAKDOWN OF GROUP C FOR THE A-4

Mass Ratio = 4

ITEM	WEIGHT IN POUNDS
1. External Fins	610
2. Jet Deflectors & Drive	385
3. External Vanes & Drive	95
TOTAL	1090

Since neither the final stage nor the intermediate stages of the satellite rocket will have external fins or external vanes

$$k = \frac{385}{(27,300)^{7/6}}$$

$$W_C = 385\left(\frac{W}{27,300}\right)^{7/6} \ . \qquad (95)$$

For the initial stage, however, the presence of the external stabilizing fins must be taken into account, giving

$$\left(W_L\right)_1 = 1090\left(\frac{W}{27,300}\right)^{7/8}. \tag{96}$$

e. Group P* - Propellants and Fuels

Group P* includes the weight of all items whose mass is a function of time. These items are the propellants used in the rocket motor and the fuel used to drive the turbine. If ν represents the ratio of the propellant weight to the gross weight of a given stage, then

$$W_P = \nu W. \tag{97}$$

The weight of the turbine fuel required has been found from Eq. (56) to be

$$W_p = 7.69\ V_P'\ p'(10)^{-6}$$

where W_p is the weight of the turbine fuel required in pounds, V_P' is the volume of the propellants in ft^3, and p' is the pressure difference across the propellant pumps in lbs/ft^2. This equation may be written in a more convenient form. Let W_P be the propellant weight, p the pressure difference across the propellant pumps in psi, and γ_P be the specific weight of the propellants in $lbs/in.^3$, then

$$W_p = .64\,\frac{W_P}{\gamma_P}\,p\,(10)^{-6}. \tag{98}$$

The ratio of the weight of the turbine fuel to the gross weight may be expressed as

$$\frac{W_p}{W} = \frac{.64\,p}{\gamma_P}\,\frac{W_P}{W}(10)^{-6}. \tag{99}$$

Allowing 1% of the propellant weight for evaporation and losses incurred during the starting of the rocket motors, the total weight of Group P* then becomes

$$W_P^* = 1.01\,W_P + \frac{.64\,p}{\gamma_P}\nu\,W(10)^{-6} = \left(1.01 + \frac{.64\,p}{\gamma_P}(10)^{-6}\right)\nu\,W. \tag{100}$$

Let ϵ represent the quantity $\left(1.01 + \frac{.64\,p}{\gamma_P}(10)^{-6}\right)$ and let ν^* represent the ratio of W_P^* to W then

$$\nu^* = \epsilon\,\nu \tag{101}$$

and $$W_P^* = \nu^*\,W. \tag{102}$$

f. Group L* - Payload

The weight of Group L* is defined as that portion of the stage which is carried as payload. Therefore for the initial and intermediate stages, where $(W_L{}^*)_j$ represents the stage payload and (W_{j+1}) represents the gross weight of the succeeding stage

$$(W_L{}^*)_j = W_{j+1} \,. \tag{103}$$

For the final stage, however, the stage payload is defined as

$$(W_L{}^*)_N = W_L{}' \,, \tag{104}$$

where $W_L{}'$ represents the weight of the fixed equipment occurring in the final stage. $W_L{}'$ in turn is defined as

$$W_L{}' = W_L + W_I + W_Q \,, \tag{105}$$

where W_L represents the satellite rocket payload, W_I represents the weight of the intelligence providing equipment (brains), and W_Q represents the weight of the orbital power plant together with its fuel.

The weight of the vehicle payload (W_L), which includes instrumentation for the observation and measurement of physical phenomena, and communications equipment including that used for the telemetering of data, has been arbitrarily established at 500 pounds.

The weight of the intelligence providing equipment (W_I) is made up of the two auto pilots, one providing trajectory control intelligence, and the other providing orbital control intelligence. An allowance of 180 pounds has been provided for this item.

The weight of the orbital power plant which is used to power the orbital control devices and the communications equipment has been established at 400 pounds.

Therefore

$$W_L{}' = 1080 \text{ lbs.}$$

2. EQUATIONS GOVERNING SATELLITE ROCKET GEOMETRY

Several of the expressions for the weight of a particular functional group, as developed above, require a knowledge of the geometric size of the satellite rocket before they can be evaluated. For this reason the following expressions governing the rocket geometry will be developed.

Studies of the shape of the satellite rocket indicate that it should be roughly conical in shape, increasing to a maximum diameter, then decreasing in diameter to a value at the aft end such that the base drag will be a minimum. Fig. 1 illustrates the simplified shape parameters and defines the notation used.

The volume of the satellite rocket payload compartment has been arbitrarily established at 35 cubic feet, corresponding to a weight of fixed equipment (W_L') of 1080 pounds. Furthermore, it is assumed that 40% of the tank volume in an intermediate stage, or in the initial stage, can be placed forward of the aft end of the succeeding stage (see Fig. 1). From the above the following conditions are defined:

a. Volume of the satellite rocket payload compartment $V_\alpha = 35$ ft^3.

b. The 'effective' volume of an intermediate stage tank compartment $V_x = .6(V_p)_x$ where $(V_p)_x$ represents the volume of propellants required.

c. The 'effective' volume of the initial stage tank compartment $V_1 = .6(V_p)_1$ where $(V_p)_1$ represents the volume of propellants required.

Length of Vehicle Payload Compartment

$$V_\alpha = 35 \text{ ft}^3 = 60480 \text{ in.}^3 = \frac{\pi}{3} m\,(m \tan\phi)^2$$

$$\frac{(60480)(3)}{\pi} = 57754 = m^3 \tan^2\phi$$

$$m = 38.6541\,(\cot\phi)^{2/3}. \tag{106}$$

Vehicle Radius at Aft End of Payload Compartment

$$\rho_N = m \tan\phi\,. \tag{107}$$

Length of Final Stage Tank Compartment

$$V_N = V_p = \frac{W_p}{\gamma_p},$$

where W_p = propellant weight (lbs), γ_p = specific weight of propellant (lbs/in.^{3}) and V_p = propellant volume (in.^{3}).

Let $y = m + \delta_N$

$$V_N = \frac{W_p}{\gamma_p} = \frac{\pi}{3}\left[y(y\tan\phi)^2 - \frac{\rho_N}{\tan\phi}(\rho_N)^2\right]$$

$$= \frac{\pi}{3\tan\phi}\left(y^3\tan^3\phi - \rho_N{}^3\right)$$

$$y^3\tan^3\phi = \frac{3\,W_p\tan\phi}{\pi\,\gamma_p} + \rho_N{}^3$$

$$y = \left(\frac{3}{\pi}\frac{W_p\tan\phi}{\gamma_p} + \rho_N{}^3\right)^{1/3}\frac{1}{\tan\phi}$$

$$\delta_N = y - m = y - \frac{\rho_N}{\tan \phi}$$

$$\delta_N = \left[\left(.955 \frac{W_P \tan \phi}{\gamma_P} + \rho_N^3 \right)^{1/3} - \rho_N \right]_N \frac{1}{\tan \phi} \; . \qquad (108)$$

Length of Final Stage

$$\lambda_N = m + \delta_N + s_N \; . \qquad (109)$$

Vehicle Radius at End of Final Stage

$$\rho_{N-1} = \lambda_N \tan \phi \; . \qquad (110)$$

Effective Length of Intermediate Stage Tank Compartment

Derivation is the same as for Eq. (108) except that

$$V_x = (.6 \, V_P)_x = .6 \left(\frac{W_P}{\gamma_P} \right)_x \; .$$

Therefore

$$\delta_x = \left[\left(.573 \frac{W_P \tan \phi}{\gamma_P} + \rho_x^3 \right)^{1/3} - \rho_x \right]_x \frac{1}{\tan \phi} \; . \qquad (111)$$

Length of an Intermediate Stage

$$\lambda_x = \lambda_{x+1} + \delta_x + s_x \; . \qquad (112)$$

Vehicle Radius at the End of an Intermediate Stage

$$\rho_{x-1} = \lambda_x \tan \phi \; . \qquad (113)$$

Effective Length of the Initial Stage Tank Compartment

$$V_1 = (.6 V_P)_1 = .6 \left(\frac{W_P}{\gamma_P} \right)_1 = \frac{\pi}{3} \left[r_b^2 \left(\frac{r_b}{\tan \phi} \right) - \rho_1^2 \left(\frac{\rho_1}{\tan \phi} \right) + r_b^2 \left(\frac{r_b}{\tan \psi} \right) - r_c^2 \left(\frac{r_c}{\tan \psi} \right) \right]$$

$$.573 \left(\frac{W_P}{\gamma_P} \right)_1 = \left(\frac{r_b^3}{\tan \phi} + \frac{r_b^3}{\tan \psi} - \frac{\rho_1^3}{\tan \phi} - \frac{r_c^3}{\tan \psi} \right)$$

$$.573 \left(\frac{W_P}{\gamma_P} \right)_1 \tan \phi = \left(r_b^3 + r_b^3 \frac{\tan \phi}{\tan \psi} - \rho_1^3 - r_c^3 \frac{\tan \phi}{\tan \psi} \right)$$

$$.573 \frac{W_p}{\gamma_p} \tan \phi = r_b^3 (1 + U) - \rho_1^3 - r_c^3 U , \tag{114}$$

where $U = \dfrac{\tan \phi}{\tan \psi}$ (115)

$$\tan \psi = \frac{(r_c - r_d)}{s_1} \tag{116}$$

$$r_b = .5 \lambda_1 \frac{d}{\ell_0} \tag{117}$$

From Eq. (116)

$$\frac{r_c - r_d}{s_1} = \tan \psi = \frac{r_b - r_d}{\lambda_1 (1 - \ell/\ell_0)} .$$

Therefore

$$r_c = \frac{s_1 (r_b - r_d)}{\lambda_1 (1 - \ell/\ell_0)} + r_d . \tag{118}$$

Given the values of ℓ/ℓ_0 and d/ℓ_0, Eqs. (114), (115), (116), (117) and (118) are a set of implicit expressions from which the value of λ_1 may be obtained.

$$\delta_1 = \lambda_1 - \lambda_2 - s_1 . \tag{119}$$

Length of Motor Compartment (any stage)

The length of the motor compartment may be taken as

$$s_j = (d_p + s_{au} + \ell_m + c)_j , \tag{120}$$

where d_p is the diameter of the turbine fuel (hydrogen peroxide) tank, s_{au} is the length along the axis of the vehicle of the auxiliary units, ℓ_m is the length of the rocket motor, and c an arbitrary allowance for clearance between the units.

From Eq. (59)

$$(d_p)_j = .0624 \left(\frac{W_p}{\gamma_p} p \right)_j^{1/3} .$$

The volume that the auxiliary units will occupy varies directly with the weight of the auxiliary units, and the length that this group occupies along the axis of the vehicle will vary as the cube root of the volume. From the A-4 for a mass ratio of 4, W_{au} = 390 lbs; s_{au} = 28 in.,

since $W_{au} \sim V_{au} \sim s_{au}^3$

$$W_{au} = k\, s_{au}^3$$

$$k = \frac{390}{(28)^3} = .01777 .$$

From Eq. (90)

$$.01777\ (s_{au})_j^3 = \left(45 + .000116\, \frac{p}{\gamma_P}\, \frac{W_P}{t_b}\right)_j$$

$$(s_{au})_j^3 = \left(2530 + .00653\, \frac{p}{\gamma_P}\, \frac{W_P}{t_b}\right)_j$$

$$(s_{au})_j = \left(2530 + .00653\, \frac{p}{\gamma_P}\, \frac{W_P}{t_b}\right)_j^{1/3} . \tag{121}$$

Therefore

$$s_j = \left[c + .0624\left(\frac{W_P}{\gamma_P}\, p\right)^{1/3} + \left(2530 + .00653\, \frac{p}{\gamma_P}\, \frac{W_P}{t_b}\right)^{1/3} + \ell_m\right]_j \tag{122}$$

where ℓ_m is the length of the rocket motor[4].

Effective Propellant Tank Radius

For the purpose of simplicity the propellant tanks are assumed to be right circular cylinders having a length ℓ_T and an effective radius r_T such that the tank will contain the required volume of propellant.

$$(V_P)_N = \left(\frac{W_P}{\gamma_P}\right)_N = (\pi\, r_T^2\, \ell_T)_N$$

$$(r_T)_N^2 = \left(\frac{W_P}{\pi \gamma_P \ell_T}\right)_N$$

$$(r_T)_N = \left[.564\left(\frac{W_P}{\gamma_P\, \ell_T}\right)^{1/2}\right]_N . \tag{123}$$

For an intermediate or the initial stage

$$.6\, V_P = \frac{.6\, W_P}{\gamma_P} = \pi\, r_T^2\, \delta$$

$$r_T^2 = \frac{.6\, W_P}{\pi \gamma_P\, \delta}$$

$$r_T = .437\left[\frac{W_P}{\gamma_P\, \delta}\right]^{1/2} . \tag{124}$$

Effective Propellant Tank Length

For the final stage the propellant tank length $(\ell_T)_N$ is given by

$$(\ell_T)_N = \delta_N . \tag{125}$$

In the intermediate and final stages it was assumed that 40% of the volume of the propellants could be placed in a tank which surrounds the succeeding stage motor compartment. The propellant tank will then have the external shape of a right circular cylinder, the aft end being flat and the forward end having an indentation in the form of a truncated right circular cone. The portion of the tank from its aft end to the aft end of the indentation will contain 60% of the volume of the propellants and will have a length equal to the effective length of the tank compartment for that stage. The portion of the tank from the aft end of the indentation to the forward end of the tank will contain 40% of the propellant volume and will have the same external diameter as the aft portion of the tank. The diameter of the aft end of the indentation will be assumed to be one-third of the external diameter of the tank, while the diameter of the indentation at the forward end of the tank will be assumed to be two-thirds of the external diameter.

If V represents the total tank volume, d the external diameter of the tank, δ the effective tank compartment length and ℓ_T the total propellant tank length, then

$$\ell_T = k\delta + \delta \ . \tag{126}$$

From a consideration of the tank proportions as described above

$$.4V = \pi \frac{d^2}{4} k\delta - \pi \frac{k\delta}{3}\left[\left(\frac{d}{3}\right)^2 + \left(\frac{d}{3}\right)\left(\frac{d}{6}\right) + \left(\frac{d}{6}\right)^2\right]$$

$$.6V = \pi \frac{d^2}{4}\delta \ .$$

Therefore

$$\frac{\pi d^2 \delta}{2.4} = \frac{\pi d^2 k \delta}{2.16}$$

and

$$k = .90 \ .$$

Then from Eq. (126) the length of the propellant tank in the initial stage or in an intermediate stage is

$$\ell_T = 1.90\ \delta \ . \tag{127}$$

3. COMPUTATION PROCEDURE

Since the evaluation of the gross weight of a multi-staged satellite rocket by use of the expressions developed in the preceding section is an iteration, or trial and error process, the computational work involved is considerable. This numerical work may be greatly facilitated, however, by the use of a suitable computation form.

APPENDIX II.

DETAILED SUMMARY OF CALCULATIONS

In order to investigate the effect of the various weight parameters considered in earlier sections of this report, it was necessary to estimate the weight of a great many satellite rockets having different given design conditions. A detailed summary of the results of these computations is given below in Table 14. For convenience the various cases investigated have been grouped by the propellant system utilized. The propellant system characteristics fuel-oxidant mole ratio α, and theoretical specific impulse at sea level I_o, may be obtained from Table 20 of ref. 2 and page 41, ref. 1. The remaining weight parameters along with the results of the computations are listed in the body of the table.

Table 14

DETAILED SUMMARY OF CALCULATIONS

Item	N	n	h (mi)	W_L' (lbs)	ℓ/ℓ_o	(p_c)	(p_c) Alt	ν	$\frac{\Sigma W_p*}{W}$	$\frac{W_B}{W_{av}}$	$\frac{\Sigma W_B}{W}$	ℓ_o (in.)	d (in.)	W (lbs)
HYDRAZINE-OXYGEN PROPELLANT SYSTEM														
1	2	5.0	350	1080	.6481	300	300	.8048	.8710	.1211	.1236	571	187.6	228,100
2	3	4.0	350	1080	.6321	400	150	.6635	.8587	.0999	1288	528.2	144.4	86,943
3	3	4.5	350	700	.6857	150	150	.6707	.8471	.1282	.1450	541	158	91,580
4	3	4.5	350	700	.6507	300	150	.6621	.8388	.1316	.1530	524	144	85,250
5	3	4.5	350	700	.6549	450	150	.6580	.8316	.1337	1600	484	135	83,800
6	3	4.5	350	700	.6479	600	150	.6553	.8287	.1348	1619	518	140	83,690
7	3	4.5	350	700	.6591	150	300	.6694	.8452	.1305	1485	534	158	96,330
8	3	4.5	350	700	.6981	300	300	.6610	.8371	.1346	1553	511	144	89,440
9	3	4.5	350	700	.6467	450	300	.6566	.8315	.1376	.1610	501	138	87,840
10	3	4.5	350	700	.6403	600	300	.6540	.8266	.1387	.1653	506	138	87,000
11	3	4.5	350	700	.6990	150	450	.6688	.8427	1347	.1504	535	158	101,700
12	3	4.5	350	700	.6660	300	450	.6602	.8122	.1372	.1572	512	144	93,370
13	3	4.5	350	700	.6547	450	450	.6560	.8298	.1375	.1627	506	140	91,620
14	3	4.5	350	700	.6435	600	450	.6534	.8246	.1415	.1675	505	138	91,400
15	3	4.5	350	700	.6382	150	600	.6684	.8353	.1372	.1519	539	160	106,530
16	3	4.5	350	700	.6362	300	600	.6599	.8354	.1395	.1575	547	151	97,300
17	3	4.5	350	700	.6343	450	600	.6556	.8284	.1424	.1646	506	140	95,800
18	3	4.5	350	700	.6422	600	600	.6529	.8234	.1445	.1693	506	138	96,560
19	3	4.5	350	1080	.6325	400	150	.6578	.8515	.1049	.1359	533.4	146	86,101
20	3	5.0	350	500	.6568	400	150	.6532	.8542	.1134	.1339	443.4	119.6	42,161
21	3	5.0	350	700	.6443	400	150	.6532	.8460	.1111	.1416	478	129.2	56,150
22	3	5.0	350	1080	.6876	150	150	.6657	.8616	.1011	.1260	541.6	158.8	88,080
23	3	5.0	350	1080	.6553	300	150	.6572	.8525	.1054	.1346	528	148	84,031
24	3	5.0	350	1080	.6420	400	150	.6532	.8433	.1100	.1440	537.8	147.6	86,500
25	3	5.0	350	1080	.6419	450	150	.6530	.8461	.1078	.1409	525	144	82,794
26	3	5.0	350	1080	.6354	600	150	.6503	.8421	.1092	.1453	524	142	82,295
27	3	5.0	350	1080	.4000	400	150	.6477	.8162	.1207	.1723	854	85.4	94,236
28	3	5.0	350	1080	.4000	400	150	.6509	.8269	.1159	.1612	741	103.8	91,787
29	3	5.0	350	1080	.4000	400	150	.6552	.8338	.1137	.1547	676.8	121.8	93,934
30	3	5.0	350	1080	.4000	400	150	.6694	.8434	.11166	.1469	613.6	159.6	110,782
31	3	5.0	350	1080	.5000	400	150	.6469	.8229	.1188	.1651	845.4	84.6	90,159
32	3	5.0	350	1080	.5000	400	150	.6523	.8399	.1109	.1477	651.4	117.2	86,771
33	3	5.0	350	1080	.5000	400	150	.6691	.8516	.1081	.1381	558.06	167.4	104,513
34	3	5.0	350	1080	.8000	400	150	.6648	.8562	.1054	.1324	524.9	199.4	94,832
35	3	5.0	350	1080	.6000	400	150	.6507	.8439	.1093	.1430	641.2	115.4	83,029
36	3	5.0	350	1080	.6000	400	150	.6579	.8513	.1068	.1364	559.4	145.4	88,123
37	3	5.0	350	1080	.6000	400	150	.6688	.8562	.10606	.1331	521.3	177.2	101,014
38	3	5.0	350	1080	.7000	400	150	.6462	.8303	.1180	.1574	863.3	86.4	87,597
39	3	5.0	350	1080	.7000	400	150	.6521	.8501	.1069	.1368	592.5	130.4	81,914
40	3	5.0	350	1080	.7000	400	150	.6700	.8578	.1053	.1313	505.2	192	101,624

Table 14 (cont'd.)

DETAILED SUMMARY OF CALCULATIONS

Item	N	n	h (mi)	W_L' (lbs)	ℓ/ℓ_0	(p_c)	(p_c) Alt	ν	$\frac{\Sigma W_p^*}{W}$	$\left(\frac{W_B}{W}\right)_{av}$	$\frac{\Sigma W_B}{W}$	ℓ_0 (in.)	d (in.)	W (lbs)
HYDRAZINE - OXYGEN PROPELLANT SYSTEM (CONT'D.)														
41	3	5.0	350	1080	.8000	400	150	.6460	.8313	.1184	1565	889.43	89	88,274
42	3	5.0	350	1080	.8000	400	150	.6507	.8495	.10724	.1371	609.4	134	80,689
43	3	5.0	350	1080	.6000	400	150	.6464	.8274	.1183	.1604	849.5	85	88,469
44	3	5.0	350	1080	.6647	300	300	.6634	.8515	.1086	.1373	531	150	96,460
45	3	5.0	350	1500	.6470	400	150	.6532	.8395	.1103	.1480	595	164	120,540
46	3	5.0	350	2000	.6518	400	150	.6532	.8342	.1134	.1536	652.3	181	164,466
47	3	5.5	350	1080	.6449	400	150	.6497	.8379	.1148	.1498	547.1	150.2	88,026
48	3	5.75	350	1080	.6450	400	150	.6482	.8346	.1173	.1532	551.7	151.6	89,087
49	3	6.0	350	1080	.6459	400	150	.6468	.8313	.1199	.1567	556.4	153	90,498
50	3	6.25	350	1080	.6467	400	150	.6455	.8281	.1225	1601	561.1	154.6	91,984
51	3	6.5	300	700		300	300	.66	.7752	.1750	.2202	600	120	157,000
52	4	5.0	350	1080	.6859	300	300	.5581	.8267	.1110	.1616	605	166.6	94,330
53	4	6.5	300	700		300	300	.55	.7477	.1770	.2464	552	110.4	121,047
54	5	5.0	350	1080	.6160	300	300	.4797	.8074	.1108	.1816	647.7	185.4	98,230
HYDRAZINE-FLUORINE PROPELLANT SYSTEM														
55	2	5.0	350	1080	.6931	300	300	.7716	.8605	.1124	.1274	453.5	131.4	88,505
56	2	5.75	300	700	.6316	300	300	.755	.8101	.16875	1863	608	164	196,000
*57	2	5.75	300	700	.6371	300	300	.755	.8155	.1655	.1803	576	157	166,200
58	2	6.5	300	700		300	300	.783	.8486	.1437	.1461	250	332	131,000
*59	2	6.5	300	700		300	300	.783	.8644	.1220	.1270	230	278	81,700
60	2	6.5	300	700		300	300	.783	.8431	.1472	.1520	325	209	143,000
*61	2	6.5	300	700		300	300	.783	.8612	.1240	1306	286	165	85,000
62	2	6.5	300	700		300	300	.783	.8399	.1492	1555	378	182	151,000
63	2	6.5	300	700		300	300	.783	.8361	.1517	.1596	427	166	162,000
*64	2	6.5	300	700		300	300	.783	.8548	.1280	1376	380	144	92,500
65	2	6.5	300	700		300	300	.783	.8305	.1552	1656	510	144	181,000
*66	2	6.5	300	700		300	300	.783	.8506	.1307	.1423	445	122	98,300
67	2	6.5	300	700		300	300	.783	.8243	.1592	1723	590	132	208,000
*68	2	6.5	300	700		300	300	.783	.8463	.1335	1470	507	110	105,000
**69	2	6.5	300	700		300	300	.783	.8485	.1342	1445	518.4	104.4	100,000
70	2	6.5	300	700		300	300	.783	8216	.1524	1753	680	136	223,000
71	2	6.5	300	700		300	300	.783	.8189	.1627	.1782	658	122	239,000
*72	2	6.5	300	700		300	300	.783	.8421	.1362	.1517	565	102	112,500
73	2	6.5	300	700		300	300	.783	.8142	.1657	1830	720	114	275,000
*74	2	6.5	300	700		300	300	.783	.8384	.1385	1558	620	96	120,000
75	2	6.5	300	700		300	300	.783	.8095	.1687	.1883	787	109	324,000
*76	2	6.5	300	700		300	300	.783	.8345	.1410	1501	677	92	129,000
77	3	4.0	300	700	.7652	300	300	.638	.8644	.1106	.1224	458	123.6	52,600
78	3	4.0	300	700	.6400	300	300	.638	.8424	.1196	.1457	467	119.4	58,640
*79	3	4.5	300	700	.6534	300	300	.615	.8307	.1285	.1560	438	121.4	52,300
80	3	4.5	300	700	.6477	300	300	.615	.8305	.1260	.1554	434	119.2	50,730
81	3	4.5	300	700	.6575	300	300	.626	.8571	.1147	.1281	457	128	47,460
82	3	4.5	300	700	.6320	300	300	.626	.8341	.1242	.1527	481	120.8	52,992
83	3	5.0	300	700	.6338	300	300	.6192	.8467	.1211	.1384	467	124.2	47,300
84	3	5.0	300	700	.6265	300	300	.6192	.8261	.1297	.1605	475	126.8	52,330
85	3	5.0	350	1080	.6568	300	300	.6263	.8427	.1081	.1400	475.7	132.6	60,450
86	3	5.0	350	1080	.6763	150	150	.6359	.8545	.1021	.1279	508.6	146	61,890
87	3	5.0	350	1080	.6509	300	150	.6279	.8454	.1051	1364	488.5	134	58,640
88	3	5.0	350	1080	.6356	450	150	.6238	.8400	.1065	.1403	483	130	57,340
89	3	5.0	350	1080	.6196	600	150	.6211	.8341	.1085	.1453	489	132	57,300
90	3	5.5	300	700	.6278	300	300	.6148	.8394	.1267	1459	471	126	47,640
91	3	5.5	300	700	.6501	300	300	.6148	.8196	.1350	.1669	469.4	120	52,497
92	3	6.5	195	700		300	300	.6200	.7848	.1640	2049	463.2	95	68,010

* 90% A-4 Weight

**86.3% A-4 Weight

Table 14 (cont'd.)

DETAILED SUMMARY OF CALCULATIONS

Item	N	n	h (mi)	W_L' (lbs)	ℓ/ℓ_0	(p_c)	(p_c) Alt	ν	$\frac{\Sigma W_p}{W}$	$\left(\frac{W_B}{W}\right)_{av}$	$\frac{\Sigma W_B}{W}$	ℓ_0 (in.)	d (in.)	W (lbs)
HYDRAZINE-FLUORINE PROPELLANT SYSTEM (CONT'D.)														
93	3	6.5	300	700	.6500	300	300	.6300	.7894	.1640	.2018	427	115.7	80,000
94	3	6.5	300	700	.6422	300	300	.6080	.8263	.1365	.1593	481	121.6	49,130
95	3	6.5	300	700	.6471	300	300	.6080	.8068	.1449	.1805	486	134	54,260
96	3	6.6	300	700		300	300	.5420	.7599	.1645	.2298	449.5	90	68,500
97	3	6.5	300	700		300	300	.6300	.7812	.1670	.2107	488	97.6	83,600
98	3	6.5	300	700	.5500	300	300	.6330	.8033	.1555	.1870	421	124.2	73,700
99	3	6.5	300	700	.6000	300	300	.6330	.8008	.1545	.1906	409	131.4	73,000
100	3	6.5	300	700	.6500	300	300	.6330	.8023	.1540	.1881	400	139.4	72,500
101	3	6.5	300	700	.7000	300	300	.6330	.8039	.1535	.1859	395	148.2	72,100
102	3	6.5	300	700	.7500	300	300	.6330	.8058	.1530	.1844	388	156.2	71,800
103	3	6.5	300	700	.7750	300	300	.6330	.8059	.1528	.1844	387	162.6	71,425
104	3	6.5	300	700	.6500	300	300	.6370	.8090	.1500	.1807	225	163.6	73,500
105	3	6.5	300	700	.6500	300	300	.6415	.8121	.1490	.1782	209.2	192.2	77,200
106	3	6.5	300	700	.6500	300	300	.6465	.8147	.1488	.1767	203.8	219	82,375
107	3	6.5	300	700		300	300	.6590	.7796	.1701	.2156	578	115.6	142,000
108	3	6.5	800	700		300	300	.6795	.7719	.1751	.2251	683	136.6	238,599
109	3	6.5	905	700		300	300	.6900	.7414	.1924	.2572	839	167.8	505,620
110	3	6.5	1000	700		300	300	.7000	.7435	.1890	.2553	842.5	192.4	600,000
111	3	7.5	300	700	.6315	300	300	.6030	.8117	.1474	.1752	498	134	52,800
112	3	7.5	300	700	.6558	300	300	.6030	.7932	.1556	.1949	501	140	58,320
113	3	8.0	300	700	.6494	300	300	.6010	.8065	.1518	.1804	506	140	54,380
114	3	8.0	300	700	.6580	300	300	.6010	.7870	.1603	.2007	513	144	60,300
115	4	5.0	350	1080	.6423	300	300	.5221	.8177	.1094	.1641	548.4	149.8	61,035
116	4	6.5	300	700		300	300	.5150	.7466	.1675	.2434	461	92.2	70,000
117	4	6.5	700	700		300	300	.5400	.7534	.1628	.2389	500.8	100.1	90,100
118	4	6.5	1000	700		300	300	.5700	.7464	.1759	.2494	612	122.4	169,380
119	5	5.0	350	1080	.6253	300	300	.4459	.7931	.1112	.1905	670.3	178.4	66,160
120	5	6.5	escape vehicle	700		300	300	.5950	.7457	.1735	.2537	2790	558	1,090,000
HYDROGEN-OXYGEN PROPELLANT SYSTEM														
121	2	5.0	350	1080	.6882	300	300	.6962	.8183	.1547	.1646	612	179	63,172
122	2	6.5	300	700		300	300	.7400	.7835	.1955	.2124	370	750	170,000
*123	2	6.5	300	700		300	300	.7400	.8214	.1595	.1685	310	502	69,400
124	2	6.5	300	700		300	300	.7400	.7736	.2030	.2233	540	472	228,000
*125	2	6.5	300	700		300	300	.7400	.8132	.1640	.1776	410	316	75,800
126	2	6.5	300	700		300	300	.7400	.7644	.2085	.2333	640	412	303,000
*127	2	6.5	300	700		300	300	.7400	.8076	.1677	.1839	490	276	82,000
128	2	6.5	300	700		300	300	.7400	.7576	.2130	.2407	720	374	420,000
*129	2	6.5	300	700		300	300	.7400	.8025	.1712	.1813	550	252	88,700
130	2	6.5	300	700		300	300	.7400	.7437	.2225	.2559	860	328	2,000,000
*131	2	6.5	300	700		300	300	.7400	.7928	.1777	.2005	660	220	105,000
*132	2	6.5	300	700		300	300	.7400	.7847	.1832	.2097	740	200	124,000
*133	2	6.5	300	700		300	300	.7400	.7777	.1880	.2175	810	185	147,000
*134	2	6.5	300	700		300	300	.7400	.7703	.1930	.2259	880	174	183,000
*135	2	6.5	300	700		300	300	.7400	.7622	.1985	.2350	940	165	250,000
*136	2	6.5	300	700		300	300	.7400	.7577	.2015	.2401	1015	152	312,000
137	3	5.0	350	1080	.6666	300	300	.5480	.7771	.1542	.2003	569.9	161.6	47,790
138	3	6.5	300	700		300	300	.5440	.7363	.1950	.2455	624	128	38,500
139	3	6.5	300	700	.6504	300	300	.5520	.7298	.2235	.2594	615	170.2	64,700
140	3	6.5	300	700		300	300	.5600	.7391	.2001	.2530	790	158	88,713
141	4	5.0	350	1080	.6347	300	300	.4488	.7443	.1531	.2339	678.8	183.6	49,023
142	5	5.0	350	1080	.6257	300	300	.3791	.7044	.1539	.2756	551.2	208.2	54,951

*90% A-4 Weight

Table 14 (cont'd.)

DETAILED SUMMARY OF CALCULATIONS

Item	N	n	h (mi)	W_L (lbs)	ℓ/ℓ_0	(p_c)	(p_c) Alt	ν	$\frac{\Sigma W_p}{W}$	$\left(\frac{W_B}{W}\right)_{av}$	$\frac{\Sigma W_B}{W}$	ℓ_0 (in.)	d (in.)	W (lbs)
ALCOHOL-OXYGEN PROPELLANT SYSTEM														
143	3	5.0	350	1080	.6939	300	300	.6942	.8485	.1152	.1451	590	176.8	169,700
144	4	5.0	350	700	.6586	300	300	.5883	.8253	.1154	.1674	711.4	199.6	148,500
145	4	6.5	300	700	.6899	300	300	.6000	.7729	.1613	.2243	804	235.4	275,000
146	4	6.5	300	700		300	300	.5840	.7203	.1925	.2771	720	144	287,000
147	5	5.0	350	1080	.6397	300	300	.5083	.8056	.1151	.1872	794	218.6	150,000
148	6	5.0	350	1080	.6213	300	300	.4465	.7818	.1163	.2115	919	246.6	162,070
ANALINE ACID PROPELLANT SYSTEM														
149	3	5.0	350	1080	.6989	300	300	.7286	.8579	.1107	.1383	641.4	190.8	282,182
150	4	5.0	350	1080	.6586	300	300	.6240	.8403	.1091	.1548	709.2	202	223,964
151	4	6.5	300	700		300	300	.6200	.6949	.2097	.2846	1000	200	980,000
152	5	5.0	350	1080	.6084	300	300	.5427	.8203	.1094	.1748	822.5	227.2	221,407
153	6	5.0	350	1080	.6366	300	300	.4791	.7986	.1102	.1968	966.1	262.2	234,893
INITIAL STAGE - HYDRAZINE-OXYGEN SUBSEQUENT STAGES - HYDRAZINE-FLUORINE														
154	3	5.0	350	1080	.6775	150	150	.6456	.8568	.1020	.1275	521	150	69,100
155	3	5.0	350	1080	.6500	300	150	.6374	.8470	.1051	.1358	503	138	65,760
156	3	5.0	350	1080	.6600	450	150	.6334	.8420	.1070	.1413	500	136	64,590
157	3	5.0	350	1080	.6254	600	150	.6309	.8371	.1088	.1458	518	138	64,465
HYDRAZINE-HYDROGEN PEROXIDE PROPELLANT SYSTEM														
158	3	5.0	350	1080	.6809	300	300	.6848	.8475	.1110	.1444	583.5	169	134,030
159	4	5.0	350	1080	.6445	300	300	.5794	.8327	.1095	.1584	657.1	180.4	120,463
160	5	5.0	350	1080	.6382	300	300	.4998	.8090	.1112	.1824	744.2	200.2	126,410
161	6	5.0	350	1080	.6232	300	300	.4385	.7845	.1129	.2076	861.5	230	137,776

APPENDIX III

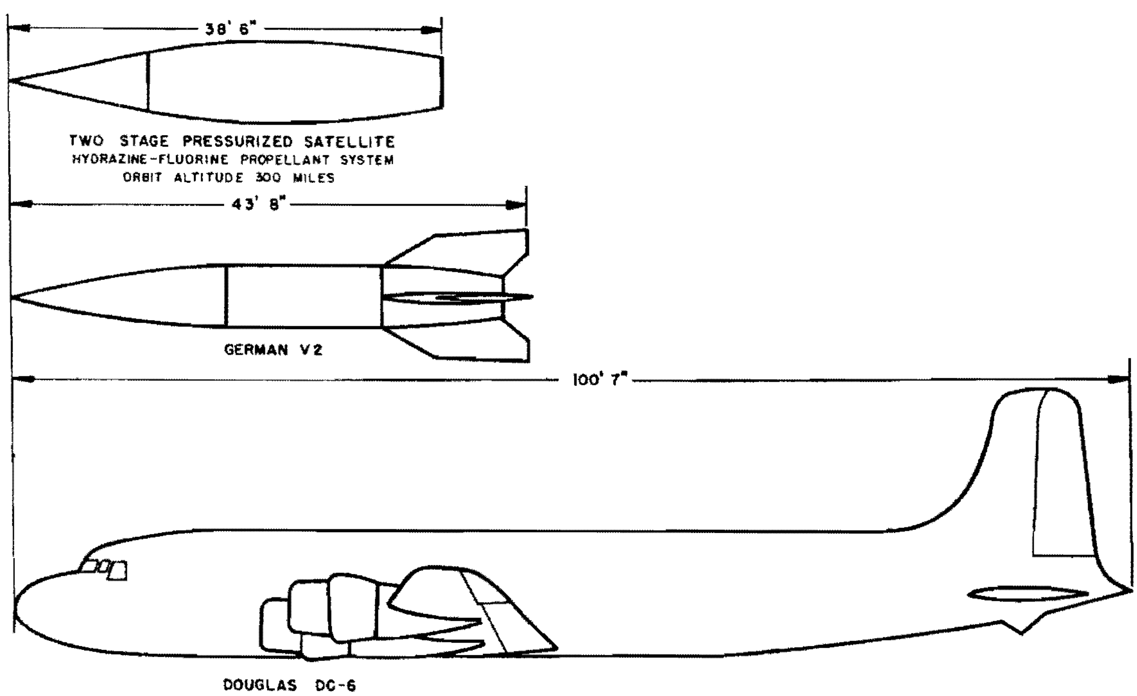
DESIGN OF A PRESSURIZED SATELLITE ROCKET

W. F. Ballhaus

SUMMARY

The use of pressurized primary structure as a basis for the development of a man-made satellite of reasonable size is shown to enhance the possibility of the future construction of such a vehicle. A required gross weight of 48,000 pounds is estimated to be sufficient to enable establishment of a 500 pound payload in a 300 mile altitude orbit about the earth.

The design conditions and assumptions made in the calculations are presented to support the results obtained. It may be seen that such a vehicle is not a superhuman undertaking when it is compared with other commonly accepted successful aircraft as shown in (Fig. 28).



SIZE COMPARISON
FIG. 28

DISCUSSION

This study was undertaken to verify the conclusions reached in El Segundo Report No. 20636 by a more complete treatment of a two-stage rocket propelled satellite utilizing a hydrazine-fluorine propellant system. Original estimates based upon the use of 90% V-2 structural weight and 100% of the other V-2 items indicated that a gross weight of 166,200 pounds would be required to satisfy the performance requirements. Assuming that no external fins are to be used on this vehicle and that the structure is to be simplified somewhat, the control surface weight and miscellaneous weight assumptions were modified so that the gross weight could be reduced to about 77,000 pounds.

The use of pressurized primary structure will account for savings in structure weight of such magnitude that the gross weight can be reduced to 48,000 pounds with a vehicle 38.5 feet long.

This appreciable saving in primary structural weight is accomplished by taking advantage of the high allowable stresses of stainless steel in tension. Enough tension is induced by pressurizing separate compartments of the vehicle to balance any compression loads that may be applied in any flight conditions. The minimum gage was set arbitrarily at .020. This results in high margins of safety in all the covering skin. The high margins eliminate the necessity for considering bending loads induced by the controls, and joint efficiencies, and thus simplifies the consideration of design conditions and the computations required for stress analysis. The design loads and stress analysis are presented in pages 85 and 86.

The primary structure and general arrangement of major items is shown in Fig. 29.

The design conditions are discussed in the following section.

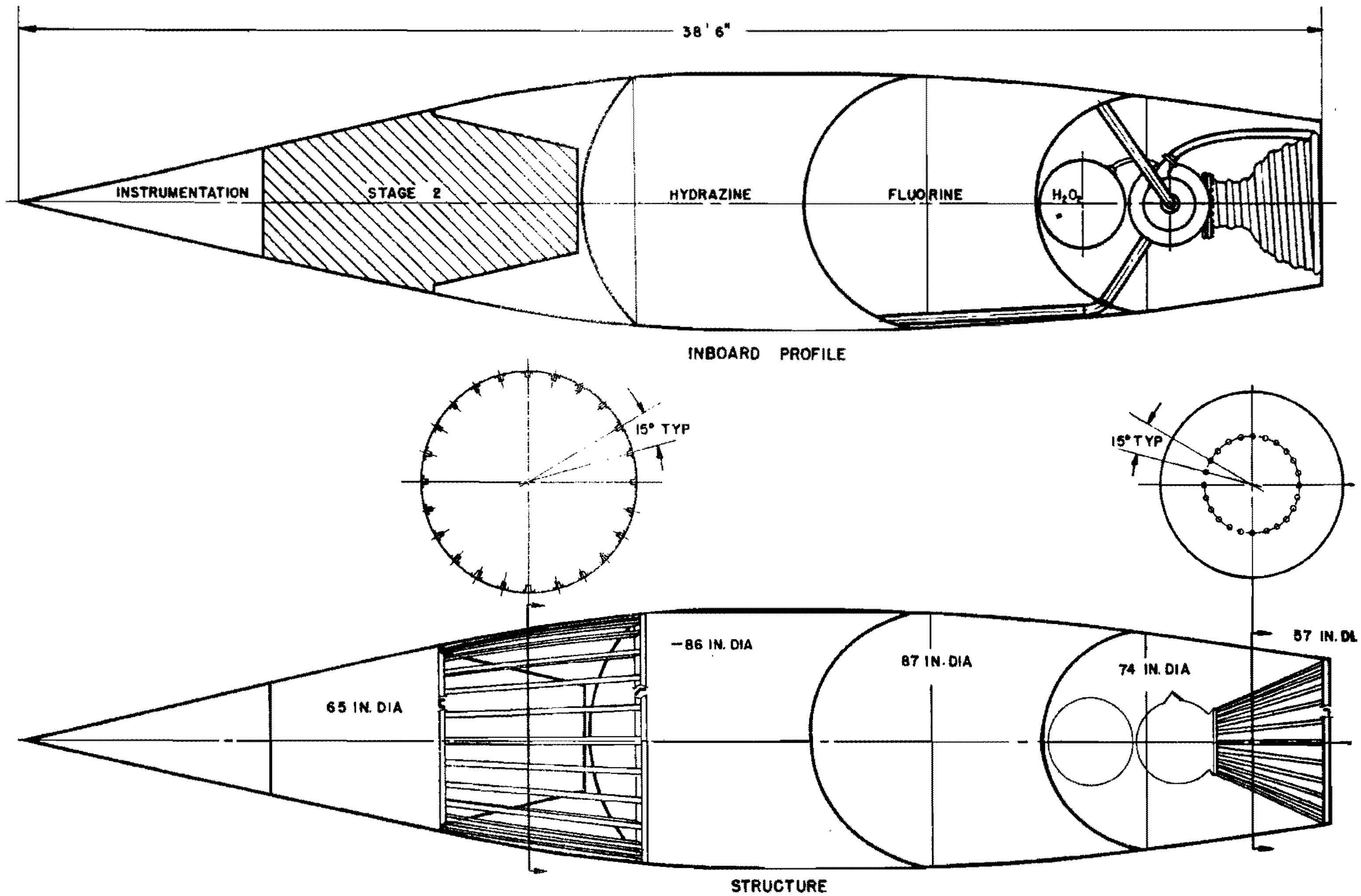
DESIGN CONDITIONS

The limit loads to be applied to the stage one structure reach maximum values at two specific points:

1. Start of flight
2. End of first stage burning.

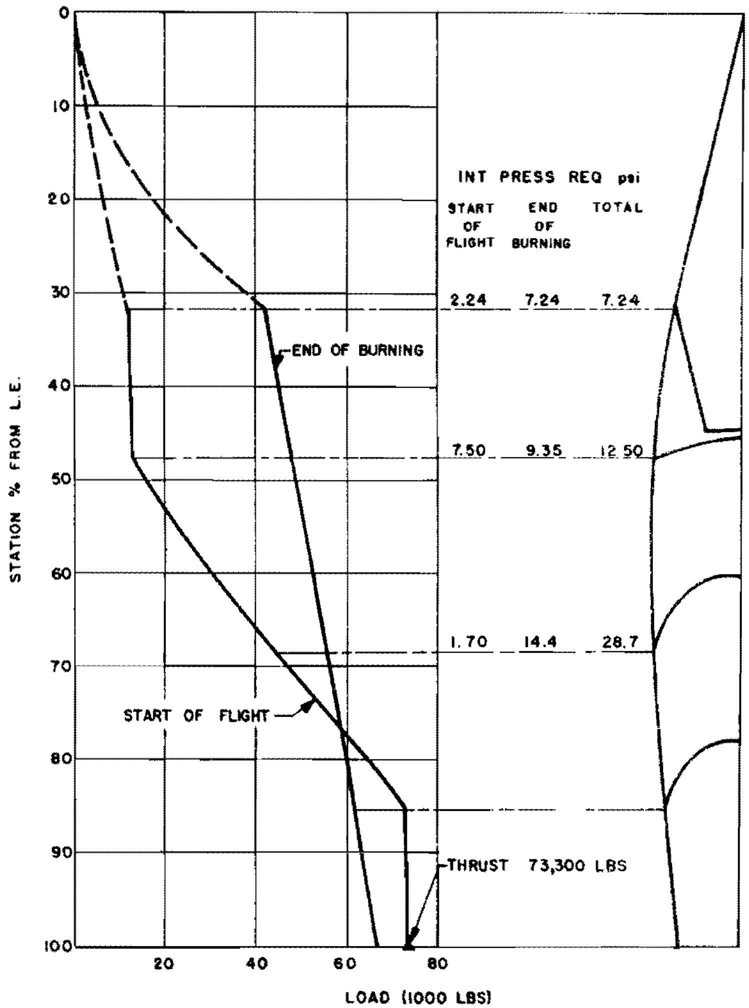
At the start of the flight the hydrostatic pressures of the liquid fuels and the loads from motor to structure are largest. At the end of burning, the load factor has increased to 6.5. All of the instruments, controls, and supporting secondary structure, as well as the affected primary structure, must be designed for this load factor. The primary structure affected is the compression structure supporting stage 2. The tension structure must be pressurized so that it will support stage 2 in all flight conditions. This requirement is easily visualized by referring to Fig. 30.

The minimum gages supply adequate strength at all sections with considerable margins, so a few simple computations are all that is required to indicate that the structural strength will be adequate. The computations are carried out on pages 85 & 86.



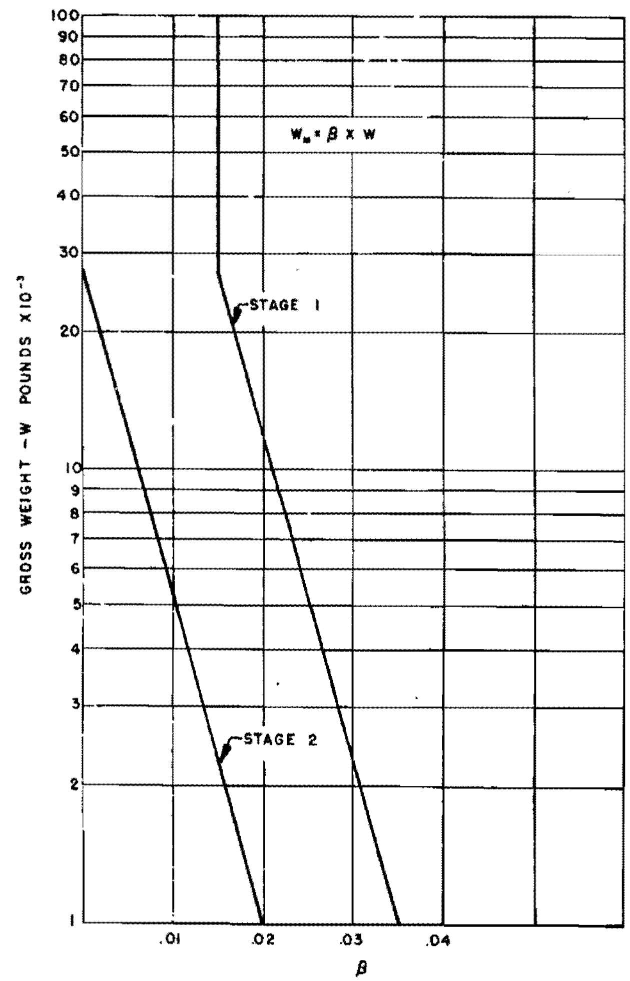
TWO STAGE PRESSURIZED SATELLITE
ORBIT ALTITUDE 300 MILES

FIG. 29



AXIAL COMPRESSION LOADS

FIG. 30



MISCELLANEOUS STRUCTURE WEIGHT FACTOR, β, VS GROSS WEIGHT

FIG. 31

APPROXIMATE STRESS ANALYSIS

Thrust = 73,300 lbs
Base Dia = 57 in.
Internal pressure required $= \dfrac{73{,}300}{\pi/4\ (57)^2} = 28.7$ psi

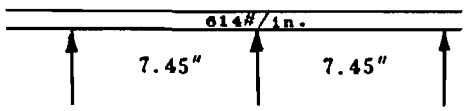
24 struts supporting motor:

Load per strut $= \dfrac{41 \times 73{,}300 \times 1.5}{37.5 \times 24} = 5{,}000$ lbs comp.

Use 1-3/8 in. × .035 in CM steel tube

$w = 4.17$ lbs/100 in.
$W = \dfrac{24 \times 41 \times 4.17}{100}$
$= 41.0$ lbs

Ring connecting 24 struts:



Load/in. = 73,300 × 1.5 ÷ π57 = 614 lbs/in.

$M_{max} = \dfrac{614 \times (7.45)^2}{12} = 2{,}840$ in-lb

$S = \dfrac{2840}{150{,}000} = 0.0189$ in.^{3}

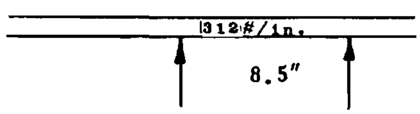
Use 1-1/4 in. × 1-1/4 in × 1/8 in. ∠

$w = 1.01$ lbs/ft
$W = 15.1$ lbs

Stress in. .020 skin:

$f = p\dfrac{R}{t} = \dfrac{28.71 \times 37.5}{.020} = 53{,}800$ psi.

Ring supporting stage 2:



Load/in. 1.5 × 6540 × 6.5 ÷ π65 = 312 lbs/in.

$M_{max} = \dfrac{312 \times (8.5)^2}{12} = 1875$ in-lb

$S = \dfrac{1875}{150{,}000} = .0125$ in.^{3}

Use 1 in.× 1 in. × 1/8 in. ∠

$w = .80$ lbs/ft
$W = 13.6$ lbs

24 struts supporting stage 2:

Load/strut $= \dfrac{1.5 \times 6540 \times 6.5}{24} \times \dfrac{75}{74} = 2690$ lbs

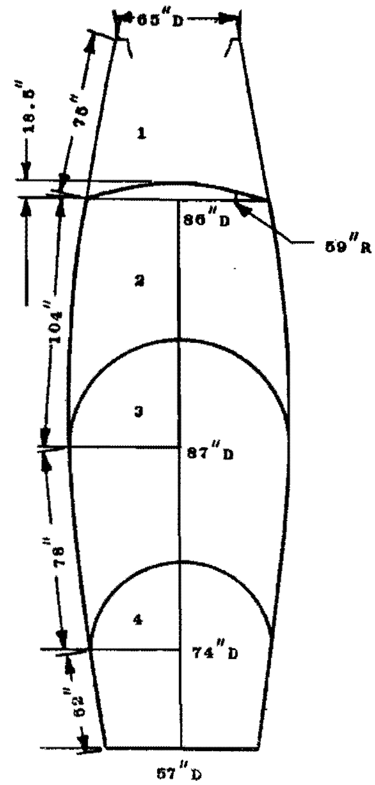
Use 2 in. × 1 in. × .032 in. hats doubler along skin .015 in.

$w = .066$ lbs/in.
$W = 119$ lbs

Ring at bottom of struts: use 1 in.× 1 in. × 1/8 in. ∠

$W = 21.6$ lbs

COMPUTATION OF SHELL WEIGHT



Skin covering 1:

$$A_1 = \frac{75\pi}{2}(65 + 86)$$
$$= 17{,}780 \text{ in.}^2$$

Skin between 1 + 2:

$$A_{12} = 2\pi Rh = 2\pi \times 59 \times 18.5 \text{ in.}$$
$$= 6{,}850 \text{ in.}^2$$

Skin covering 2:

$$A_2 = \frac{104\pi}{2}(86 + 87)$$
$$= 28{,}300 \text{ in.}^2$$

Skin between 2 + 3:

$$A_{23} = 2\pi R^2 = 2\pi\,(43.5)^2$$
$$= 11{,}890 \text{ in.}^2$$

Skin covering 3:

$$A_3 = \frac{78\pi}{2}(87 + 74)$$
$$= 19{,}700 \text{ in.}^2$$

Skin between 3 + 4:

$$A_{34} = 2\pi \times (37)^2$$
$$= 8{,}590 \text{ in.}^2$$

Skin covering 4:

$$A_4 = \frac{62\pi}{2}(74 + 57) = 12{,}760 \text{ in.}^2$$

Total area: 105,870 in.^{2}

Weight (.020) = .006 × 105,870 = 635.2 lbs
From stress analysis (p. 85) W_1 = 211
Primary structure = 846 lbs
+ 20% for laps, fittings, etc.
Total wt. = 1,015 lbs

ASSUMPTIONS USED IN THE ESTIMATE CALCULATIONS

The results of an analysis of the structure, motor, and controls of several rockets lead to the development of a number of expressions relating either directly or indirectly to the weight of each component of the vehiclė to the gross weight. It is estimated that the establishment of a 500 pound payload in a 300 mile altitude orbit around the earth requires a two-stage vehicle having a fuel to gross weight ratio of $\nu = .765$ for each stage. If the load factor is restricted to $n = 6.5$ the thrust can be computed from

$$F = 6.5\ (1 - \nu)\ W\ .$$

Assuming a specific impulse of $I = 335$ lb sec/lb, the burning time is estimated by

$$t_b = .9\ (335/6.5)\left(\frac{\nu}{1 - \nu}\right)$$

$$= 161 \text{ seconds.}$$

The weight and size of the motor, pumps, plumbing, and turbine depends upon the rate of propellant consumption which, since the burning is at a constant rate, is simply the weight of propellants divided by the burning time:

$$\text{Rate of Burning} = \frac{F}{t_b} = \frac{\nu W}{t_b}$$

$$= \frac{.765W}{161}$$

$$= .00475\ W\ .$$

The size relations derived were given in terms of burning rate, but for this study the relations in terms of gross weights are more convenient. The expressions relating gross weight and component weights and sizes used here are, therefore, listed in pages 88 and 89. It should be noted that stage 2 weight is held constant at 6540 pounds.

ASSUMPTIONS USED IN THE ESTIMATE CALCULATIONS

Vehicle Weight:			
Gross weight			48,000
Propellant weight	$= \nu W = .765 \times 48000$	= 36,700	
Combustion chamber nozzle, etc.	$= .0159 \times W$ $= .0159 \times 48000$	= 764	
Turbine, pumps, etc.	$= .00546W + 45$ $= .00546 \times 48000 + 45$	= 307	
Turbine fuel	$= .0153 \times W$ $= .0153 \times 48000$	= 735	
Plumbing	$= .0038W + 10$ $= .0038 \times 48000 + 10$	= 192	
Surfaces and controls	$= 526\ (W/27300)^{7/6}$ $= 526\ (48000/27300)^{7/6}$	= 1,016	
Miscellaneous structure	$= .015W^* = .015 \times 48000$	= 720	
Payload		= 6,540	
Primary structure**		= 1,015	
		47,989 lbs	

*See Fig. 32 for Factor β. β depends on gross weight.

**Primary structure weight was calculated from a consideration of the actual members required to supply necessary strength under the assumed design conditions.

ASSUMPTIONS USED IN THE ESTIMATE CALCULATIONS

Vehicle Size:

Length as required by volume of propellants and general arrangement.

Volumes:

Hydrazine $= \dfrac{.765 \times .458 \times W}{62.4 \times 1.011} = .00555\ W\ \text{ft}^3$

Fluorine $= \dfrac{.765 \times .542 \times W}{62.4 \times 1.108} = .00647\ W\ \text{ft}^3$

Combustion Chamber Diameter $= \sqrt{\dfrac{4}{\pi}\,\dfrac{6.5(1-\nu)W}{1.3 \times 300}} = .0707\sqrt{W}$ inches

Chamber Length = Diameter

Exhaust Diameter $= \sqrt{8} \times$ chamber diameter

Pump and Turbine Cube:

$$s_{au} = \left(3700 + 94.5\,\frac{.765 \times W}{161}\right)^{1/3} \text{ inches}$$

Radius of H_2O_2 Sphere:

$$r = (.091 \times .765 \times W)^{1/3} \text{ inches}$$

where W is gross weight in pounds.

APPENDIX IV.

PRELIMINARY STRUCTURAL ANALYSIS

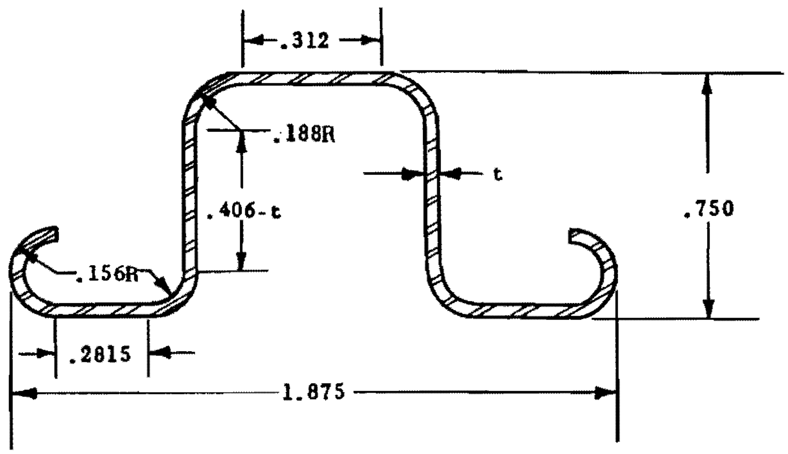
The shell structure of the proposed satellite rocket is made of a stabilized 18-8 type stainless steel skin over a grid framework of longitudinal and transverse stiffeners of the same material.

A minimum gage of .020 inches has been arbitrarily established for the skin of the satellite rocket.

Studies of the amount of material required to adequately protect the contents of the payload compartment against possible damage due to meteorite collisions indicate that a gage of .020 inches will be satisfactory for this purpose.

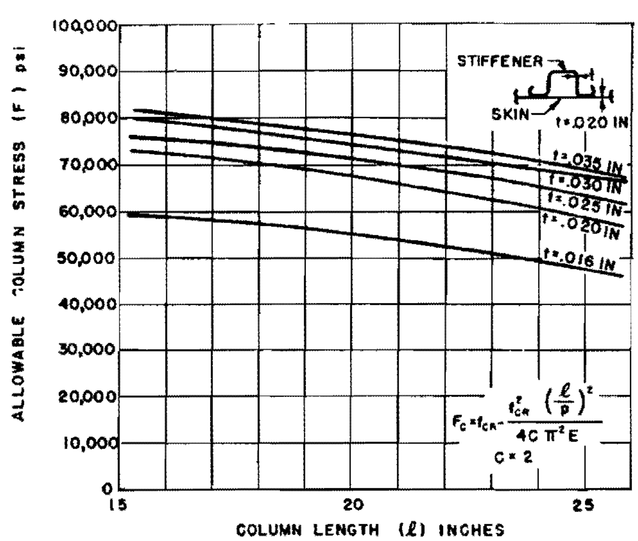
Allowable Load for the Longitudinal Stiffeners as Columns

The longitudinal stiffeners are made of 1/2 hard stabilized 18-8 stainless steel sheet and have the cross-sectional shape shown below.



The variation of the allowable column stress for the longitudinal stiffeners, computed from methods given in ref. 15, with column length as given in Fig. 32.

The allowable column loads for the longitudinal stiffeners used in the satellite rocket are given in Table 15.



ALLOWABLE COLUMN STRESS vs COLUMN LENGTH
FOR LONGITUDINAL STIFFENERS OF VARIOUS GAGES
FIG. 32

Table 15

ALLOWABLE COLUMN LOADS

Skin Gage = .020 inches

Column Length			20 in.	21 in.	22 in.	23 in.	24 in.	25 in.
Stiffener	Gage t (inches)	Area (in.^{2})	F_c (lbs)	F_c (lbs)	F_c (lbs)	F_c (lbs)	F_c (lbs)	F_c (lbs)
A	.016	.0587	3210	3145	3070	2970	2880	2790
B	.020	.0721	4890	4760	4620	4510	4380	4270
C	.035	.125	9550	9380	9220	9000	8800	8700

It should be noted that the allowable column loads as given in Table 15 have been reduced by a factor of 10 per cent from the values given in Fig. 32 to allow for the beam action of the longitudinal stiffeners due to the aerodynamic pressure on the skin of the satellite rocket.

Applied Loads

In the following analysis the satellite rocket shell will be analyzed for axial tension and compression loads. The use of reduced column allowables as given in Table 15 obviates the necessity of considering longitudinal pressure. Further, the effect of bending of the satellite rocket is small and has been generally neglected in this analysis.

The inertia load applied to the satellite rocket structure at a given point is determined from the dead weight load as given in Table 16 and the applied load factor as given in Table 8. The pressure loads are given in Table 8 and the bending moments are given in Fig. 27.

Table 16

AXIAL DEAD WEIGHT LOADS

Station Y	Dead Wt (W) Distributed To Station (lbs)	$\sum_{Y=0}^{Y} W$ (lbs)	$\sum_{Y=565}^{Y} W$ (lbs)
0			
25	36	36	85963
50	108	144	85927
75	276	420	85819
95	221	641	85543
115	1683	2324	85322
140	1978	4302	83639
162	1325	5627	81661
187	890	6517	80336
212	4363	10880	79446
237	4991	15871	75083
260	3519	19390	70092
285	1908	21298	66573
310	5544	26842	64665
335	13721	40563	59121
360	7116	47679	45400
385	8665	56344	38284
410	17972	74316	29619
430	4565	78881	11647
450	4037	82918	7082
475	939	83857	3045
500	625	84482	2106
525	934	85416	1481
545	365	85781	547
565	182	85963	182

SATELLITE ROCKET SHELL STRUCTURE

COMPRESSION LOADS

Station Y	Number of Longitudinals A	B	C	Column Length (inches)	n	$\sum_0^Y W$ (lbs)	f_c (lbs)	F_c (lbs)	M.S.
0									
25			4	25	5	36	180	34800	high
50			4	25	5	144	720	34850	high
75	4		4	20	5	420	2150	51040	high
95	4		4	20	5	641	3205	51040	high
115	12		4	25	5	2324	11620	68280	high
140	12		4	22	5	4302	21510	73720	high
162	20		4	25	5	5627	26135	120200	high
187	20		4	25	5	6517	32585	120200	high
212	20		4	25	5	10880	54400	90600	.6
237	20		4	23	5	15871	79355	104310	.3
260		20	4	25	5	19390	96940	120200	.2
285	20		4	25	1.44	21298	30650	90600	high
310	20		4	25	1.44	26842	38600	90600	high
335	24		4	25	1.44	40563	58400	101760	.7
360	24		4	25	1.44	47679	68500	101760	.5
385	24		4	25	1.44	56344	81000	101760	.25
410		24	4	20	1.44	74316	107000	155500	.45
430		24	4	20	1.44	78881	113300	155500	.4
450		24	4	25	1.44	82918	119100	137300	.15
475		24	4	25	1.00	83857	83857	137300	.6
500	20		4	25	1.00	84482	84482	90600	.07
525	20		4	20	1.00	85416	85416	102400	.2
545	20		4	20	1.00	85781	85781	102400	.2
565	20		4		1.00	85963	85963	102400	.2

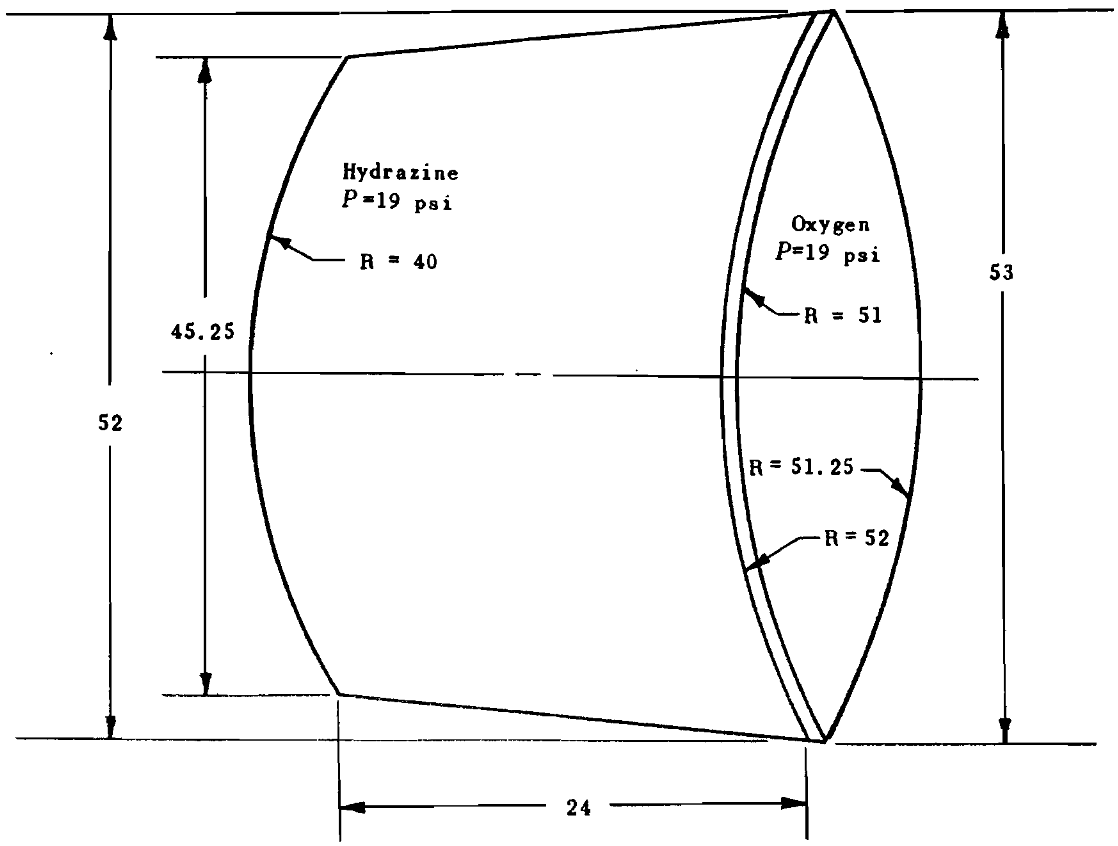
TENSION LOADS

O.K. by inspection

TANKS

STAGE III

MAIN PROPELLANT TANKS



Hydrazine Tank - Top

$$f_t = \frac{Pr}{2t} = \frac{19 \times 40}{2 \times .020} = 19000 \text{ psi}$$

F_{tu} = 38200 psi Material - 24 STAL

M.S. = 1.0

Hydrazine Tank - Wall

$$f_t = \frac{Pr}{t} = \frac{19 \times 25}{.020} = 23700 \text{ psi}$$

F_{tu} = 38200 psi Material - 24 STAL

M.S. = .6

Hydrazine Tank - Bottom

$$f_t = \frac{Pr}{2t} = \frac{19 \times 52}{2 \times .020} = 24700 \text{ psi}$$

F_{tu} = 38200 psi Material - 24 STAL

M.S. = .5

Oxygen Tank - Top

$$f_t = \frac{Pr}{2t} = \frac{19 \times 51}{2 \times .020} = 24200 \text{ psi}$$

F_{tu} = 38200 psi   Material - 24 STAL

M.S. = .6

Oxygen Tank - Bottom

$$f_t = \frac{Pr}{2t} = \frac{19 \times 51.25}{2 \times .020} = 24350 \text{ psi}$$

F_{tu} = 38200 psi   Material - 24 STAL

M.S. = .6

AUXILIARY PROPELLANT TANKS

Hydrogen Peroxide Tank (Spherical)

$$f_t = \frac{Pr}{2t} = \frac{400 \times 3.6}{2 \times .020} = 36000 \text{ psi}$$

F_{tu} = 38200 psi   Material - 24 STAL

M.S. = .06

Hydrazine Tank (Spherical)

$$f_t = \frac{Pr}{2t} = \frac{222 \times 9.45}{2 \times .040} = 26200 \text{ psi}$$

F_{tu} = 26300 psi   Material - 24 STAL

M.S. = .004

Oxygen Tank (Spherical)

$$f_t = \frac{Pr}{2t} = \frac{192 \times 7.6}{2 \times .020} = 36500 \text{ psi}$$

F_{tu} = 38200 psi   Material - 24 STAL

M.S. = .04

Helium Tanks (Spherical)

Tank A

$$f_t = \frac{Pr}{2t} = \frac{3000 \times 4.45}{2 \times .078} = 85700 \text{ psi}$$

F_{tu} = 89500 psi   Material - 18-8 Stainless Steel 1/2 Hard

M.S. = .04

Tank B

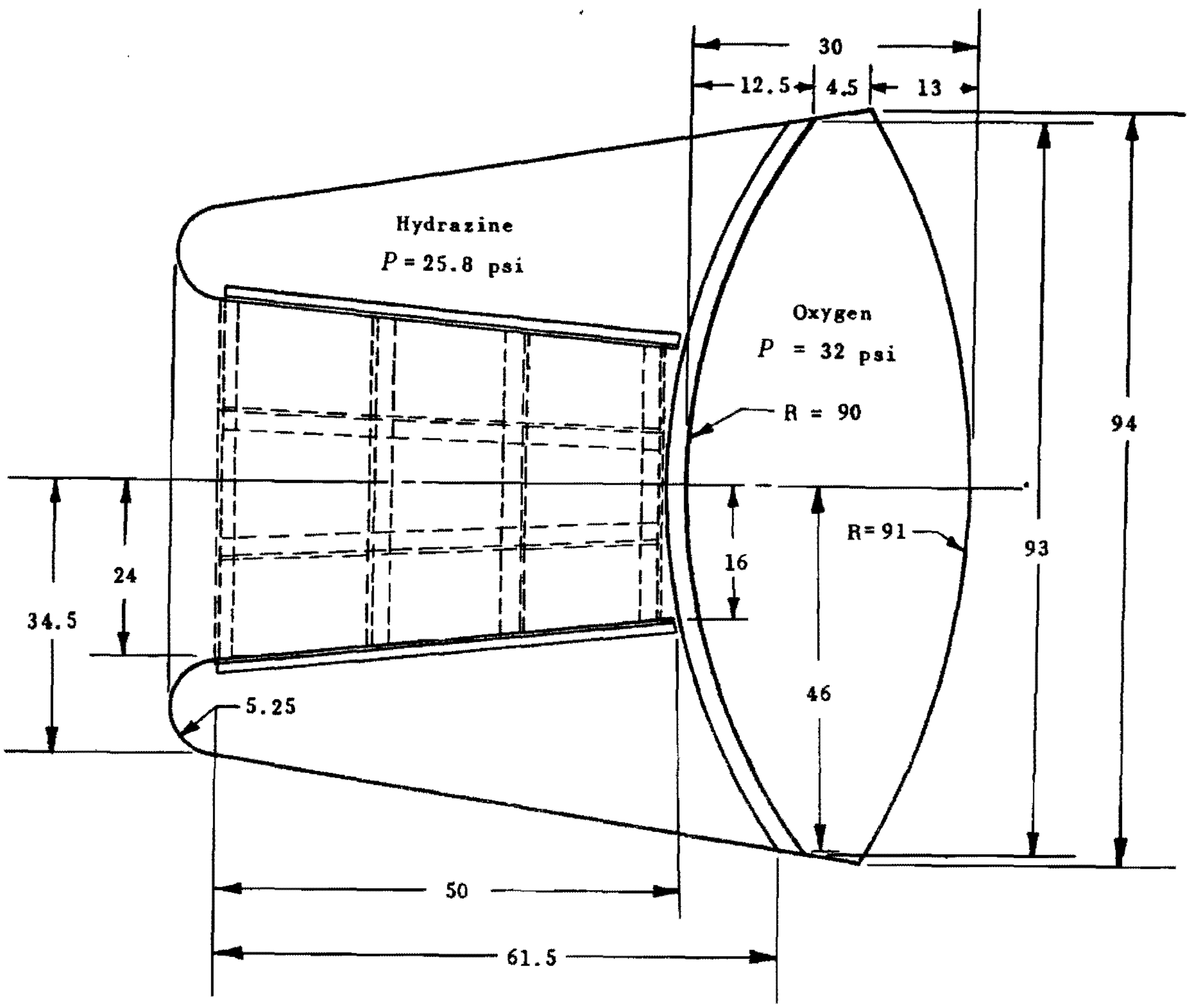
$$f_t = \frac{Pr}{2t} = \frac{3000 \times 5.15}{2 \times .093} = 83100 \text{ psi}$$

F_{tu} = 89500 psi   Material - 18-8 Stainless Steel 1/2 Hard

M.S. = .08

STAGE II

MAIN PROPELLANT TANKS



Hydrazine Tank - Top

$$f_t = \frac{Pr}{2t} = \frac{25.9 \times 5.25}{2 \times .020} = 34000 \text{ psi}$$

F_{tu} = 38200 psi Material - 24 STAL

M.S. = .1

Hydrazine Tank - Outer Wall

$$f_t = \frac{Pr}{t} = \frac{25.8 \times 40.5}{.051} = 20500 \text{ psi}$$

F_{tu} = 23500 psi Material - 24 STAL

M.S. = .1

Hydrazine Tank - Bottom

$$f_t = \frac{Pr}{2t} = \frac{6.2 \times 90}{2 \times .020} = 14000 \text{ psi}$$

F_{tu} = 38200 psi   Material - 24 STAL

M.S. = high

Oxygen Tank - Top

$$f_t = \frac{Pr}{2t} = \frac{32 \times 90}{2 \times .020} = 72000 \text{ psi}$$

F_{tu} = 89500 psi   Material - 18-8 Stainless Steel 1/2 Hard

M.S. = .2

Oxygen Tank - Bottom

$$f_t = \frac{Pr}{2t} = \frac{32 \times 91}{2 \times .020} = 72800 \text{ psi}$$

F_{tu} = 89500 psi   Material - 18-8 Stainless Steel 1/2 Hard

M.S. = .2

Oxygen Tank - Wall

$$f_t = \frac{Pr}{t} = \frac{32 \times 46.75}{.020} = 74800 \text{ psi}$$

F_{tu} = 89500 psi   Material - 18-8 Stainless Steel 1/2 Hard

M.S. = .2

AUXILIARY PROPELLANT TANKS

Hydrogen Peroxide Tank (Spherical)

$$f_t = \frac{Pr}{2t} = \frac{400 \times 4.7}{2 \times .040} = 23500 \text{ psi}$$

F_{tu} = 26300 psi   Material 24 STAL

M.S. = .1

Helium Tanks (Spherical)

Tank A

$$f_t = \frac{Pr}{2t} = \frac{3000 \times 7.33}{2 \times .125} = 87000 \text{ psi}$$

F_{tu} = 89500 psi   Material - 18-8 Stainless Steel 1/2 Hard

M.S. = .03

Tank B

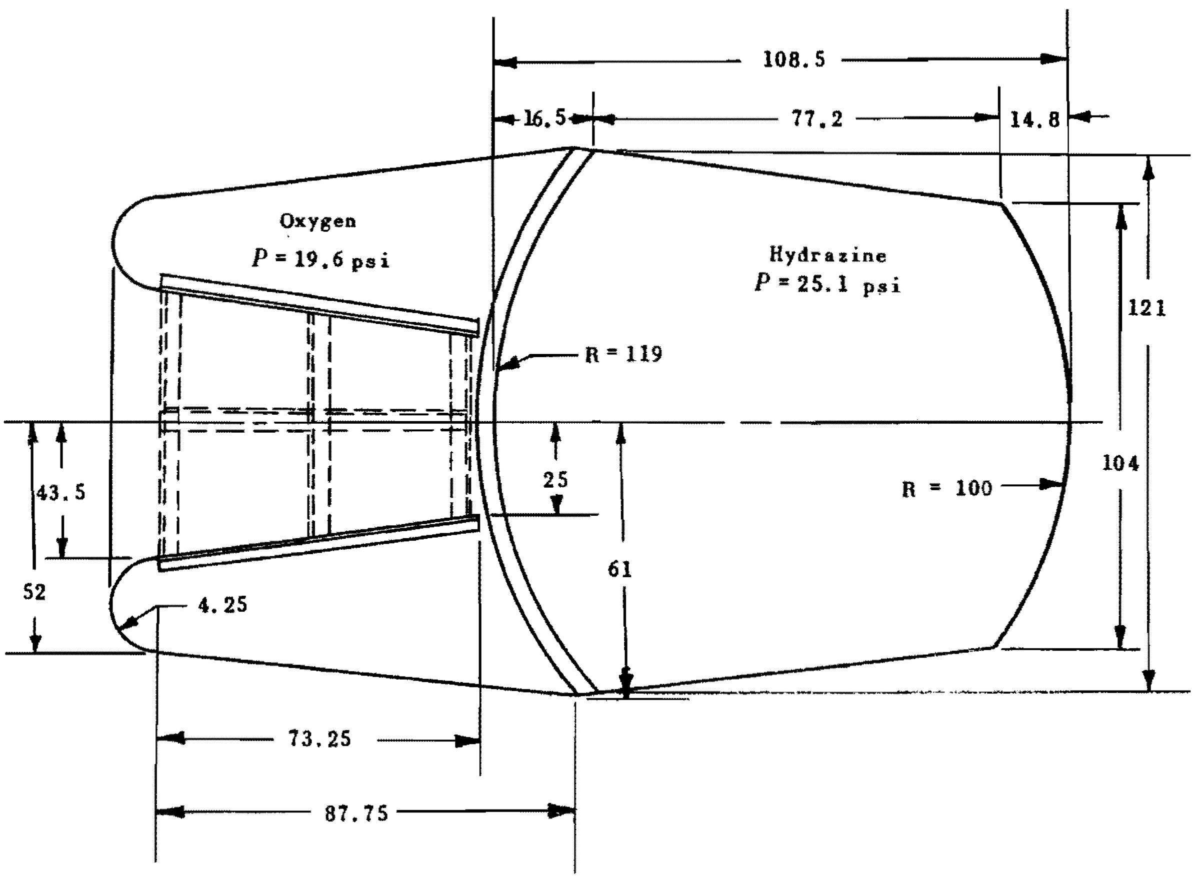
$$f_t = \frac{Pr}{2t} = \frac{3000 \times 3.37}{2 \times .062} = 81500 \text{ psi}$$

F_{tu} = 89500 psi   Material - 18-8 Stainless Steel 1/2 Hard

M.S. = .1

STAGE I

MAIN PROPELLANT TANKS



Oxygen Tank - Top

$$f_t = \frac{Pr}{2t} = \frac{19.6 \times 4.25}{2 \times .020} = 20800 \text{ psi}$$

F_{tu} = 38200 psi Material - 24 STAL

M.S. = .8

Oxygen Tank - Bottom

$$f_t = \frac{Pr}{2t} = \frac{5.5 \times 119}{2 \times .020} = 16400 \text{ psi}$$

F_{tu} = 38200 psi Material - 24 STAL

M.S. = high

Oxygen Tank - Outer Wall

$$f_t = \frac{Pr}{t} = \frac{19.6 \times 56.5}{.051} = 21700 \text{ psi}$$

F_{tu} = 23500 psi Material - 24 STAL

M.S. = .08

Hydrazine Tank - Top

$$f_t = \frac{Pr}{2t} = \frac{25.1 \times 119}{2 \times .020} = 74700 \text{ psi}$$

F_{tu} = 89500 psi Material - 18-8 Stainless Steel 1/2 Hard

M.S. = .2

Hydrazine Tank - Bottom

$$f_t = \frac{Pr}{2t} = \frac{25.1 \times 100}{2 \times .020} = 62800 \text{ psi}$$

F_{tu} = 89500 psi Material - 18-8 Stainless Steel 1/2 Hard

M.S. = .4

Hydrazine Tank - Wall

$$f_t = \frac{Pr}{t} = \frac{25.1 \times 56}{.020} = 70200 \text{ psi}$$

F_{tu} = 89500 psi Material - 18-8 Stainless Steel 1/2 Hard

M.S. = .3

AUXILIARY PROPELLANT TANKS

Hydrogen Peroxide Tanks (Spherical) - 2 Required

$$f_t = \frac{Pr}{2t} = \frac{400 \times 10.75}{2 \times .025} = 86000 \text{ psi}$$

F_{tu} = 89500 psi Material - 18-8 Stainless Steel 1/2 Hard

M.S. = .04

Helium Tanks (Spherical)

Tank A

$$f_t = \frac{Pr}{2t} = \frac{3000 \times 10.45}{2 \times .188} = 83400 \text{ psi}$$

F_{tu} = 89500 psi Material - 18-8 Stainless Steel 1/2 Hard

M.S. = .07

Tank B

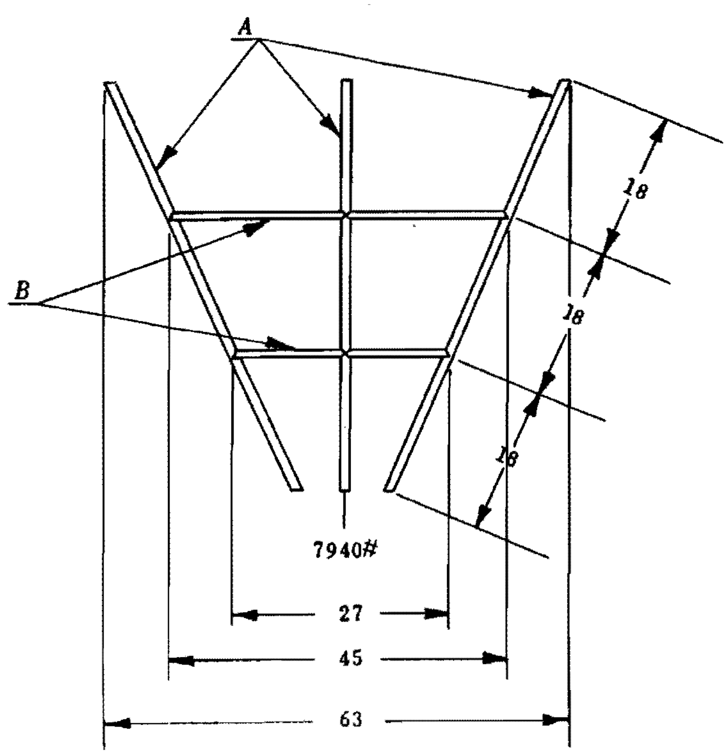
$$f_t = \frac{Pr}{2t} = \frac{3000 \times 6.94}{2 \times .125} = 83200 \text{ psi}$$

F_{tu} = 89500 psi Material - 18-8 Stainless Steel 1/2 Hard

M.S. = .07

ROCKET MOTOR SUPPORT STRUCTURE

STAGE III



Thrust per column = 2210 lbs

Column length = 18 inches

A O D = .625 inches t = .028 inches

B O D = .375 inches t = .028 inches

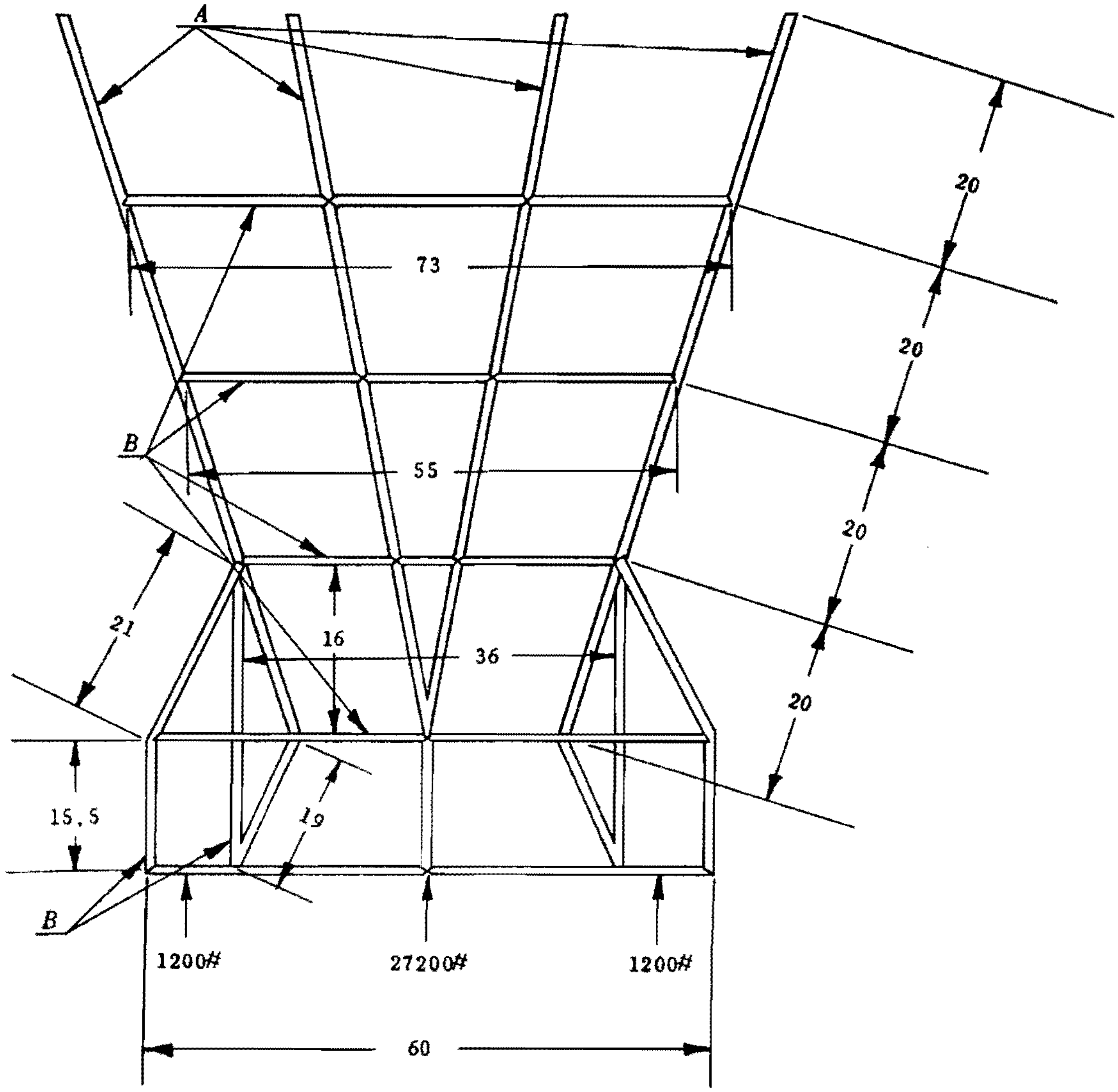
f_c = 53300 psi

F_{cy} = 85000 psi

Material - X4130 Steel Tube

M.S. = .6

STAGE II



Thrust per column = 4490 lbs

Column length = 20 inches

A O D = .750 inches t = .028 inches

B O D = .500 inches t = .028 inches

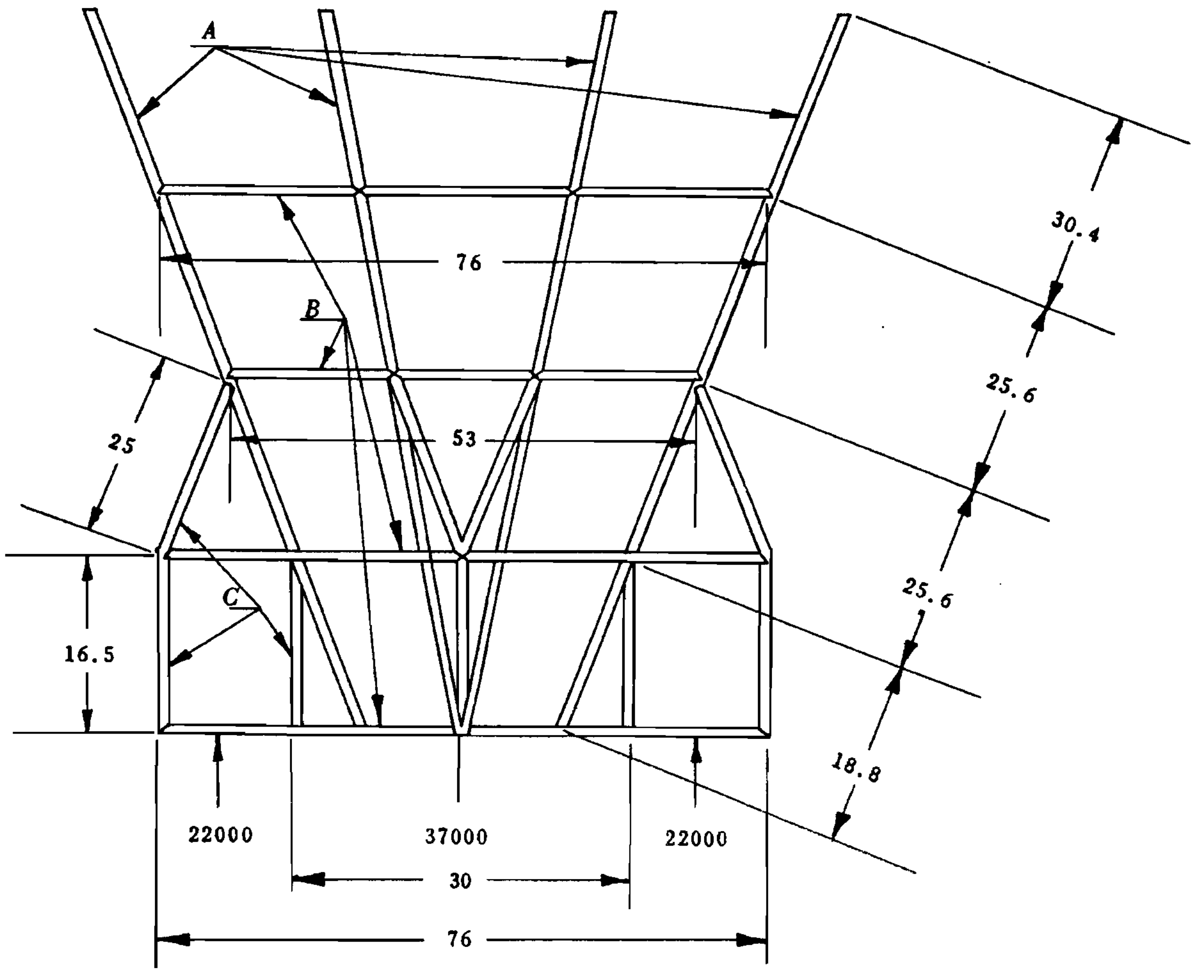
f_c = 70700 psi

F_{cy} = 85000 psi

Material - X4130 Steel Tube

M.S. = .2

STAGE I



Thrust per column = 17850 lbs

Column length = 30.4 inches

A	O D	= 1.50 inches	t = .049 inches
B	O D	= .750 inches	t = .035 inches
C	O D	= 1.375 inches	t = .035 inches

f_c = 80000 psi

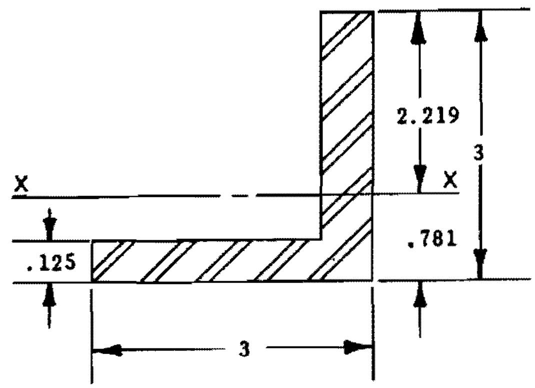
F_{cy} = 85000 psi

Material - X4130 Steel Tube

M.S. = .06

THRUST RING

STAGE III



$$k = \frac{tE_f}{\ell} = \frac{.020 \times 29 \times 10^6}{18} = 32200 \text{ psi}$$

$$\beta = \sqrt[4]{\frac{k}{4EI}} = \sqrt[4]{\frac{32200}{4 \times 29 \times 10^6 \times .64}} = .1455 \text{ inches}$$

$$M_{max} = \frac{P}{4\beta} = \frac{1985}{4 \times .1455} = 3410 \text{ inch-lbs}$$

$$f_c = \frac{M_{max}Y}{I} = \frac{3410 \times 2.219}{.64} = 11800 \text{ psi}$$

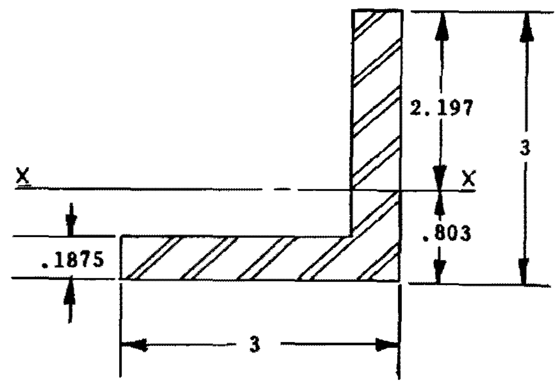
$$b/t = \frac{2.875}{.125} = 23$$

F_{cr} = 22000 psi Buckling outstanding leg

Material - 18-8 Stainless Steel 1/2 Hard

M.S. = .9

STAGE II



$$k = \frac{tE_f}{\ell} = \frac{.02 \times 29 \times 10^6}{20} = 29000 \text{ psi}$$

$$\beta = \sqrt[4]{\frac{k}{4EI}} = \sqrt[4]{\frac{29000}{4 \times 29 \times 10^6 \times .931}} = .126 \text{ inches}$$

$$M_{max} = \frac{P}{4\beta} = \frac{4000}{4 \times .126} = 7940 \text{ inch-lbs}$$

$$f_c = \frac{M_{max} Y}{I} = \frac{7940 \times 2.197}{.931} = 18700 \text{ psi}$$

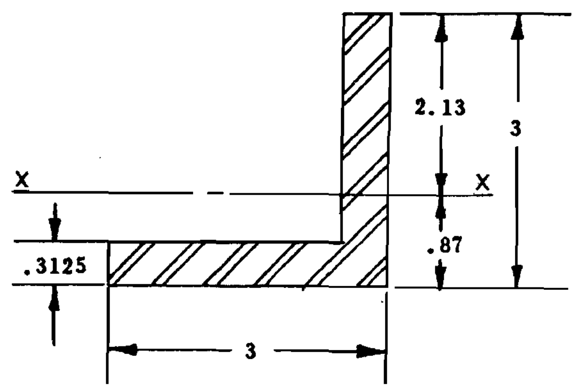
$$b/t = \frac{2.813}{.1875} = 15$$

F_{cr} = 42000 psi Buckling outstanding leg

Material - 18-8 Stainless Steel 1/2 Hard

M.S. = high

STAGE I



$$k = \frac{tE_f}{\ell} = \frac{.02 \times 29 \times 10^6}{30.4} = 19200 \text{ psi}$$

$$\beta = \sqrt[4]{\frac{k}{4EI}} = \sqrt[4]{\frac{19200}{4 \times 29 \times 10^6 \times 1.5}} = .1025 \text{ inches}$$

$$M_{max} = \frac{P}{4\beta} = \frac{15625}{4 \times .1025} = 39100 \text{ inch-lbs}$$

$$f_c = \frac{M_{max} Y}{I} = \frac{39100 \times 2.13}{1.5} = 55500 \text{ psi}$$

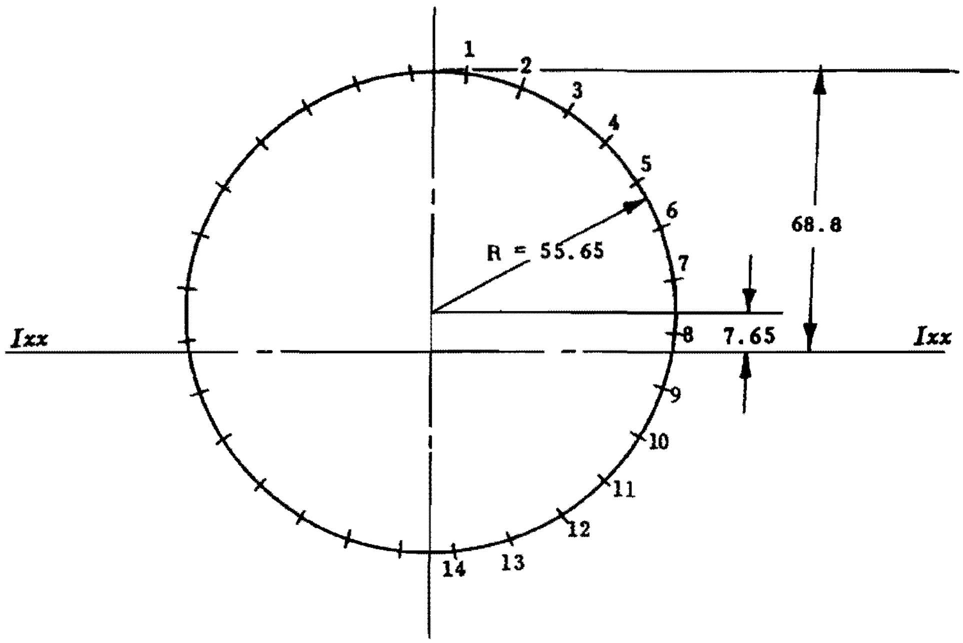
$$F_{cr} = 75000 \text{ psi}$$

Material - 18-8 Stainless Steel 1/2 Hard

M.S. = .35

BENDING OF THE ROCKET DUE TO TILT

STATION 430



I_{xx} = 5459.7 in.^{4}

M = 632000 in.-lbs (Fig. 27)

Compression in longitudinal stiffener #1 due to bending.

$$f_c = \frac{632000 \times 68.8}{2 \times 5459.7} = 3975 \text{ psi}$$

Compression in longitudinal stiffener #1 due to thrust.

f_c = 55000 psi

Therefore: Σf_c = 58975 psi

From Table 17

$$F_c = \frac{4890}{.0271} = 67500 \text{ psi}$$

M.S. = .1

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University of Minnesota Minneapolis, Minnesota Attn: Dr. Akerman	Inspector of Naval Material Federal Bldg. Milwaukee 2, Wis.	BUORD
Aerojet Engineering Corp. Azusa, California Attn: K.F. Mundt	Bureau of Aeronautics Rep. 15 South Raymond Street Pasadena, California	BUAER
Marquardt Aircraft Co. Venice, California Attn: Dr. R. E. Marquardt	Bureau of Aeronautics Rep. 15 South Raymond Street Pasadena, California	BUAER

(2) GUIDANCE & CONTROL

Belmont Radio Corporation 5921 West Dickens Avenue Chicago 29, Illinois Attn: Mr. Harold C. Mattes		AAF
Bendix Aviation Corp. Eclipse-Pioneer Division Teterboro, New Jersey Attn: Mr. R. C. Sylvander	Bureau of Aeronautics Resident Representative Bendix Aviation Corp. Teterboro, New Jersey	BUAER
Bendix Aviation Corp. Pacific Division, SPD West North Hollywood, Calif.	Development Contract Officer Bendix Aviation Corp. 11600 Sherman Way North Hollywood, California	BUORD
Bendix Aviation Radio Division East Joppa Road Baltimore 4, Maryland Attn: Mr. J. W. Hammond		AAF
Buehler and Company 1607 Howard Street Chicago 26, Illinois Attn: Mr. Jack M. Roehn		
Commanding General Army Air Forces Pentagon Washington 25, D.C. Attn: AC/AS-4, DRE-2F		

D. COMPONENT CONTRACTORS (Cont'd)

(2) GUIDANCE & CONTROL

CONTRACTOR	TRANSMITTED VIA	COGNIZANT AGENCY
Consolidated-Vultee Aircraft Corporation San Diego, California Attn: Mr. C. J. Breitwieser	Bureau of Aeronautics Representative, Consolidated-Vultee Aircraft Corp. San Diego, California	BUAER
Cornell University Ithaca, New York Attn: Mr. William C. Ballard, Jr.		AAF
Director, U.S. Navy Electronics Laboratory, San Diego, California		NAVY
Electro-Mechanical Research Ridge Field, Connecticut Attn: Mr. Charles B. Aiken		AAF
Farnsworth Television and Radio Co. Fort Wayne, Indiana Attn: Mr. J. D. Schantz	DCO, Applied Physics Laboratory Johns Hopkins University 8621 Georgia Avenue, Silver Spring, Maryland	BUORD
Federal Telephone and Radio Corp. 200 Mt. Pleasant Avenue Newark 4, New Jersey Attn: Mr. E. N. Wendell		AAF
Galvin Manufacturing Corp. 4545 Augusta Blvd. Chicago 5, Illinois Attn: Mr. G. R. MacDonald		AAF
G. M. Giannini and Co., Inc. 285 West Colorado St. Pasadena, California	Bureau of Aeronautics Rep. 15 South Raymond St. Pasadena, California	BUAER
Gilfillan Corp. 1815-1849 Venice Blvd. Los Angeles 6, California Attn: Mr. G. H. Miles		AAF
Hillyer Engineering Co. New York, New York Attn: Mr. Curtiss Hillyer	Inspector of Naval Material 90 Church Street New York 7, New York	BUAER
Kearfott Engineering Co. New York, New York Attn: Mr. W. A. Reichel	Inspector of Naval Material 90 Church Street New York 7, New York	BUAER
Lear Incorporated 110 Iona Avenue, N.W. Grand Rapids 2, Michigan Attn: Mr. R.M. Mock		AAF
Manufacturers Machine & Tool Co. 320 Washington Street Mt. Vernon, N.Y. Attn: Mr. L. Kenneth Mayer, Comptroller		AAF
Minneapolis-Honeywell Mfgr. Co. 2753 Fourth Avenue Minneapolis 8, Minnesota Attn: Mr. W. J. McGoldrick, Vice-President		AAF
Ohio State University Research Foundation Columbus, Ohio Attn: Mr. Thomas E. Davis, Staff Assistant		AAF

D. COMPONENT CONTRACTORS (Cont'd)

(2) GUIDANCE & CONTROL

CONTRACTOR	TRANSMITTED VIA	COGNIZANT AGENCY
Haller, Raymond & Brown P.O. Box 342 State College, Pennsylvania Attn: Dr. R. C. Raymond, Pres.		AAF
Office of Chief Signal Officer Engineering & Technical Services, Engineering Division Pentagon Washington 25, D.C.		ORD DEPT
Raytron, Inc. 209 E. Washington Avenue Jackson, Michigan Attn: Mr. John R. Gelzer, Vice-Pres.		AAF
L. N. Schwein Engineering Co. 5736 Washington Blvd. Los Angeles 16, California Attn: L.N. Schwein, General Partner		AAF
Senior Naval Liaison Officer U.S. Naval Electronic Liaison Office Signal Corps, Engineering Laboratory Fort Monmouth, New Jersey		NAVY
Servo Corporation of America Huntington, L.I., New York	Inspector of Naval Material 90 Church Street New York 7, New York	BUAER
Square D Co. Kollsman Instrument Division Elmhurst, New York Attn: Mr. V. E. Carbonara	Bureau of Aeronautics Rep. 90 Church Street New York 7, New York	BUAER
Stromberg-Carlson Company Rochester, New York Attn: Mr. L.L. Spencer, Vice-Pres.		AAF
Submarine Signal Company Boston, Massachusetts Attn: Mr. Edgar Horton	Development Contract Officer Massachusetts Institute of Technology Cambridge 39, Massachusetts	BUORD
Summers Gyroscope Co. 1100 Colorado Avenue Santa Monica, California Attn: Mr. Tom Summers, Jr.		AAF
Sylvania Electric Products Inc. Flushing, Long Island, N.Y. Attn: Dr. Robert Bowie	Inspector of Naval Material 90 Church Street New York 7, New York	BUORD
University of Illinois Urbana, Illinois Attn: Mr. H. E. Cunningham, Sec.		AAF
University of Pennsylvania Moore School of Electrical Engr. Philadelphia, Pa.	Commanding Officer Naval Aircraft Modification Unit Johnsville, Pa.	BUAER
University of Pittsburgh Pittsburgh, Pennsylvania Attn: Mr. E. A. Holbrook, Dean		AAF
University of Virginia Physics Department Charlottesville, Virginia Attn: Dr. J. W. Beams	Development Contract Officer University of Virginia Charlottesville, Virginia	BUORD

D. COMPONENT CONTRACTORS (Cont'd)

(2) GUIDANCE & CONTROL

CONTRACTOR	TRANSMITTED VIA	COGNIZANT AGENCY
Washington University Research Foundation 8135 Forsythe Blvd., Clayton 5, Missouri Attn: Dr. R. G. Spencer		AAF
Westinghouse Electric Corp. Springfield, Massachusetts Attn: J.K.B. Hare, Vice-Pres. (Dayton Office)		AAF
Director of Specialty Products Development Whippany Radio Laboratory Whippany, N.J. Attn: Mr. M.H. Cook		ORD DEPT
Zenith Radio Corporation Chicago, Illinois Attn: Hugh Robertson, Executive Vice-Pres.		AAF

(3) PROPULSION

CONTRACTOR	TRANSMITTED VIA	COGNIZANT AGENCY
Aerojet Engineering Corp. Azusa, California Attn: K.F. Mundt	Bureau of Aeronautics Rep. 15 South Raymond Street Pasadena, California	BUAER
Armour Research Foundation Technical Center, Chicago 16, Illinois Attn: Mr. W. A. Casler		ORD DEPT
Arthur D. Little, Inc. 30 Memorial Drive, Cambridge, Mass. Attn: Mr. Helge Holst		ORD DEPT
Battelle Memorial Institute 505 King Avenue Columbus 1, Ohio Attn: Dr. B. D. Thomas		AAF & BUAER
Bendix Aviation Corp. Pacific Division, SPD West N. Hollywood, Calif.	Development Contract Officer Bendix Aviation Corp. 11600 Sherman Way N. Hollywood, Calif.	BUORD
Bendix Products Division Bendix Aviation Corporation 401 Bendix Drive South Bend 20, Indiana Attn: Mr. Frank C. Mock		AAF BUORD
Commanding General Army Air Forces Pentagon Washington 25, D.C. Attn: AC/AS-4 DRE-2E		AAF
Commanding General Air Materiel Command Wright Field Dayton, Ohio Attn: TSEPP-4B(2) TSEPP-4A(1) TSEPP-5A(1) TSEPP-5C(1) TSORE-(1)		
Commanding Officer Picatinny Arsenal Dover, New Jersey Attn: Technical Division		ORD DEPT

D. COMPONENT CONTRACTORS (Cont'd)

(3) PROPULSION

CONTRACTOR	TRANSMITTED VIA	COGNIZANT AGENCY
Commanding Officer Watertown Arsenal Watertown 72, Massachusetts. Attn: Laboratory.		ORD DEPT
Continental Aviation and Engr. Corp. Detroit, Michigan	Bureau of Aeronautics Rep. 11111 French Road Detroit 5, Michigan	BUAER & AAF
Curtiss-Wright Corporation Propeller Division Caldwell, New Jersey Attn: Mr. C. W. Chillson		AAF
Experiment, Incorporated Richmond, Virginia Attn: Dr. J. W. Mullen, II	Development Contract Officer P.O. Box 1-T Richmond 2, Virginia	BUORD
Fairchild Airplane & Engine Co. Ranger Aircraft Engines Div. Farmingdale, L.I., New York	Bureau of Aeronautics Rep. Bethpage, L.I., N.Y.	BUAER
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. 285 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