ARMY BALLISTIC MISSILE AGENCY

REDSTONE ARSENAL, ALABAMA

4 November 1958

Report No. DSP-TN-13-58
Copy No. 4

James P. Gardner

II.1 III.4

COMMENTS ON PROBLEMS RELATING TO THE LUNAR LANDING VEHICLE

DEVELOPMENT OPERATIONS DIVISION

ABMA FORM 485, REV 1 MAR 58

ARMY—JAMCO REPRODUCTIONS HUNTSVILLE, ALABAMA

4 November 1958 Report No. DSP-TN-13-58

COMMENTS ON PROBLEMS

RELATING TO THE LUNAR LANDING

VEHICLE

by

James P. Gardner, Harry O. Ruppe and Warren H. Straly

FUTURE PROJECTS DESIGN BRANCH
STRUCTURES AND MECHANICS LABORATORY

Arranged and Processed
by
REPORTS AND PUBLICATIONS SECTION
WEAPON SYSTEM INFORMATION OFFICE

ABSTRACT

This technical note concerns some of the problems encountered with the landing of a payload on the moon. The main problem areas such as guidance, velocity control and impact considerations are discussed. Although no final conclusions or designs are intended, it is hoped that the material presented will serve as a guide for future detailed work.

TABLE OF CONTENTS

LIST OF ILLUSTRATIONS

LIST OF ILLUSTRATIONS (Continued)

SECTION I. SUMMARY

It is the intention of this report to furnish ideas, not details, of schemes for guidance and landing of a vehicle on the moon. The ideas presented are not the only ones, but are given as guides or nuclei for other ideas. Some of these schemes may have no practical value but may lead into ones which do have.

Conclusions reached are as follows:

1. It appears that multi-stage rocket retardation will be necessary to effect a soft enough landing.

2. A multi-stage or coarse and vernier guidance system could be used to advantage.

It is recommended that further consideration be given to the allowable impact velocity, the guidance and retardation needed to effect that velocity and the vehicle design for which above conditions are criteria.

SECTION II. INTRODUCTION

In this technical note, some fundamentals of the so-called "soft" lunar lander are brought forward. This is a part of a study conducted within ABMA and coordinated by Electro-Mechanical Engineering Branch, Guidance and Control Laboratory. The more important results are:

1. The guidance difficulties to insure a soft landing are rather severe.

2. The system for taking up the final impact shock presents difficulties, since:

   a. The residual impact velocity (magnitude and direction) is not very well-known. This would then call for an omnidirectional protection system which hampers payload utilization.

   b. Only "educated guesses" are available concerning the properties of the lunar surface.

   c. Since the lunar landscape is far from being smooth, there is a danger of ricochetting if by chance impacting occurrs on a slope.

3. The 500-pound lander seems to be near the lower limit where a useful payload can be carried. For very large vehicles (100 tons) the useful payload may go up to about 25% of the approaching weight.

4. These data seem to show that a manned circumlunar trip or manned lunar satellite could be launched from the earth's surface, but some other technique will have to be used for a manned lunar landing with the present state of the art.

Assume that a simple manned, 5-ton vehicle is to land back on earth (See DSP-TM-5-58, Report on the Super JUPITER Junior, p. 30). The returning vehicle has upon departure from the moon a weight of about 20 tons. The vehicle that approached the moon for the soft landing weighed approximately 80 tons. If the launch was from the earth's surface, then the initial weight of the total vehicle would have been about 10,000 tons. In contrast, for a circumlunar flight including return to earth, less than 1,000 tons would suffice. Fifteen hundred tons is enough for a lunar satellite including the return to earth.

5. Presently liquid propellant braking motors appear more promising than solid propellant braking motors for very large vehicles with payloads of at least several tons.

SECTION III. DISCUSSION

1. Fundamental Considerations

a. Velocity requirement for braking. From "Satellite Technology and Space Navigation," DSP-TN-9-58, 9 September 1958, H. Ruppe, Chapter 5.2 the following is taken:

The minimum impact velocity of an unbraked earth-to-moon probe may be about 2,320 m/sec, but for a practical 2.3-day-transfer the impact velocity goes up to about 2,750 m/sec.

The gravity loss for a vertical descent is about gT, where g is 0.164 g_o, g_o is ≈ 9.8 m/sec^2, and T is the burning time of the rocket. So the ideal velocity requirement is:

$$V_i = 2750 + 0.164\ g_o T \quad [\text{m/sec}]$$

We can write:

$$V_i = \bar{b}T = n\ g_o T \quad (\bar{b} = \text{mean acceleration})$$

Then:

$$V_i = \frac{2750}{1 - \frac{0.164}{n}} \approx 2750 + \frac{500}{n} \quad [\text{m/sec}]$$

b. Components of vehicle. The vehicle approaching the moon consists of the following components.

(1) Payload. (m_1 = payload mass, w_1 = payload weight) Here we find the true active payload (measuring device, recorders, etc.), then the transmitter and power supply, and structure to carry those elements.

(2) Shock absorption system. (W_2 = shock absorption weight) The relative weight of the shock absorption system will decrease when better guidance systems are used. The result will be softer landings.

(3) Dry motor. (W_3 = dry motor weight) In the following, the masses are called: $m_A = m_1 + m_2 + m_4$.

Solid Propellant: Here the mass of the dry motor is between 8 and 13% of the mass of the fuel used, being 13% for smaller (200 kg) and 8% for larger (10 tons) units. As an average figure, 10% will be used.

Liquid Propellant: The dry weight of the motor and feed system may be 1/60 of the thrust. The thrust equals n · mean vehicle weight, which can be estimated to be 86% of the total fuel weight. Therefore,

$$m_3 = 0.1 \frac{n}{7} m_6$$

We can combine the results for solid and liquid motors by writing

$$m_3 = 0.1 \left(\frac{n}{7} \right)^{\alpha} m_6$$

Where: $\alpha = 0$ means: solid propellant
$\alpha = 1$ means: liquid propellant

(4) Control equipment. (W_4 = control equipment weight) Attitude control equipment, horizon seeker, etc. constitute this part. The radar altimeter which triggers the braking motor is excluded.

(5) Fuel residuals. (W_5 = fuel residual weight) The fuel residuals shall be assumed to be 2% of the fuel used.

(6) Fuel used. (W_6 = fuel weight) This is the fuel actually used for braking the descent.

(7) Tankage and vehicle structure. (W_{61} = tankage and vehicle weight) This is a seperate item in liquid propellant vehicles only. Therefore, it is assumed that:

$$m_{61} = \alpha \frac{5}{100} m_6$$

An interesting conclusion can readily be drawn, the weight of the liquid propellant dry system is:

$$g_o \, m_{\ell} = g_o \left(0.1 \frac{n}{7} + \frac{5}{100}\right) m_{6\ell}$$

The solid propellant dry system is:

$$g_o\, m_s = g_o\, (0.1)\, m_{6_s}$$

which gives:

$$\frac{g_o\, m_\ell}{g_o\, m_s} = \left(\frac{n}{7} + 0.5\right) \frac{m_{6_\ell}}{m_{6_s}}$$

Therefore, for larger n-values the solid propellant system is lighter, and for smaller n-values the liquid system is lighter. In practical designs this result has been well known since usually in the same vehicle a solid propellant motor is used at a higher thrust level than a liquid propellant motor for the same ideal velocity requirement. (See Appendix B.)

(8) Radar. (w_7 = radar weight) This is the radar which triggers the braking rocket.

$$g_o\, w_7 = 3 + \frac{1}{5} \left(\frac{s}{10}\right)^4 \quad [\text{kg}] \text{ , where}$$

s = range in km.

There is

$$S = \frac{V^2}{2\bar{b}} = \frac{V^2}{2 g_o n} \approx \frac{400}{n} \quad [\text{km}]$$

which results in $g_o\, m_7 = 3 + \frac{1}{200} \left(\frac{100}{n}\right)^4 \quad [\text{kg}]$

The radar is jettisoned from the vehicle the instant the braking motor starts firing. This saves some weight, which is especially important for the lighter vehicles.

Vehicle mass approaching the moon: $M_1 = \sum_{1}^{7} m_i$

Vehicle mass at ignition: $M = M_1 - M_7$

Vehicle cutoff mass: $M - m_6$

c. The optimum deceleration. From the basic rocket equation, there is $\frac{M}{m} = e^{\frac{V_i}{c}}$, where $c = I_{sp} \cdot g_o$.

Detailed:

$$\frac{M_1 - \frac{1}{g_o}\left[3 + \frac{1}{200}\left(\frac{100}{n}\right)^4\right]}{m_A + \left[0.1 \left(\frac{n}{7}\right)^\alpha + \frac{2+5\alpha}{100}\right] \dfrac{M_1 - \frac{1}{g_o}\left[3 + \frac{1}{200}\left(\frac{100}{n}\right)^4\right] - m_A}{1 + 0.1\left(\frac{n}{7}\right)^\alpha + \frac{2+5\alpha}{100}}} = e^{\frac{2750}{c}}\, e^{\frac{500}{n \cdot c}}$$

Solid propellant: $\alpha = 0$, $I_{sp} = 260$ sec, $c = 2550$ m/sec

Liquid propellant: $\alpha = 1$, $I_{sp} = 300$ sec, $c = 2950$ m/sec with $e^{\frac{500}{nc}} = 1 + \frac{500}{nc}$, $M_1 g_o = W_1$, $m_A g_o = W_A$, $x = \frac{100}{n}$ can be written:

$$W_A = (W_1 - 3 - \frac{1}{200} x^4) \left\{ 1 - (1 - \frac{1 - \frac{x}{c75}}{e^{2750/c}}) \left[1 + 0.1 \left(\frac{14}{x}\right)^{\alpha} + \frac{2 + 5\alpha}{100} \right] \right\}$$

$e^{\frac{2750}{2550}} = 2.942$, $e^{\frac{2750}{2950}} = 2.52$

For a solid propellant:

$$W_A = 0.26 \, (W_1 - 3 - \frac{x^4}{200}) \, (1 - \frac{x}{348})$$

This shows clearly, that x should be as small as possible in order to make W_A as large as possible for a given W_1.

If we assume that n = 20 is an upper limit, then x = 5 and

$$W_A = 0.256 \, W_1 - 1.6$$

It is rather difficult to say, what constitutes a minimum useful W_A (weight-wise). Tentatively, we can formulate the following list:

Active payload	5 kg
Transmitter	5 kg
Power supply (batteries)	10 kg
Structure to carry payload	5 kg
Shock absorption system	15 kg
Control equipment	20 kg
W_A min	60 kg
This leads then to: W_1 min =	241 kg

This casts doubt upon the practicality of a 500-pound (225 kg) - lunar landing vehicle.

For a liquid propellant motor:

$$W_A = (W_1 - 3 - \frac{x^4}{200}) \left[0.354 - \left(\frac{x}{1390} + \frac{0.845}{x}\right) \right]$$

Differentiating and putting $\frac{\alpha W_A}{\alpha x} = 0$ gives: $W_1 - 3 = \frac{x^4}{40} \cdot \frac{394x - 705 - x^2}{1175 - x^2}$

Now, the following table can easily be calculated:

TABLE I

Optimum Decelerations vs Payload on Way

X_{opt}	n_{opt}	W_1	$\sqrt[5]{140W_1}$
4	25	8 kg	4.08
5	20-	17	4.73
10	10	730	10.05
20	5	35.10^3	21.79
30	3.33	753.10^3	37.40
34.25	2.92	∞	∞

So we see that, approximately, $X_{opt} = \sqrt[5]{140W_1}$ for $W_1 < 50.10^3$ kg; for $W_1 > 50.10^3$ kg there is, approximately, $X_{opt} = 25$.

Those results are interesting since they show that for large lunar landing vehicles the optimum mean deceleration comes down to values which might be acceptable even for manned vehicles.

Approximately: $W_A = (W_1 - 3 - \frac{X}{200}^4) \cdot 0.297$

for $W_1 < 50.10^3$ kg:

$$W_A \approx 0.295\ W_1\ (1 - \frac{0.7}{\sqrt[5]{140W_1}})$$

for $W_1 > 50.10^3$ kg:

$$W_A \approx 0.302\ W_1 - 591$$

The question then arises, "when is a liquid motor preferable to a solid one?" Approximately:

$$0.3\ W_1 - 500 > 0.25\ W_1 \quad \text{or}$$

$$W_1 > 10000 \text{ kg}$$

So we can conclude that liquid propellant motors for braking will be more useful for only very large vehicles, beginning with about 10 tons.

d. Example weight figures.

TABLE II

Sample Weight Breakdown for Solids

Solid Propellant

W_1 =	250 + 6.2	10,000 + 6.2	30,000 + 6.2	kg
W_7 =	6.2	6.2	6.2	

for 20 g's mean deceleration: $S = \frac{400}{n} = 20$ km,

$W_7 = 3 + 3.2 = 6.2$ kg

Fuel used:	164.2	657.2	19,710	kg
Residuals:	2.3	13.1	394	kg
Motor dry:	21.2	65.8	1,579	kg
W_A :	62.3	2639	8,317	kg
Shock absorption:	15	300	500	kg
Control equipment:	20	300	500	kg
Payload support structure:	5	100	300	kg
Payload:	22.3 (8.7% of 256.2)	1939 (19.3% of 10,006.2)	7,017 (23.4% of 30,006.2)	kg

TABLE III

Sample Weight Breakdown for Liquids

Liquid Propellant

$W_1 - W_7$ =	10,000	30,000	100,000	kg
n_{opt} =	8	5	4	--
S =	50	80	100	km
W_7 =	128	822	2,003	kg
V_{id} =	2,792	2,850	2,875	m/sec
$e^{\frac{V_{id}}{2,950}}$ =	2.573	2.626	2.651	--

TABLE III (Continued)

Fuel Used =	6,114	18,576	62,278	kg
Residuals	122	372	1,246	kg
Thrust	50,000	90,000	250,000	kg
Engine dry	800	1,500	4,000	kg
Tankage	306	929	3,114	kg
W_A	2,658	8,623	29,362	kg
Shock absorption	250	350	450	kg
Control equipment	350	350	600	kg
Payload support structure	100	250	450	kg
Payload	1,958 (19.3% of 10,128)	7,483 (24.3% of 30,822)	27,862 (27.3% of 102,003)	kg

Above W_1 = 10 tons, the liquid propellant braking motor seems to be superior to the solid propellant motor.

e. Remarks on the graphs. The calculation gives directly W_A versus weight on the way (W_1). W_A has to be split in various parts: shock absorption systems, control equipment, payload support structure, and payload. Payload is plotted as a percentage of W_1 in Figure 1. For small W_A (62.3 kg), the payload is found to be only a relatively small percentage (22.3 kg or 36%), but for large W_A (29,362 kg), a very large percentage (27,862 kg or 95%) is found. Even if W_A were a constant percentage of W_1, the payload percentage would go down for smaller vehicles. Now the payload can be split into further sub-parts: Power supply, transmitter and active payload. In the example given in Section III, Paragraph 1c, the active payload is only five out of 20 kg or 25%, and at about 15 kg payload (W_1 = 188 kg) no active payload can be carried at all. Again for heavy payloads we could have the following subdivisions:

Power supply	1,000 kg
Transmitter	500 kg
Active payload	26,362 kg
Payload	27,862 kg

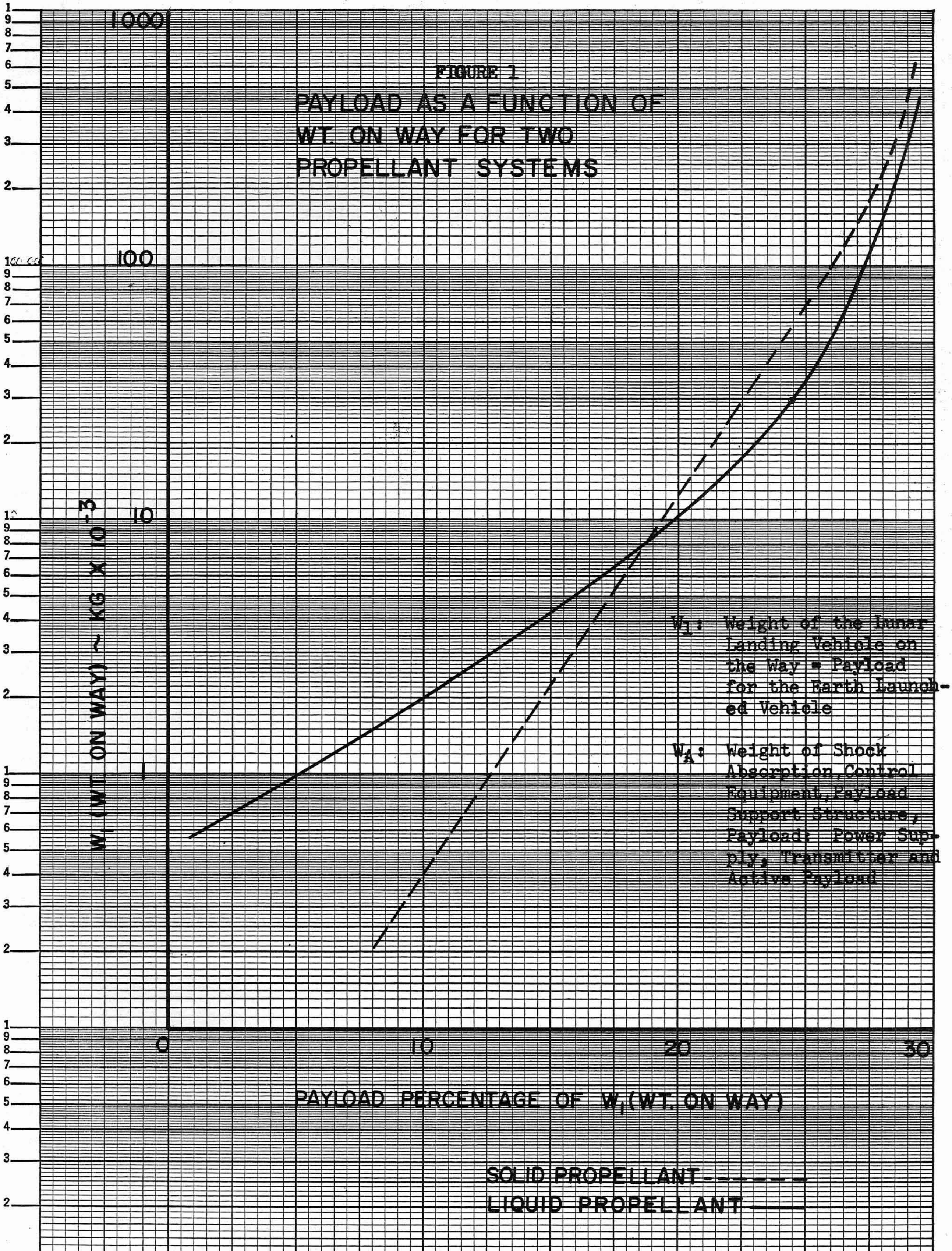


FIGURE 1

PAYLOAD AS A FUNCTION OF WT. ON WAY FOR TWO PROPELLANT SYSTEMS

Here, the active payload is 94.5% of the payload. In Figure 2, the active payload is plotted as percentage of W_1.

2. Guidance

In order to solve the problems of determination of the velocity vector and, later, alignment of the vehicle axis along that vector, the following three courses of action might be taken:

1. Insure sufficiently accurate cutoff data to the vehicle.

2. Apply mid-course correction.

3. Employ terminal guidance.

In practice, a compromise of the three courses will probably be made for a soft lunar lander.

Action (1) above involves the following difficulties:

1. Cutoff velocities to about 1.0 m/sec.

2. Guidance errors not to exceed about 0.1°.

3. Injection at correct altitude.

4. Correct lead angle needed (time of launch).

5. If launch is outside the lunar plane the difficulty is increased.

The inaccuracies allowed by a system which demands a lunar hit by controlling cutoff parameters are very low. To insure the landing of the vehicle within a particular area on the moon requires even more stringent requirements. For this, some other scheme will probably be necessary. This scheme could be a mid-course guidance system that corrects the vehicle sometime during its flight time. This could be accomplished by various methods such as the system proposed by M. W. Hunter in reference 12a.

Another mid-course guidance scheme is described in Figure 3. It could be of some use in the actual working design of such a system.

The final guidance will play a major role in the accomplishment of any space mission, particularly that of the lunar lander. Various schemes are described which might provide the nucleus of an idea for an actual working system.

3. Example Guidance Schemes

a. Mid-course guidance scheme. If the vehicle path (Fig. 3) is along path (1) so that at position (a) the viewer on board sees the

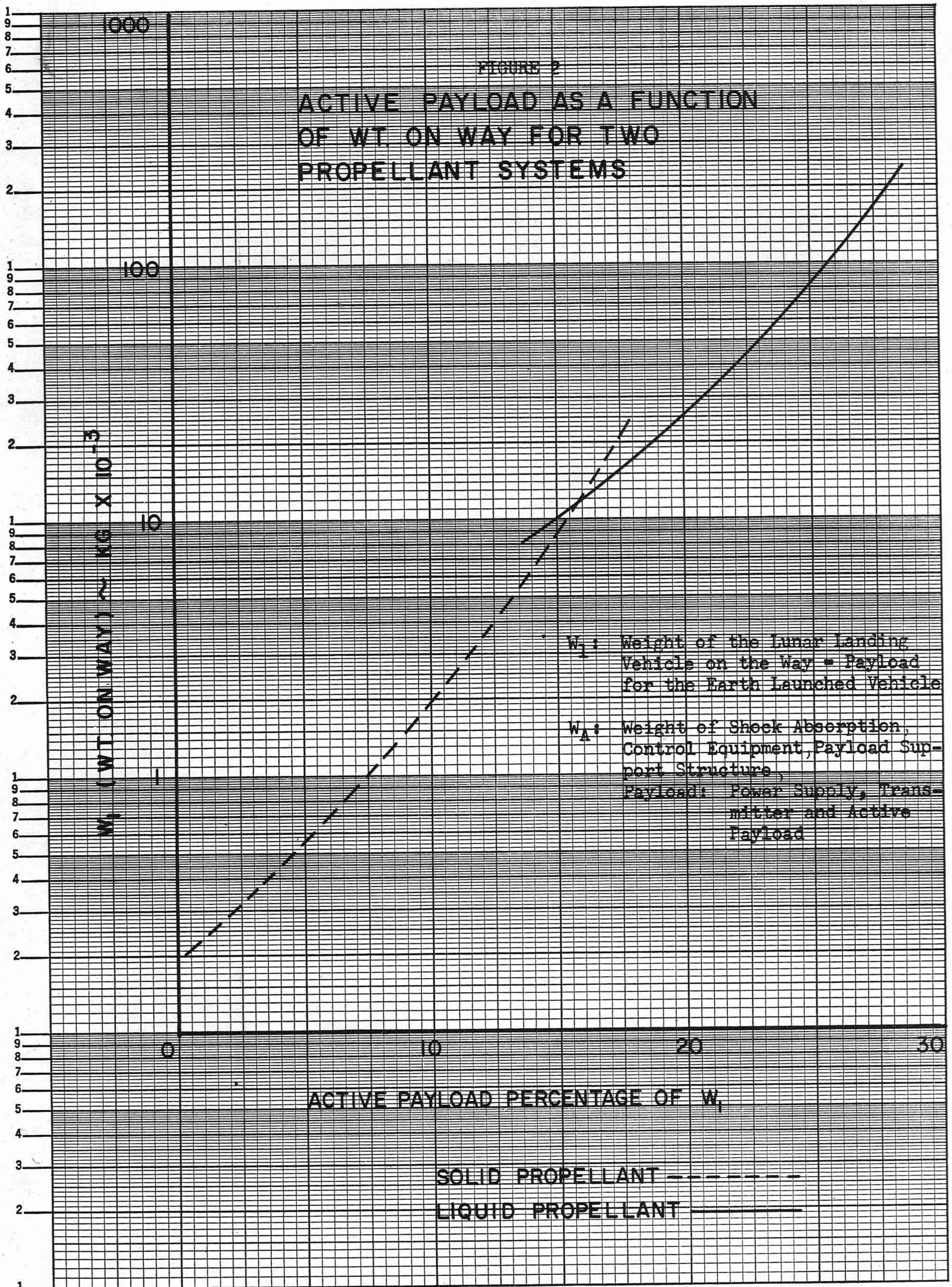


FIGURE 2

ACTIVE PAYLOAD AS A FUNCTION OF WT. ON WAY FOR TWO PROPELLANT SYSTEMS

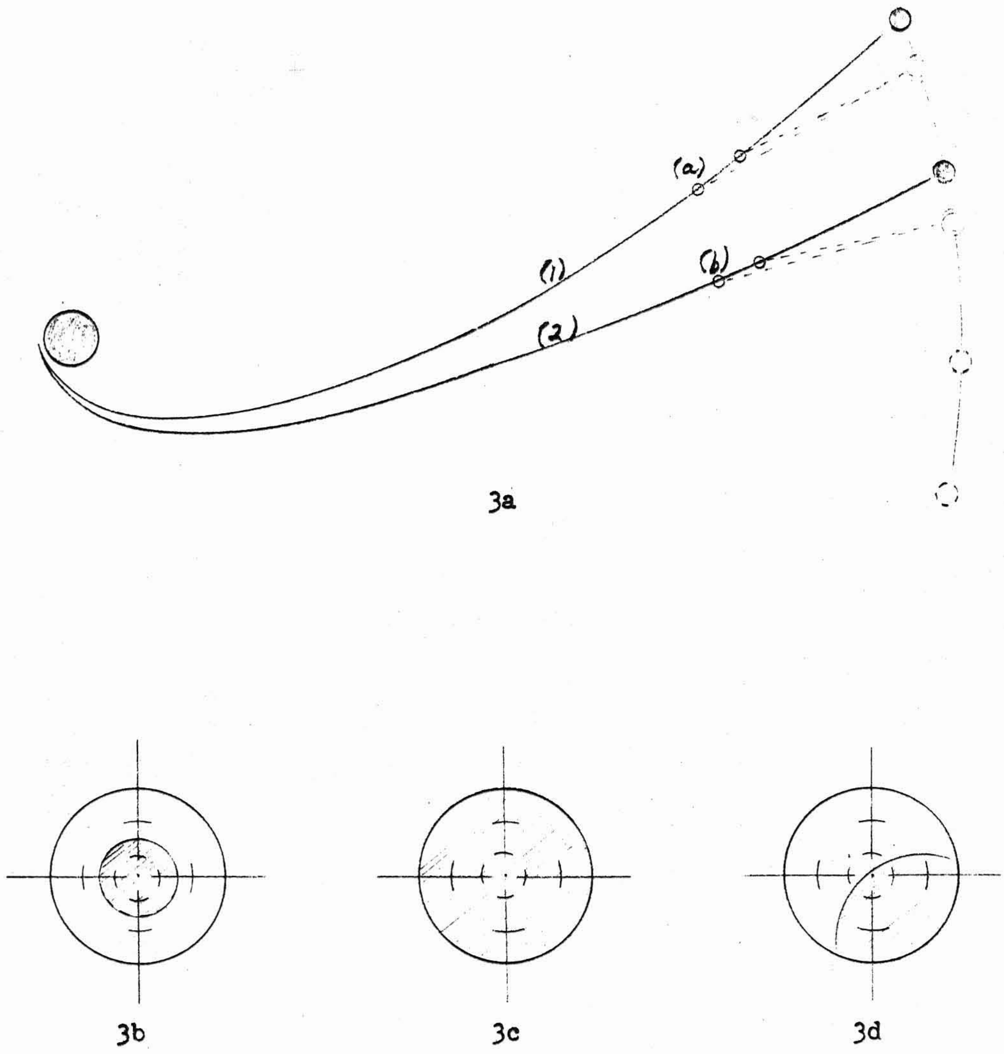


FIGURE 3. MID-COURSE GUIDANCE SCHEME

lunar disk on a screen below, as in Figure 3b, then the distance from vehicle to moon is known. If a timer indicates the flight time, then the velocity and trajectory is also known. If the vehicle flight path was along path (2) so that the disk is viewed at (b), then another set of parameters are known about the trajectory. When the disk is picked and centered in the viewer, an attitude gyro is set into motion to hold that attitude for a certain time interval. At certain time intervals later the viewer picks up the picture as shown in Figure 3c so that the following is known: The vehicle has approached a certain distance in time (t) along the precalculated path. If the picture is viewed as in Figure 3d then the distance as well as the angle between vehicle axis and line of sight are known. This can give the needed correction information to the vehicle. An accuracy of only rough magnitude need be realized for this to be useful.

A modification on this system could be for the viewer to constantly look at the disk and calculate the change in the angle between the line of sight and the space-fixed axis of the vehicle. This could be computed to give the correct alignment for the vehicle axis to coincide with the velocity vector and be periodically corrected until terminal guidance could take over.

Any usable mid-course guidance system should be viewed as only a rough and approximate guidance. It should serve as the initial phase of a vernier guidance, the mid-course being the main guidance and the terminal the vernier guidance. This compromise might reduce the weight of a system more than would be the case if either was taken as the entire system.

b. Infrared horizon seeker. With the use of an infrared horizon seeker as at 1 (Fig. 4) looking at points A and B, angles λ_L and λ_R can be measured. By the use of a rudimentary computer, they can be made equal using attitude control rockets. This gives the local vertical. Then using a radar altimeter, the vertical velocity component can be determined. This component can then be retarded by a braking rocket. If the dirft velocity, V_x, is to be retarded, some other method must be found to find that component and then orient the rocket in that attitude.

c. Surface feature lock-on system. At point A of Figure 5, the vehicle is arriving at velocity V oriented as at A. By use of a horizon seeker and using attitude control rockets, the vehicle is oriented as at B. Here, a radar fixes on the nearest prominent ground feature and angle B_1 is measured. At point C the same feature then gives angle B_2 between the feature seeker and the local vertical. Then $B_1 + B_2$ equals the total angle change. The radar gives the velocity vertical to the ground. Knowing angle B_1 and side a_1, side b_1 is found. As the vehicle moves to position C, angle B_2 is found. Radar gives a_2 so that b_2 is found. Naturally $b_1 + b_2 = x$, $a_1 - a_2 = y$, and since z is parallel to B-C, angle λ_1, equals λ_2. This triangle gives all the elements and by their solution one finds: The vertical velocity V_y, the drift

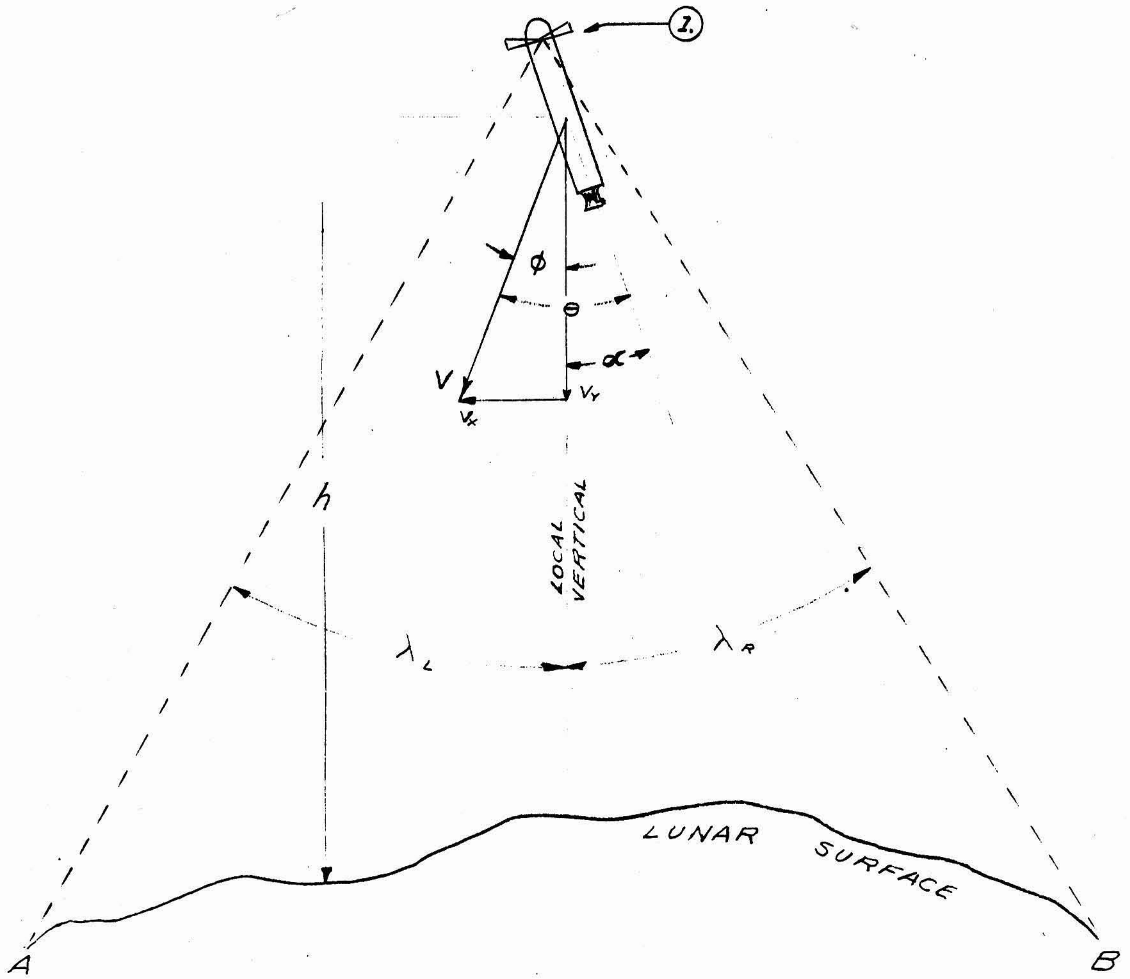


FIGURE 4. INFRARED HORIZON SEEKER

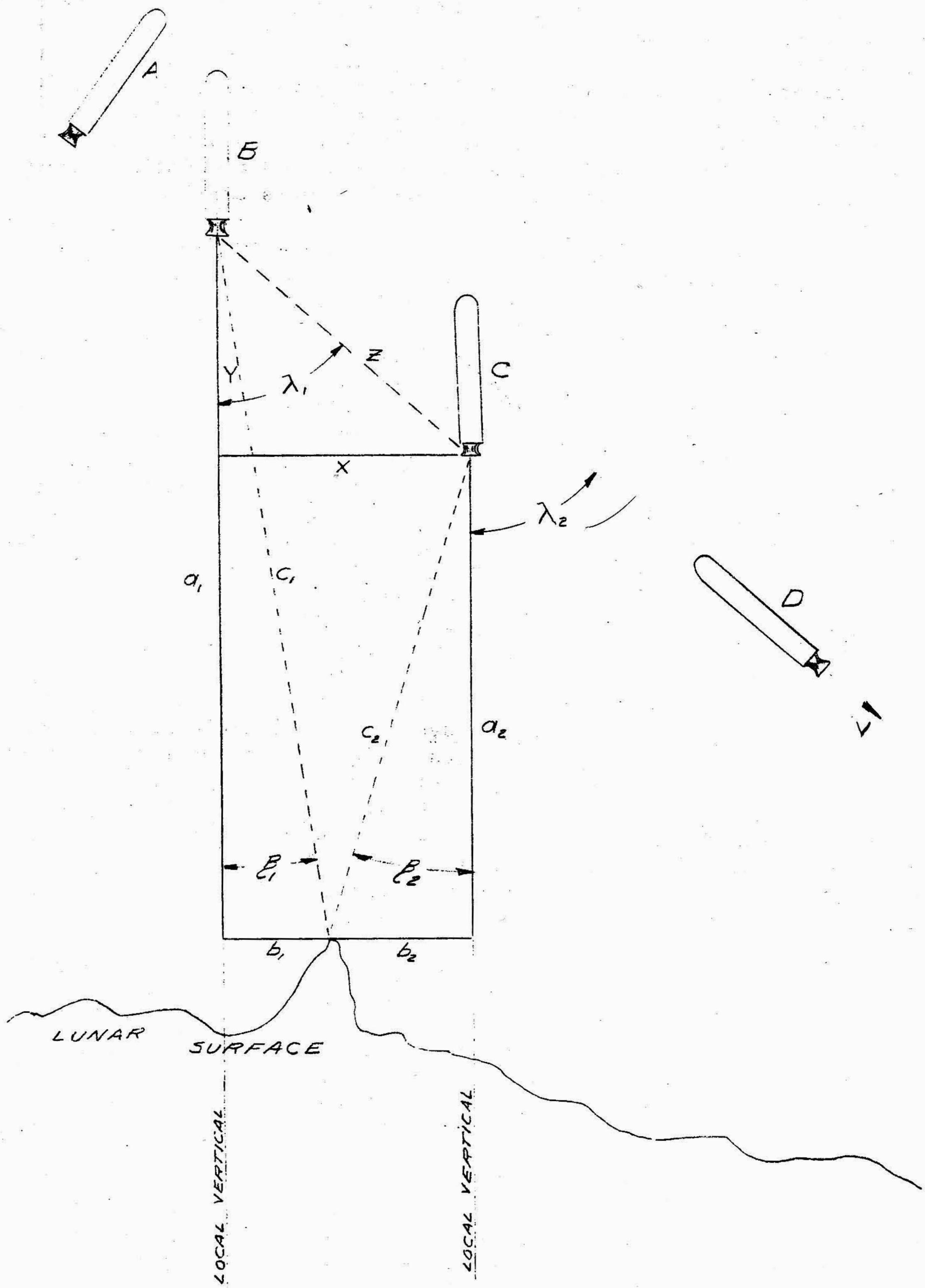


FIGURE 5. SURFACE FEATURE LOCK-ON SYSTEM

velocity V_x and the angle through which the vehicle must be turned. As a check, velocity V can be calculated from distance Z and time t. By the time it takes the vehicle to reach D, the vehicle is reoriented so that the braking impulse is along the velocity vector. Only the remaining velocity (inaccuracies of system) and that acquired by the free fall from burnout of the braking rocket will have to be compensated for in impact. The errors incurred by this method are small because the angles are rather large and the time intervals small.

d. Flare guidance system. At point (1) in Figure 6, the vehicle arriving at velocity V_1 has been aligned with the local vertical by using an attitude rocket in conjunction with the horizon seeker. Here the vehicle fires a flare along the direction of the local vertical at a relative velocity V_r. This gives the flare a total velocity of V_f, or the original V_1 plus the increased velocity along the local vertical. The flare is tracked by a seeker in the vehicle. Of course, it sees the flare as if it were on the local vertical until impact of the flare at (1). At point (A) an altimeter takes a reading and a timer is initiated. At point (B) the timer indicates the time from A to B, the altimeter measures altitude again giving the vertical distance A-C traveled, and the angle α_1, is measured. This angle gives side C-B and angle α_2. From these measurements, the vertical and horizontal components of the velocity are computed. From these components, the angle (λ) through which the attitude must be changed is computed and the original velocity V_1 is found. Then the retardation rocket can be fired.

e. Flare-surface feature system. The vehicle oriented at A in Figure 7 by a horizon seeker fires a flare along the local vertical with a velocity relative to the vehicle of a known ΔV. At point B a seeker measures the angle α between the flow at impact (I) and a natural land mark which has been tracked since the flare was fired. By computation, all of the elements are found. $V_x = \Delta V \tan \alpha$ gives the drift velocity; $V_Y = V_F \cos \alpha - \Delta V$ gives the vertical velocity. This method involves a computer but not a memory device.

f. Moon-center system. Velocity V_1 is found by referencing the time to reach a certain distance from the moon (Fig. 8). By again measuring the local vertical at position B as was done at position A, the angle γ_2 is found which gives angle γ_1. This then determines elements V_2 and λ which in turn gives the drift velocity V_n ($V_n = V_2 \sin \lambda$). However, this method demands extremely accurate measurements of the angle γ_2, which is of the order of 2 seconds of arc.

g. Doppler radar system. By use of a doppler radar which scans the ground in two perpendicular planes, the direction of maximum frequency shift is found (Fig. 9). This direction is also the direction of the velocity. The vehicle is then oriented to that direction, the braking impulse given and the velocity reascertained. With vernier controls, this can be repeated until the final velocity is of any

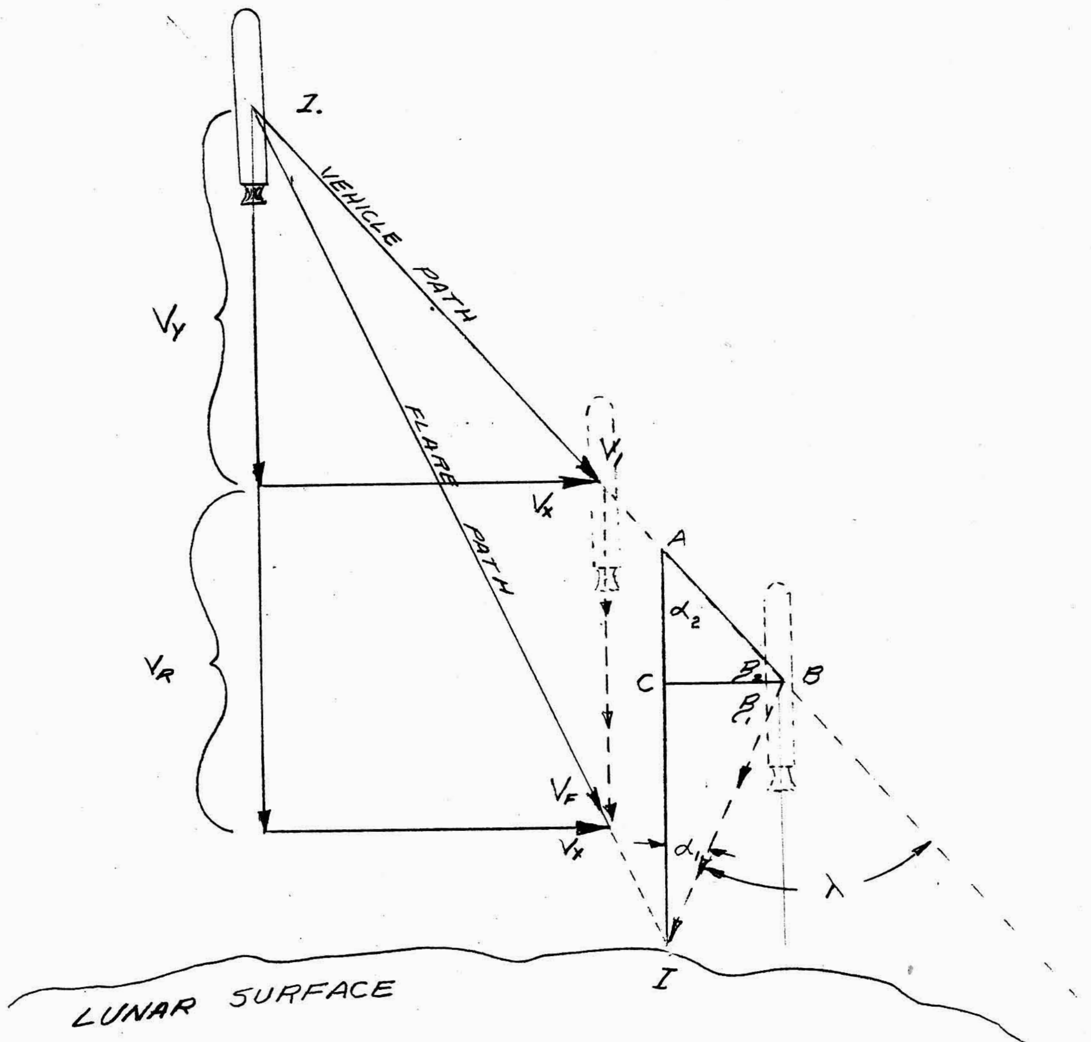


FIGURE 6. FLARE GUIDANCE SYSTEM

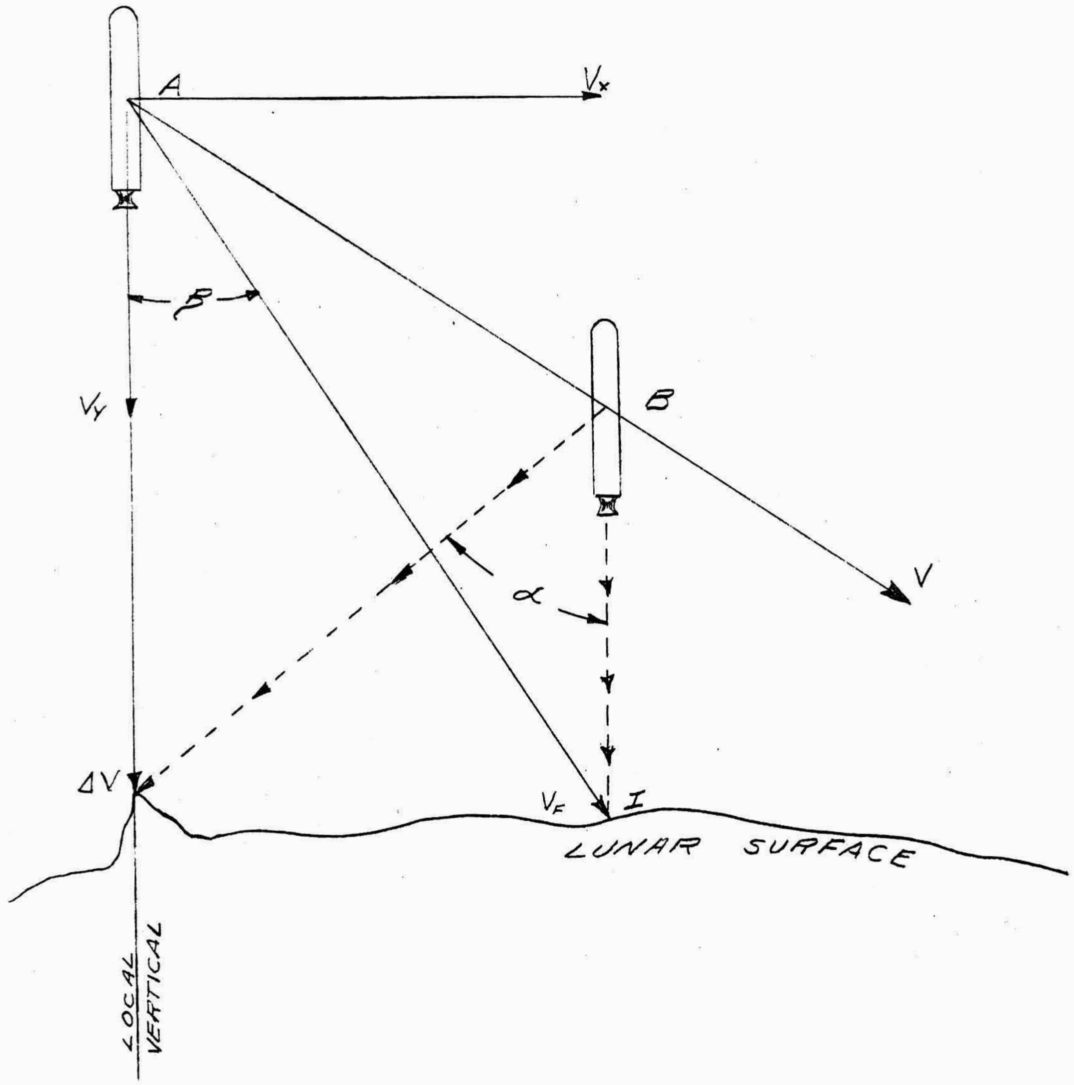


FIGURE 7. FLARE-SURFACE-FEATURE SYSTEM

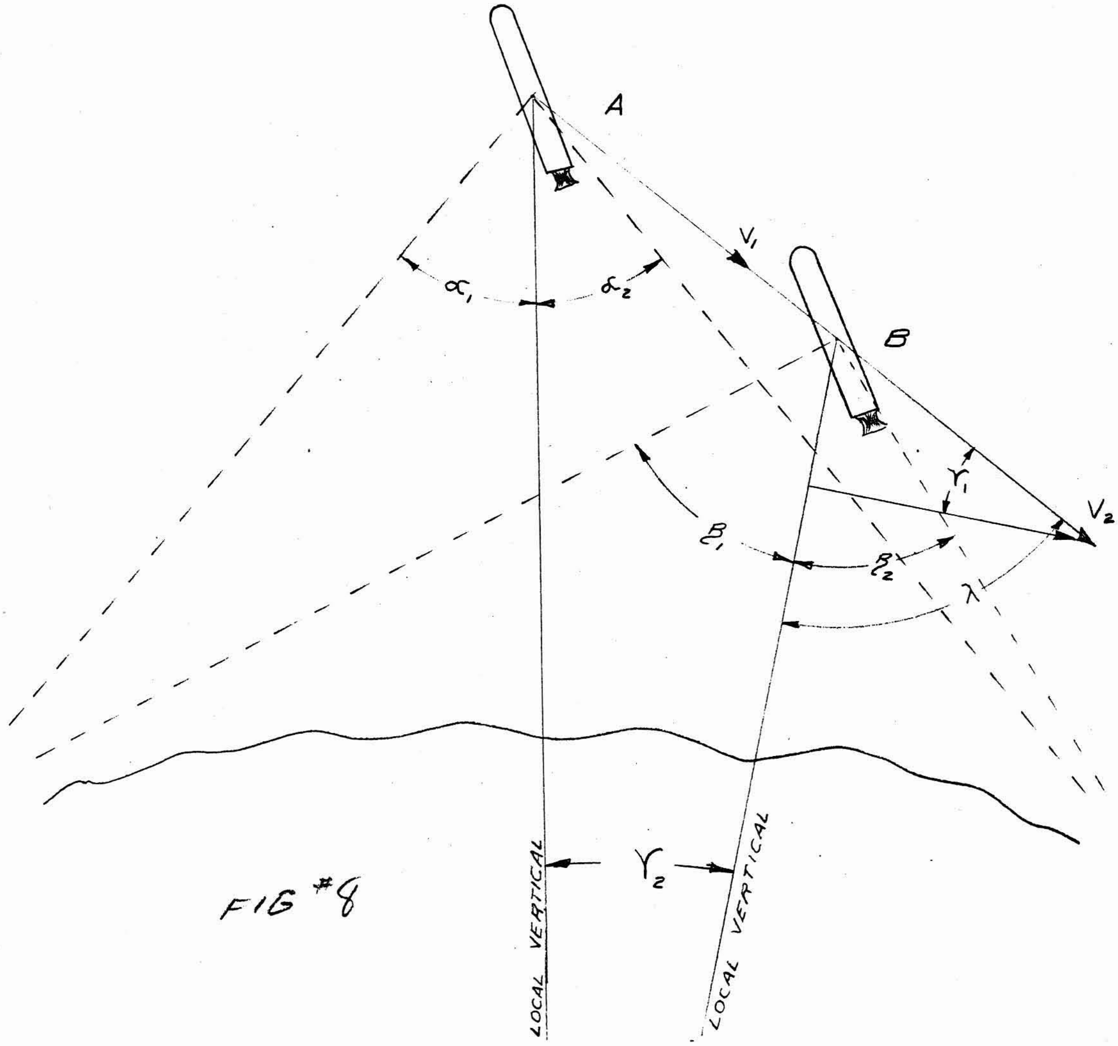


FIGURE 8. MOON CENTER SYSTEM

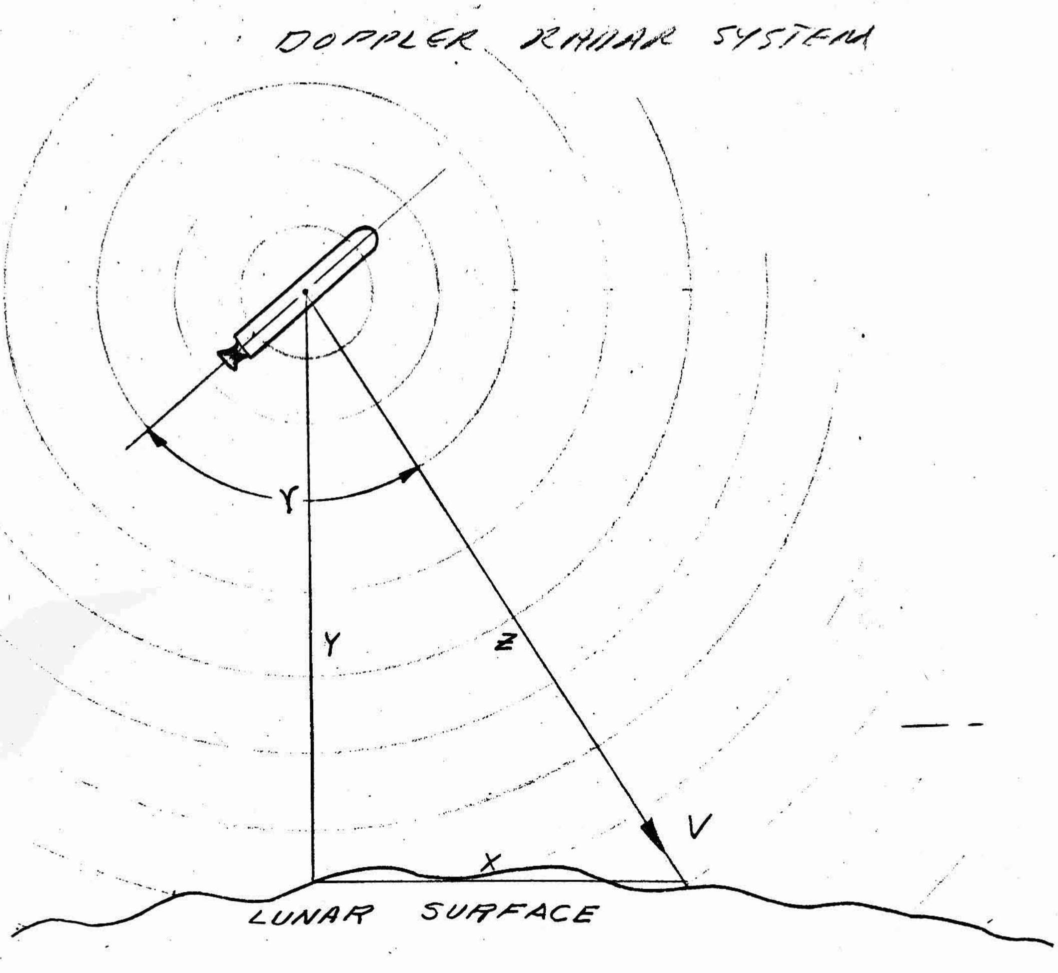


FIGURE 9. DOPPLER RADAR SYSTEM

desired magnitude. The accuracy of this method is about 0.5 to 1% in Δf, and 0.2% in angle. At present the weight is roughly 85 pounds for a range of 100,000 ft. This figure can probably be reduced considerably.

4. Other Possible Schemes

Other schemes which do not appear practical, on the surface but might induce ideas of some schemes which are practical are as follows:

(1) The sweeping out of ionized particles within the lunar atmosphere is considered. This might indicate the direction of the velocity vector if enough particles are impacted upon a revolving plate. The direction of maximum impacts might indicate the true direction of velocity. The plate might be charged and the potential difference might indicate the presence of an unbalanced system and point to the velocity direction. Some way might be devised to utilize the presence of particles which make up the lunar atmosphere. However, the extreme thinness of the lunar atmosphere makes this possibility remote. Solar radiation and micrometeorites should be taken into account.

(2) Another possible scheme suggested is to eject a ribbon of extremely low density to be deflected astern of the approaching vehicle. This means that the atmosphere must impart a dynamic pressure on this material which it does not do sufficiently for this method. A gas could be released and the direction of its motion would be opposite the vehicle's motion. That also depends on air density; therefore, it probably would not work with the thin lunar atmosphere.

(3) The possibility of utilizing the magnetic field of the moon (if it has one) has been suggested as a possible method to find the true spatial attitude of the vehicle.

5. Vehicle Design Schemes

Some simple schemes are set forth, more as ideas than proposals, for the final retarding stage of the lunar landing vehicle. The type of design will depend upon many factors which cannot be specified until a specific payload is designated.

a. Figure 10. This is a simple type vehicle which is retarded by rocket impulse. The final shock is taken up by the case of the expended rocket which acts along the penetration principle. A series of penetration spikes are incorporated into the casing wall, giving retardation also through crushing.

b. Figure 11. This type is similar to the one in Figure 10 and is similar to the Rand spike proposal. It has a single spike located within the propellant grain of the final stage rocket. The expended case and spike absorb the landing shock.

c. Figure 12. A piston fitted with a penetration spike absorbs the landing impact. The combination can be applied with a retardation rocket also.

d. Figure 13. A series of shear pins and crushable chambers may be used for final impact. Again, the gross amount of the landing energy may be absorbed by a braking rocket.

e. Figure 14. An inflated bag of liquid or gas similar to the gaseous bags for parachute drops could be used to absorb final impact. The retardation rocket can be utilized to advantage here also.

f. Figure 15. Rocket impulse alone provides the braking. The initial impulse could be started at a specified altitude with the final short impulse triggered by means of sensors that protrude below the rocket nozzle.

g. Figure 16. The final impact can be taken up by hydraulic absorption gear, with the major portion having been taken up by the retardation rocket.

h. Figure 17. The omnidirectional impacting device can be made of crushable material, fluid, or a gas, bounded by a container. Payload on this type is located centrally.

Each of the first seven types of landing vehicles has these distinct criteria: the accurate velocity vector must be known, the attitude of the vehicle must be controlled before the rocket is fired (retardation rocket), and lastly, the impact velocity must be controlled to within given limits in order for the impact absorption device to be within weight limitations. These three criteria must be optimized for each vehicle configuration in order to obtain the maximum performance from the lunar landing vehicle.

The eighth type of lander might have an application since the first two criteria above need not be observed rigidly. The omnidirectional impact protection might involve added complications in payload utilization.

6. Final Impact

One problem that confronts the lunar landing is to attain zero velocity relative to the vehicle and the moon at lunar surface. Using a single stage retardation rocket, it is improbable that zero velocity and zero altitude is accomplished simultaneously without impact. Either burnout will occur above the surface or it will occur after the vehicle has impacted. Each of these cases results in a velocity. For a given deceleration, there is a best design point for cutoff altitude if the relative velocity is to be kept as low as possible upon impact.

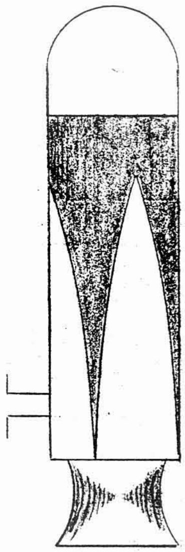


Figure 10

ROCKET RETARDED WITH INTEGRATED CASE SPIKE

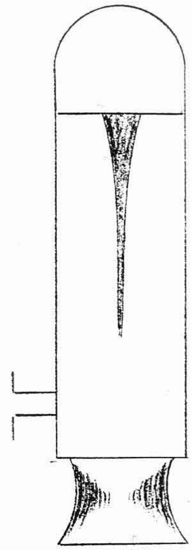


Figure 11

ROCKET AND SPIKE RETARDED

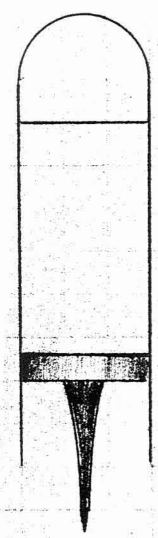


Figure 12

ROCKET, SPIKE AND PISTON RETARDED

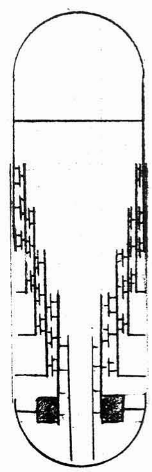


Figure 13

ROCKET AND SHEAR RETARDED

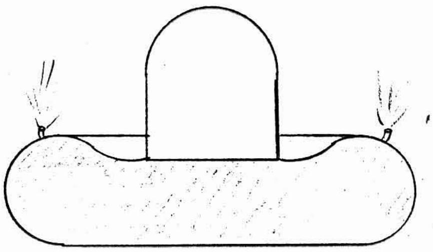


Figure 14

ROCKET AND GAS
BAG RETARDED

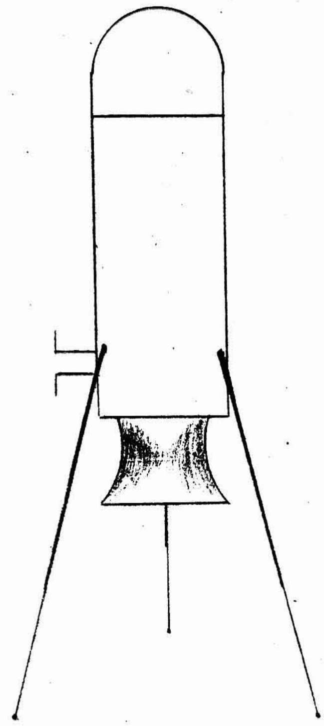


Figure 15

PHYSICALLY TRIGGERED ROCKET

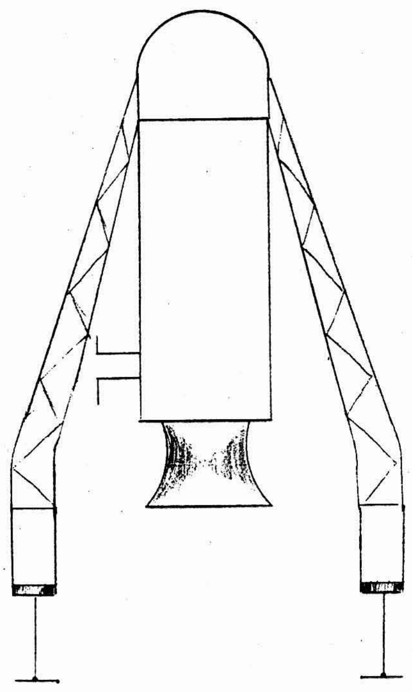


Figure 16

HYDRAULICALLY RETARDED

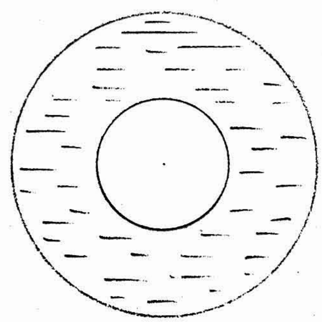


Figure 17

OMNIDIRECTIONAL PACKAGE

The velocity which might result from burnout too soon should be equal to the velocity which might be encountered if burnout occurred too late in order to minimize its maximum possible value. The assumptions taken to resolve this problem are:

(1) Exact 3,000 m/sec lunar approach velocity.

(2) No error in the precise moment of rocket retardation ignition.

(3) Altitude of ignition is exactly as design value.

(4) Approach path is perpendicular to surface.

Of course, none of the above are met, but if it is shown that under these ideal conditions the expected impact velocity is yet too high for a soft landing lunar vehicle, then it certainly will not reduce that velocity when actual conditions are considered. The analysis is simplified with the ideal assumptions; however, a more thorough analysis using actual dispersions, errors and tolerances should be made before an actual vehicle is designed. Assigning a $K\sigma$* limit to errors, the general problem can be resolved in terms of braking distance and acceleration. If on Figure 18 the $K\sigma$ limit on errors can result in burnout from point (1) to point (2), where is the design point (3)?

Let the final deceleration be mg_o with the lunar gravity equal to $0.164\ g_o$. Then:

$$V_1 = \sqrt{2 \cdot 0.164\ g_o\ (h + K\sigma)}$$

$$V_2 = \sqrt{2\ mg_o\ (K\sigma - h)} \qquad ; \text{ for } V_1 = V_2$$

$$h = K\sigma \left(\frac{m - 0.164}{m + 0.164}\right)$$

Distance traveled during braking equals:

$$S = \frac{400{,}000}{n} \text{ meters}$$

m = final deceleration, $n = m/2.5$ = mean deceleration.

This is found from the arrival velocity (3,000 m/s) and the deceleration assumed.

The burning time dispersion under temperature controls is about 1% for rockets fabricated with care. This is a good average figure which can be taken for solid propellant rockets which a lunar lander would use. Therefore, we can assign σ = 1% of S.

*See Appendix 'A'

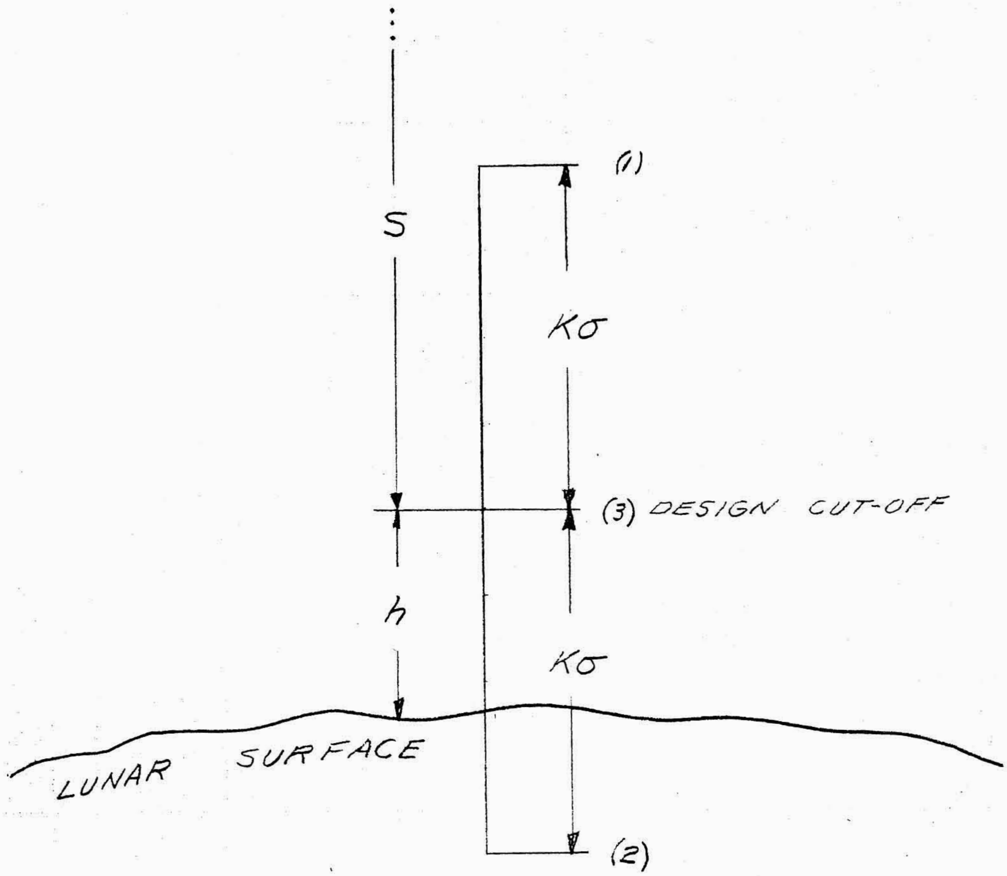


FIGURE 18. BURNING PATH LENGTH DEVIATION

Then: $h = K\sigma \left[\frac{m - 0.164}{m + 0.164}\right] = K\sigma \left[\frac{2.5\ n - 0.164}{2.5\ n + 0.164}\right]$

$$= .01K \times \frac{400{,}000}{n} \times \frac{2.5\ n - 0.164}{2.5\ n + 0.164}$$

$$\approx 4000\ \frac{K}{n}\ \text{meters}$$

Using the altitude and the acceleration the velocity is determined:

$$V_{max}\ \text{impact} \approx 160 \sqrt{\frac{K}{n}}$$

Assuming that K = 1, 2, 3 and the deceleration (n) = 5, 10, 20 the following table can be formulated:

TABLE IV

Max Impact Velocity vs Mean Deceleration

K = 1		K = 2		K = 3	
n	V	n	V	n	V
5	77	5	109	5	133
10	54	10	77	10	94
20	38	20	54	20	68

Figure 19 is taken from Table IV. From the table or the curve, it can be seen that even under "idealized" conditions, the impact velocity may be too high for a soft landing lunar vehicle. For practical systems, it appears that the landing vehicle must utilize either a multi-level thrust rocket or a multi-stage rocket. If this is not done, perhaps a closer control over path length deviation can be accomplished. Yet, the utmost limit would appear to be 0.1%, and this still gives over 45 m/sec impact velocity for n = 5 and K = 3.

7. Impact Velocity Vs Design

The problem of bringing a scientific payload to rest on the lunar surface in working condition is a straight-forward application of basic principles. The complexity of the problem is strictly a function of the demands imposed by the final approach conditions. The problem in general involves three basic parameters: (1) deceleration limits,

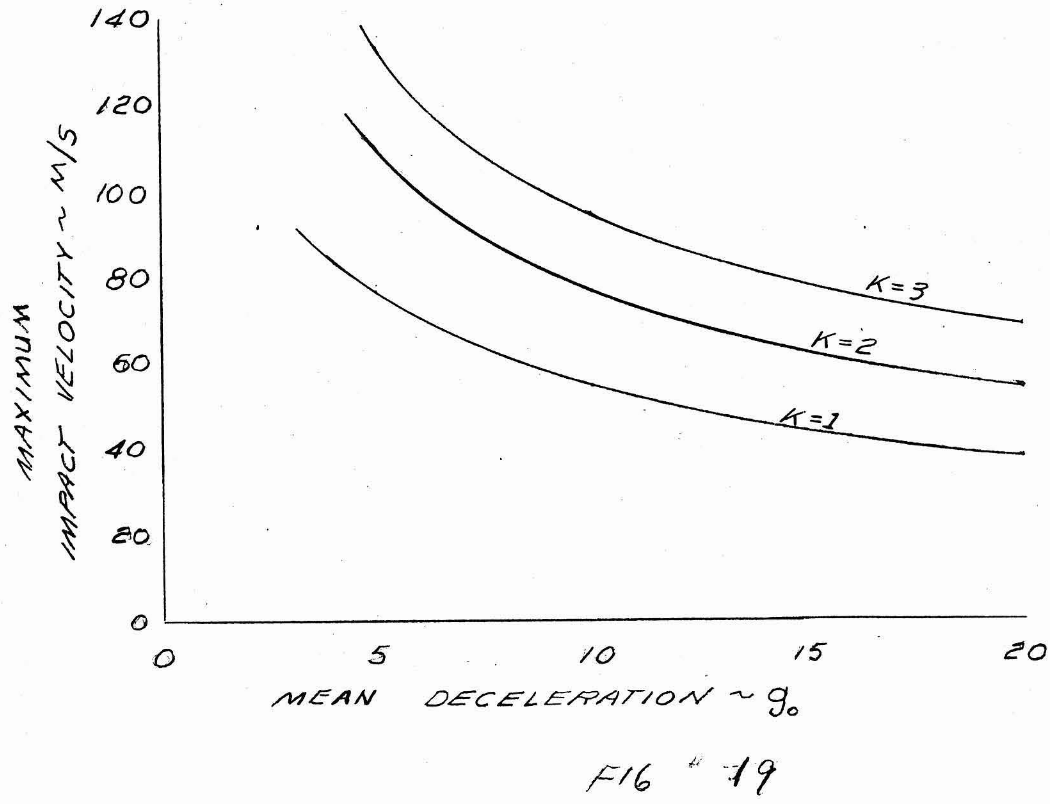


FIGURE 19

MAXIMUM IMPACT VELOCITY
VS
MEAN DECELERATION

(2) spatial attitude of the approaching vehicle, and (3) final approach velocity.

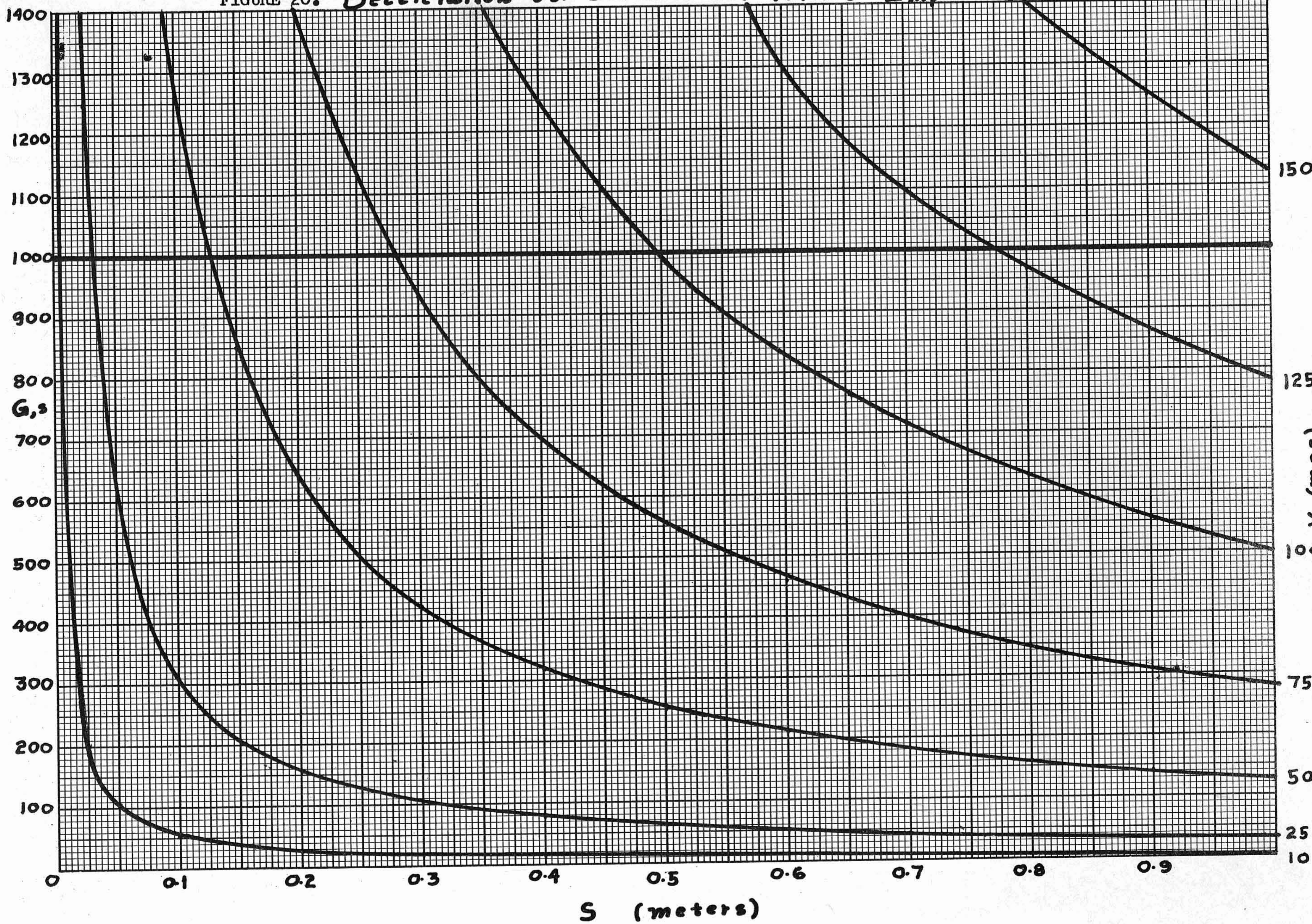
a. Deceleration. As shown in Figure 20, the maximum allowable deceleration is the major factor controlling the amount of impact protection required. As the allowable deceleration limit is reduced, for a given impact velocity the travel distance through which the deceleration must take place must increase. As this travel distance increases, the amount of shock absorbing material required will increase whether it be a spike or a deformable-body-type shock absorber.

The maximum allowable deceleration is dependent upon the type of payload under consideration. For small, compact instrument packages, an upper limit of 1,000 g's is reasonable. As the weight and complexity of the landing vehicle increases, the maximum deceleration limit will decrease rapidly. This decrease is imposed by structural limitations rather than the limitations of the individual components. In the lower limiting case of a manned vehicle which must land and eventually lift off, the upper deceleration limit will probably be imposed by structural limitations since excessive shock absorption equipment would constitute a severe weight penalty.

b. Spatial attitude. The final attitude or orientation of the approaching vehicle will be the governing factor in regard to the type of shock absorbtion system selected. If the longitudinal axis of the vehicle can be oriented within tolerances of ± 2 degrees with the velocity vector, the penetration spike is the simplest and most logical choice. If these close spatial attitude tolerances cannot be achieved, a multi-directional shock absorption system will be required. This type system results, for omnidirectional protection, in a spherical shape with the shock absorbtion material surrounding the payload. This approach will eliminate the spatial attitude problem, but may interfere with the deployment of payload apparatus, radio transmission, etc. This solution bears, along with it's increased reliability, a weight penalty.

c. Approach velocity. Final approach velocity vector orientation will be an important factor in the selection of a shock absorption system. The reliability of the penetration spike decreases as the horizontal component of the velocity increases. Since composition of the lunar surface is undetermined, the worst conditions, which would be hard rock, must be assumed. If the approach velocity vector makes an angle of more than 30° with the local perpendicular, (assuming vehicle alignment of ± 2 degrees) the penetration spike becomes ineffective. Since the lunar surface may be extremely rugged in the impact area, these imposed tolerances drastically reduce the reliability of the penetration spike. The multi-directional shock absorption system, with its previously mentioned disadvantages, is relatively independent of approach conditions.

FIGURE 20. Deceleration vs. Distance for Various Impact Velocities.



In general, the type of shock absorption system selected will depend upon the degree of attitude and final velocity control which can be achieved.

The magnitude of the impact velocity of the vehicle is the overall governing factor regardless of type of shock absorption system selected. It is believed that the most direct approach is from an energy standpoint as follows:

E_k = Total energy of impact body

m = mass of body

u = final impact velocity of body

$$E_k = \frac{m}{2} u^2$$

F = decelerating force

S = distance through which deceleration takes place

k = work done

$$k = E_k = 1/2\ m\ u^2$$

$$F \cdot S = 1/2\ m\ u^2$$

$$F = \frac{m\ u^2}{2\ s}$$

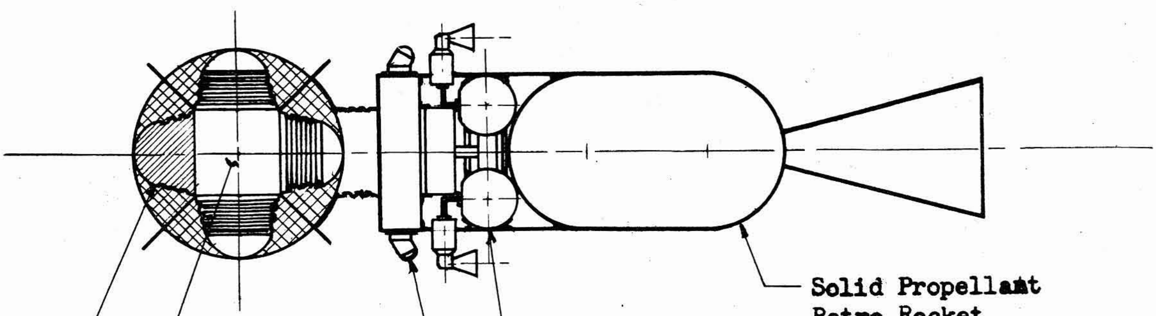
The work done may be the work required to drive the penetration spike into the lunar surface or the work required to crush a deformable body surrounding the payload.

In either case, it is equal to $\int F\, ds$ where F is the resistance of the lunar material or the force required to deform the protecting material. In neither case will this force be constant; therefore, we can only assume a constant deceleration in the most ideal case. The shape of the deceleration curve will be a function of the lunar surface composition and is therefore unpredictable. For calculation of a deformable body, a constant deceleration may be assumed, but a margin of safety must be included.

8. Possible Designs

A possible approach to the design of a vehicle is shown in Figure 21. In this approach the payload is protected on all sides, and the deceleration is accomplished over the distance which the deformable material is displaced. Figure 22 describes a modified spike version with a larger payload. By proper selection of path length and material strength the entire kinetic energy of the impact body may be absorbed.

FIGURE 21



500 Lb VERSION

MAIN DATA:

Total Wt.	505 lb.	PAYLOAD ---- 30 lb.
Propellant Wt.	330 lb.	
Motor Wt.	40 lb.	
Crash Protection	35 lb.	
Structure	20 lb.	
Attitude Control	50 lb.	

5000 lb. VERSION (See next page)

MAIN DATA:

Total Wt.	5000 lb.	PAYLOAD ---- 323 lb.
Propellant Wt.	3393 lb.	
Motor Wt. (combined)	459 lb.	
Structure (including spike)	325 lb.	
Attitude Control & Vernier	500 lb.	

24 Oct. 58 / JPG
FAT

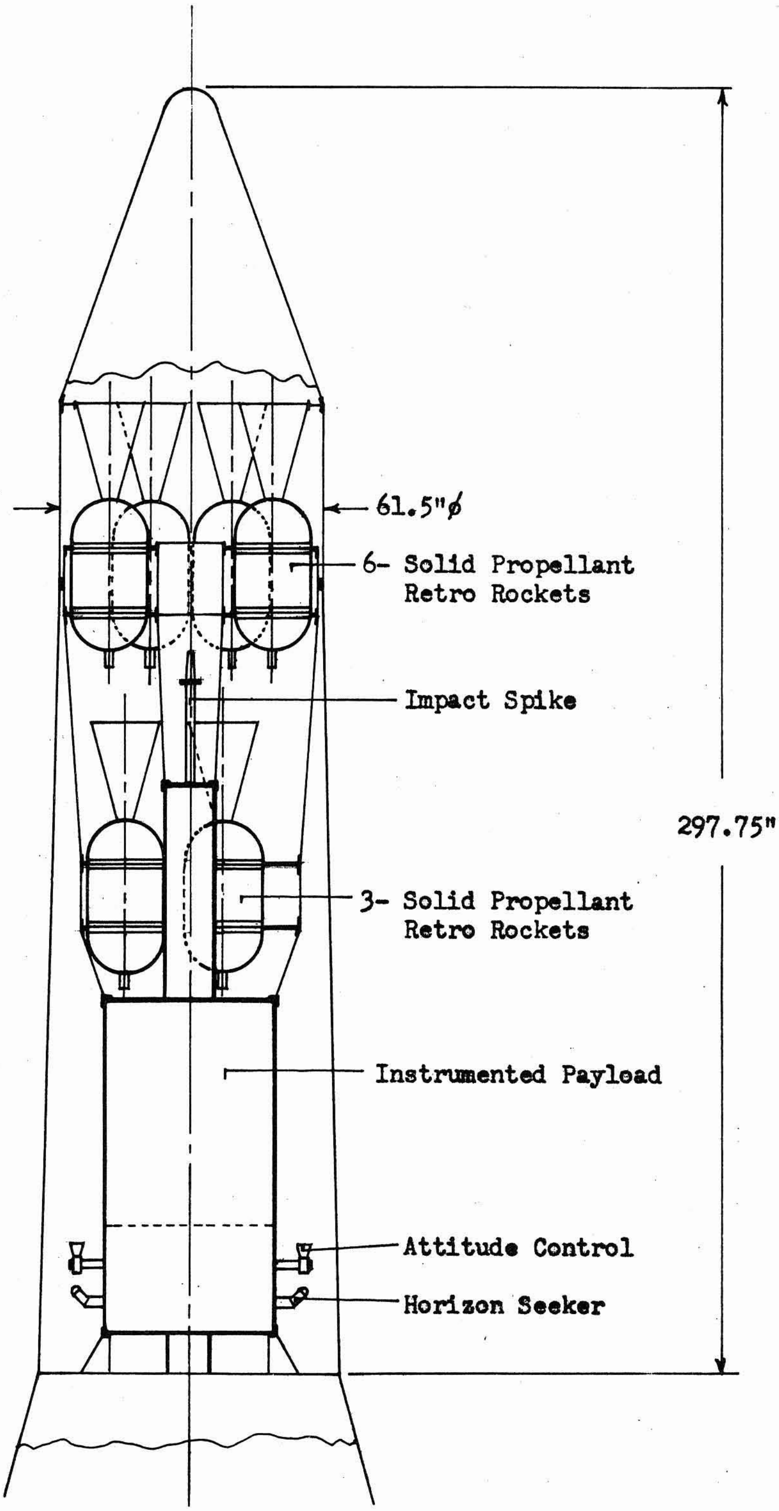


FIGURE 22. MULTIPLE ROCKET RETARDATION

23 Oct 58

In this version the payload may separate upon impact and thus the energy of the payload and protection will have to be absorbed by the shock absorber. The main advantage of this method is its independence of approach angle and vehicle attitude.

The Rand proposal for a lunar lander which utilizes a spike for absorbing up to 500 fps impact velocity is described in Rand Corporation Report RM-1720 and Report 1725. Tabel V and Figure 23 describes this vehicle. In the former report of 28 May 1956, it is stated that guidance errors will contribute to the drift velocity (velocity tangent to the moon's surface) of the landing vehicle. This is assuming a 3 σ limit on guidance errors. But the landing spike is designed upon only the vertical velocity component of the total velocity vector. The latter report of 4 June 1956 gives for the total allowable lateral velocity component a value of 2 to 3 percent of the axial velocity component. This would limit the lateral velocity to 15 fps at the maximum. Also the angle of body axis with the surface has to be greater than 60°.

Random errors in the approach may increase the lateral velocity up to 500 fps (a tangent hit). There is no assurance that spinning will maintain the desired attitude upon impact because it is not known at that time. Also the likelihood of impacting upon level ground is slim. Therefore, one feels dubious about the utility of the scheme as is.

The two reports in themselves are fairly comprehensive but put together, the conclusion one gains from them is that the Rand Corporation has, in as many words, shown that the penetration spike principle has a severe limitation--that of vertical impact with velocity vector and spike axis aligned perfectly near the parallel.

TABLE V

Space-Vehicle Weights (Rand)
(All values in pounds)

Braking-rocket propellant	204
Rocket nozzle and case	36
Altimeter	10
Landing spike	5
Outer structure	5
Spin and vernier assembly	10
Radio equipment	10
Batteries	10

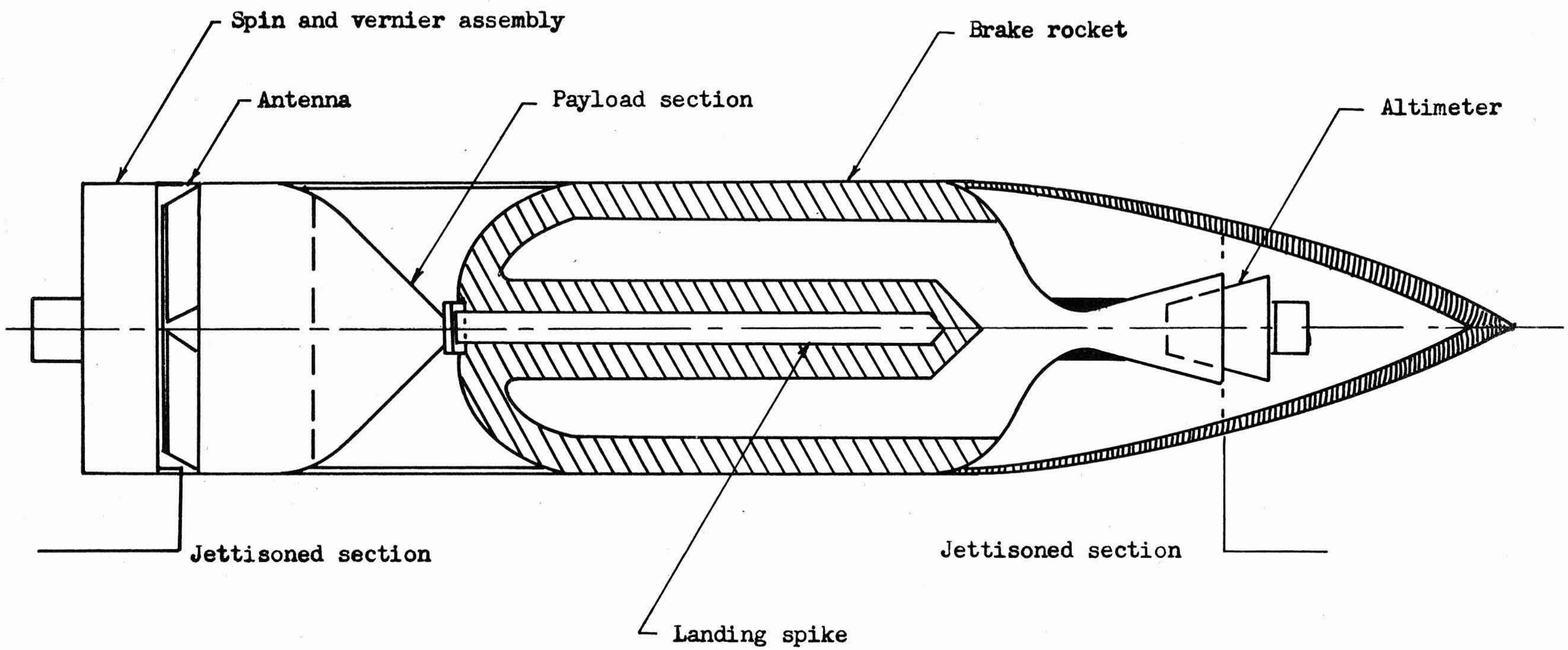


Fig. 23. General Arrangement, Payload Stage

TABLE V (Continued)

Instrument payload assembly	30
Gross (at end of power flight)	320
Less jettisoned components	-20
Gross (at start of landing phase)	300
Gross (at touch-down)	96

SECTION IV. CONCLUSIONS

It is concluded that the liquid motor is better for the braking of a lunar landing vehicle only when the weight is above the 18,000-pound class. This size vehicle soft lands about 18.75% of the 18,000 pound which approaches the moon. Below the 18,000-pound approaching weight, the solid rocket engine appears more favorable. Practical designs may shift this transition point some, but low payloads are better braked with solid propellant engines and large ones with liquid propellant engines.

Due to the limitations of cutoff parameter accuracies, the conclusion is reached that multiple guidance is necessary to insure the landing of the vehicle, especially if a particular area is desired on the moon.

Inaccuracies inherent in the vehicle and sensing components appear to dictate the use of multi-stage or controllable thrust retardation rockets. This complicates vehicle design and there seems to be no one best vehicle design for all sizes of vehicles.

It is also concluded that for each vehicle weight range there can be found an optimum design including guidance, structure and propulsion (retardation).

It is lastly concluded that detailed analyses of systems and designs should be continued.

SECTION V. RECOMMENDATIONS

It is recommended that the following areas be studied and analyzed before final decisions are rendered on vehicular design:

1. Trajectory studies should be calculated more precisely and analysis made of such.

2. Analysis of velocity and guidance tolerances at injection should be performed and their bearing upon vehicle design considered.

3. Multiple guidance versus single guidance should be analyzed as to complexity and increased probability of mission success.

4. Braking rocket controls and systems should be studied to accomplish allowable impact velocities.

5. Guidance and propulsion should be mated and optimum conditions sought between systems, for the ascending phase as well as the descending phase (from earth launch to lunar landing).

6. Landing gear design should be studied with a view to gain optimization of mission.

7. Design of the payload structure and the payload itself should be studied in order to facilitate sound analysis of the previous areas.

BIBLIOGRAPHY

Existing literature pertaining to the moon which was published before 27 Feb 58 is referenced in:

A Survey of the Lunar Project (u), Secret by H. Ruppe, ABMA Report No. DS-TN-170, 27 Feb 58.

Literature having lunar applications which has appeared since that time:

1. Ruppe, H. and Straly, W.; "Derivation and Calculation of Initial Values of Moon-Hitting Trajectories;" ABMA Report No. DSP-TM-3-58, 30 May 58.

2. Straly, W. and Ruppe, H.; "Some Earth-to-moon Flight Paths as Simulated by Analog Computor;" ABMA Report No. DSP-TN-2-58, 1 Jul 58.

3. Gibbons, J. and Shaver, F.; "Simulation of Flight Paths to the Moon;" ABMA Report No. DC-TN-215, 14 Mar 58.

4. "Lunar Probe Flight; Introduction to Flight Geometry and Accuracy Problems;" ABMA Report No. DA-TN-58-58, 22 Aug 58.

5. Lieske, H., Rand Corporation; "Lunar Instrument Carrier-Ascent Flight Mechanics (U);" SECRET, RM 1727, 4 June 1956.

6. "First Approximate Trajectory for AM 11 and 14;" (CONFIDENTIAL) ABMA, ORDAB-DAP, DF No. C-115796, 15 Jul 58.

7. The Martin Co.; "The Military Requirement For Moon Base (U);" SECRET, Oct 57.

8. Carr, R., Jet Propulsion Lab; "Lunar Coasting Trajectories;" 4/7/58.

9. Carr, R., Jet Propulsion Lab; "Addendum to Lunar Coasting Trajectories;" 4/10/58.

10. "Proceedings of 1st Lunar and Planetary Exploration Colloquium;" Vol. 1, No. 1, 13 May 58. "Proceedings of 2nd Lunar and Planetary Exploration Colloquium;" Vol. 1, No. 2, 13 May 58.

11. Nelson, S.; "Trip Report: Second Lunar and Planetary Exploration Colloquium;" Rand Corp., Santa Monica, Calif., 15 June 58.

12. Material presented at the 1958 IX IAC, Amsterdam/the Netherlands:

a. Hunter, M. W.; "Impulsive Midcourse Correction of a Lunar Shot."

b. Mickelwait, A. B.; "Analytical and Numerical Studies of Three-Dimensional Trajectories to the Moon."

c. Singer, S. F.; "Scientific Problems in Cislunar Space and Their Exploration with Rocket Vehicles."

13. Gates, G. R., JPL; "Terminal Guidance of a Lunar Probe;" 14 May 58.

14. Lieske, H., Rand Corp.; "Lunar Trajectory Studies;" 26 Feb 58.

15. Walters, L. G., Navigation; "Lunar Trajectory Mechanics" Vol 6, No. 1, 1958.

16. "Proposal For a Study on Lunar Research Flights" Part I of II; Aeronutronics Systems, Inc., 11 Apr 58.

17. "The Russian Literature of Satellites, Part I;" International Physical Index, Inc., New York, PP 107-174.

Moscow, V. A.; "Certain Problems of Moon Flight Dynamics, Egorov;" 1957.

18. Kiess, C. C. and Lassovzsky, K.; "The known Physical Characteristics of the Moon and the Planets;" ARDC-TR-58-41, Astic Doc No. AD 115-617, July 1958.

19. Gold, L.; "Earth-Moon Rocket Trajectories I;" ARS Preprint 646-58, June 1958.

APPENDIX A

COMMENTS ON PROBABILITY

Given observations of l_1, l_2, l_3,... l_n which are different only because of random influences, then the true observation is approximately:

$$x = \frac{1}{n} \sum_{1}^{n} \ell_k$$

The mean error of one single observation is:

$$\sigma^2 = \frac{1}{n-1} \sum_{i}^{n} (\ell_i - \frac{\Sigma \ell_k}{n})^2$$

The mean error σ of $\sum_k \ell_k$ is of course smaller:

$$\sigma^2 = \frac{\sigma^2}{n}$$

Therefore, if $\frac{\Sigma \ell_k}{n} = L$ is taken as the design value of "burning length", then:

$$\sigma = \sqrt{\frac{1}{n-1} \left[\sum_{1}^{n} (\ell_i - L)^2 \right]}$$

The probability of observing L between the interval a...b (see figure 24) is:

$$P = \frac{1}{\sqrt{2\pi}} \int_{\frac{a-L}{\sigma}}^{\frac{b-L}{\sigma}} e^{-\frac{x^2}{2}} dx$$

Gauss-Laplace Random error law

or if:

$$a = L - k\sigma \quad , \; b = L + k\sigma$$

Then the probability of a deviation for L between $\pm k\sigma$

$$P = \frac{1}{\sqrt{2\pi}} \int_{-k}^{+k} e^{-\frac{x^2}{2}} dx$$

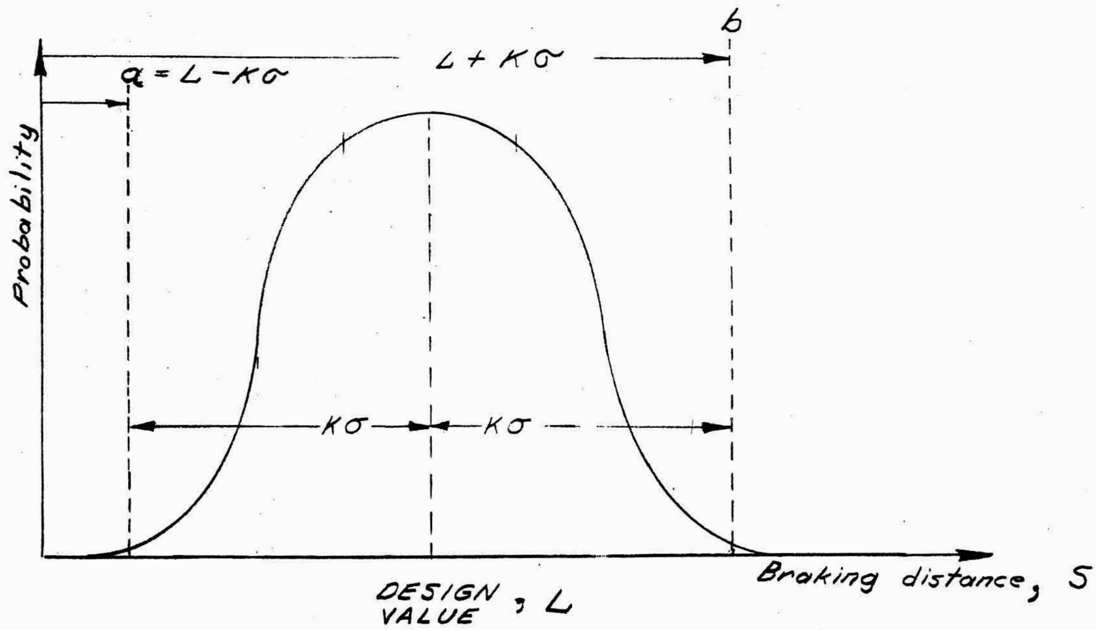


FIGURE 24. PROBABILITY CURVE

Army-Redstone Arsenal, Ala.

In tabular form:

TABLE VI

Probability Table

K	% of observations within interval
0.500	38
0.674	50
1.000	68
2.000	95.5
2.580	99
3.000	99.7
3.490	99.96
∞	100

APPENDIX B

SOLID AND LIQUID PROPELLANT COMPARISON

Let us make a weight comparison between a liquid and a solid propellant rocket motor for a given required ideal velocity (V_{ir}). Then there is:

$$V_{ir} = g_o I_{sp} \ln \frac{M}{M - m_6}$$

There follows:

$$m_6 = M \left(1 - e^{-\frac{V_{ir}}{g_o I_{sp}}}\right)$$

If $k \cdot g_o$ is the gravity field concerned, then approximately:

$$V_{ir} = V_{id} + k g_o T, \text{ and}$$

$$V_{id} \approx n g_o T \quad (n = \text{mean acceleration in units of } g_o)$$

$$V_{ir} = V_{id} + \frac{k V_{id}}{n} = V_{id} \left(1 + \frac{k}{n}\right)$$

So finally:

$$m_6 = M \left(1 - e^{-\frac{V_{id}\left(1 + \frac{k}{n}\right)}{g_o I_{sp}}}\right)$$

If now this is put in the equation:

$$\frac{g_o m_\ell}{g_o m_s} = \left(\frac{n}{7} + 0.5\right) \frac{m_{6\ell}}{m_{6s}}$$

we get for $M_s = M_\ell$ and the same required V_{id}:

$$\frac{g_o m_\ell}{g_o m_s} = \left(\frac{n_\ell}{7} + 0.5\right) \frac{1 - e^{-\frac{V_{id}}{g_o} \cdot \frac{1 + \frac{k}{n_\ell}}{I_{sp_\ell}}}}{1 - e^{-\frac{V_{id}}{g_o} \cdot \frac{1 + \frac{k}{n_s}}{I_{sp_s}}}}$$

Limiting cases:

$$\alpha : \quad \frac{V_{id}}{g_o} \quad \frac{1 + \frac{k}{n}}{I_{sp}} \quad > \quad 2.3$$

Then:

$$\frac{g_o m_\ell}{g_o m_s} \approx \frac{n_\ell}{7} + 0.5$$ or the liquid system is lighter for $n_\ell < 3.5$.

$$\beta : \quad \frac{V_{id}}{g_o} \frac{1 + \frac{k}{n}}{I_{sp}} < 0.2$$

$$\frac{g_o m_\ell}{g_o m_s} \approx \left(\frac{n_\ell}{7} + 0.5\right) \frac{\left(I + \frac{k}{n_\ell}\right) \cdot I_{sps}}{\left(I + \frac{k}{n_s}\right) \cdot I_{sp\ell}}$$

For

$$\frac{k}{n_\ell} \approx \frac{1}{4}, \frac{k}{n_s} \approx 0, \qquad \frac{I_{sps}}{I_{sp\ell}} = \frac{2}{3} \quad :$$

$$\frac{g_o m_\ell}{g_o m_s} = \left(\frac{n_\ell}{7} + 0.5\right) \frac{5}{4} \cdot \frac{2}{3} = \frac{5}{6} \left(\frac{n}{7} + 0.5\right)$$

Here the liquid system is lighter for $n_\ell < 4.9$.

APPROVAL

Report No. DSP-TN-13-58

H. H. KOELLE
Chief, Future Projects Design Branch

C. L. BARKER, Jr.
Chief, Astronautical Engineering Section

W. A. MRAZEK
Director, Structures and Mechanics Laboratory

DISTRIBUTION:

ORDAB-X	ATTN:	Commander, ABMA	(1)
ORDAB-C	ATTN:	Chief	(2)
ORDAB-CL			(1)
ORDAB-D	ATTN:	Director	(4)
ORDAB-DI	ATTN:	Mr. Saurma	(1)
ORDAB-DIR			(1)
ORDAB-DY	ATTN:	Mr. Constan	(1)
ORDAB-DA			(2)
ORDAB-DC			(2)
ORDAB-DF			(2)
ORDAB-DG			(2)
ORDAB-DL			(2)
ORDAB-DM			(2)
ORDAB-DV			(4)
ORDAB-DR			(2)
ORDAB-DT			(2)
ORDAB-DS			(2)
ORDAB-DSR			(1)
ORDAB-DSD			(1)
ORDAB-DSF			(1)
ORDAB-DSN			(1)
ORDAB-DSL			(1)
ORDAB-DSP			(15)
ORDAB-DST			(1)
ORDAB-DSC			(1)
ORDAB-HT			(8)