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PGR3

THE OPERATION OF FIXED WING V/STOL AIRCRAFT FROM CONFINED SPACES

by Lt Cdr D R Taylor. C ENG, MI MECH E., RN.

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Introduction

The purpose of this study has been to seek the means of operating fixed wing jet V/STOL aircraft at useful weights from confined spaces by exploiting ballistic as well as aerodynamic principles.

The methods proposed are built around the Harrier type of aircraft with an integrated lift and propulsion system and may not read across to other types without modification. Nevertheless it seems not unlikely that other fixed wing V/STOL aircraft could benefit in greater or lesser degree from the use of the semi-ballistic launching technique.

The subject has been considered mainly from a naval viewpoint but the application of some of the methods to land based V/STOL operations may be readily envisaged.

A successful solution to the take-off problem is not only a technical improvement upon existing methods. In the naval sphere it opens up a range of options reflecting back on the original problem and to some extent modifying the type of solution to be sought. This thesis cannot therefore be an entirely technical discourse although the basic technical problem is central and the bulk of the work is concerned with it.

The tactical and strategic implications provide material for a separate thesis but only a very broad and simple treatment has been possible in the time available. An attempt has been made to formulate a new line of approach with the pitfalls of over simplication very much in mind. The views expressed are the author's and do not necessarily reflect those of any official body.

Synopsis and Conclusions

1. The semi-ballistic launching technique is described and mathematical expressions for the resulting trajectory derived. Various methods of solution are described and the effects of varying the parameters are shown. Optimum conditions are determined and theoretical predictions of performance made. Possible instrumentation requirements are suggested.

Conclusion Very large reductions in launch speed (of the order of 60% to 70%) are possible compared with conventional short take-off methods.

2. The performance of the Harrier in semi-ballistic flight is examined and found to agree closely with the general theory. Stability and control are discussed.

Conclusion Similar large reductions in launch speed will apply to the Harrier. No insuperable control or stability problems are foreseen.

3. Various means of achieving semi-ballistic flight are discussed and compared with existing launching methods. The most promising methods are examined in some detail.

Conclusions a. The launching systems proposed are well within current standards of technology, knowledge and skill. All offer marked reductions in take-off space required compared with existing methods.

b. The Ski-Jump system, though not giving minimal space, is much smaller and also cheaper than any existing system and is immediately applicable to existing ships and aircraft.

c. The Flexible Launcher, though not ideal for launching Harriers, may prove to be a useful missile launcher and should be further examined to this end.

4. The application of the preferred launching methods to ships is examined. The effects of ship motion are discussed and also aircraft requirements in terms of weight, space, workshops, accommodation and manpower. The problems of equipping merchant vessels in wartime are examined and solutions proposed.

Conclusions a. Fixed wing jet V/STOL aircraft can be operated effectively from ships much smaller than hitherto.

b. The Through Deck Cruiser can be made a much more effective fighting ship by the use of Ski-Jumps or similar devices.

c. The Catamaran configeration allied with one of the suggested launching devices offers the prospect of effective aircraft carrying ships of Frigate size.

d. Ocean going merchant vessels can carry and operate an effective force of V/STOL aircraft.

5. The case for operating V/STOL fixed wing aircraft from small ships is discussed in broad terms against a background of possible tactical and strategic options. The possibility of a change in doctrine resulting from the use of fixed wing V/STOL aircraft at sea is suggested.

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CHAPTER 1

The Semi-Ballistic Method of Launching Fixed Wing Jet V/STOL Aircraft

1.1 The operation of fixed wing VTOL aircraft from small platforms on ships at sea is an accomplished fact. However a vertical take-off requires thrust to weight ratios in the order 1.15:1 with consequent limitation of the aircraft's range and load carrying capability. For most naval purposes these limitations are not acceptable.

1.1.1 The aircraft performance is greatly improved if a rolling take-off is used, it then becomes a very potent naval weapon. This technique is now well established and is as follows:

The aircraft accelerates along the runway or flight deck under undeflected engine thrust. At a pre-determined air speed or distance run, the pilot vectors the thrust downwards so that it has a vertical as well as a horizontal component and rotates the aircraft to a flying attitude. At this point the sum of the aerodynamic lift and the vertical component of engine thrust is greater than the weight of the aircraft which therefore leaves the ground. As the aircraft continues to accelerate under the horizontal component of engine thrust the wings bear an increasing share of the weight and the thrust may be progressively returned to the undeflected position.

1.1.2 Although this technique gives take-off distances very much shorter than those possible for similar conventional aircraft, it still demands an unobstructed flight deck of 500 feet or more. This can only be provided in large ships. In any ship, large or small, but particularly in warships, deck space is at a premium and the take-off space is, apart from flying operations, wasted space, severely curtailing other functions and requirements of a warship. In other words unless the take-off distance of the aircraft at operational weights is brought to the irreducable minimum the full potential of the ship/aircraft combination is not realised.

1.1.3 An inherent feature of any VTOL aircraft is that it is controllable in all axes at all speeds from zero to its maximum speed. It is this unqiue feature, hitherto unexploited in full, which it is proposed to use to further improve the take-off performance of VTOL aircraft at thrust to weight ratios less than unity.

1.2 The Semi-Ballistic Launching Technique

1.2.1 The method proposed is illustrated in Fig 1. The aircraft is launched at velocity V_L at an angle α to the horizontal.

V_L therefore has a horizontal component Uo and a vertical component Vo.

The aircraft has its thrust deflected downward through an angle such that it has both horizontal and vertical components

After being launched the aircraft follows the semi-ballistic flight path shown (vertical scale exaggerated). The pilot maintains a constant optimum angle of incidence throughout this flight path and the aircraft accelerates due to the horizontal component of engine thrust until aerodynamic lift is equal to that proportion of its weight unsupported by engine thrust plus sufficient to cancel vertical momentum gained during the descending phase of the trajectory. This occurs at point P which may be called the flyaway point. The process takes longer than a rolling take-off but V_L is a fraction of the speed required for the rolling take-off and consequently the deck space required to achieve V_L is very small.

In effect the technique is similar to a rolling take-off but the runway is in the sky.

Obviously values of Uo and Vo, α and the thrust deflection angle must be selected such that point P occurs at a safe height.

1.3 Determination of Semi-Ballistic Trajectory

1.3.1 Referring to Fig 2 consider the forces acting upon the aircraft.

FIG 1 SEMI - BALLISTIC LAUNCH TECHNIQUE

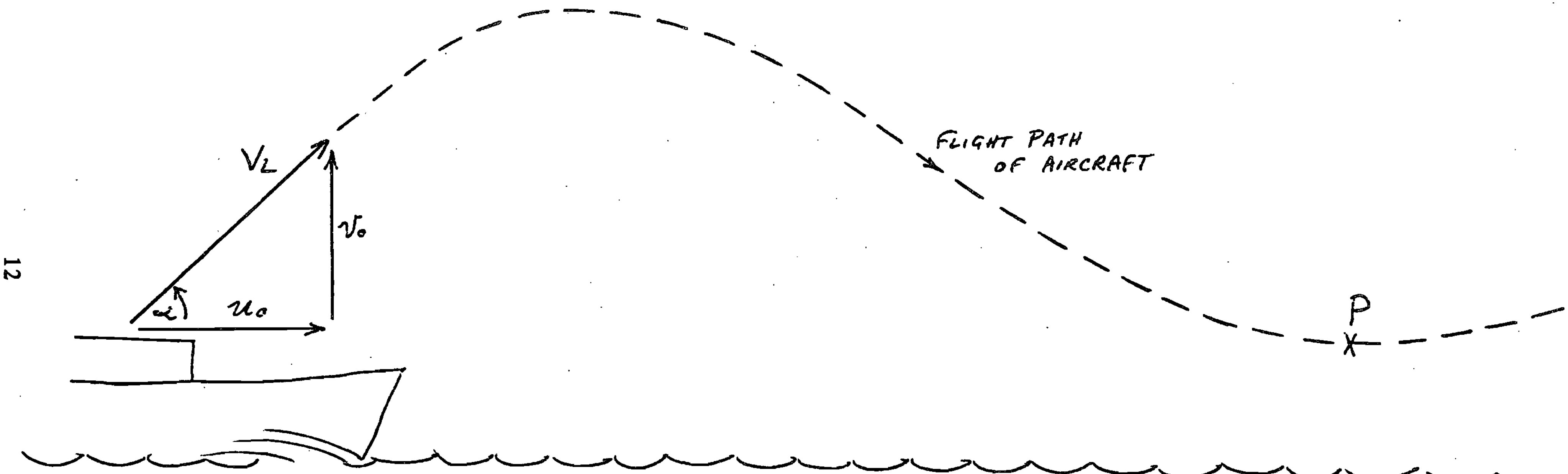
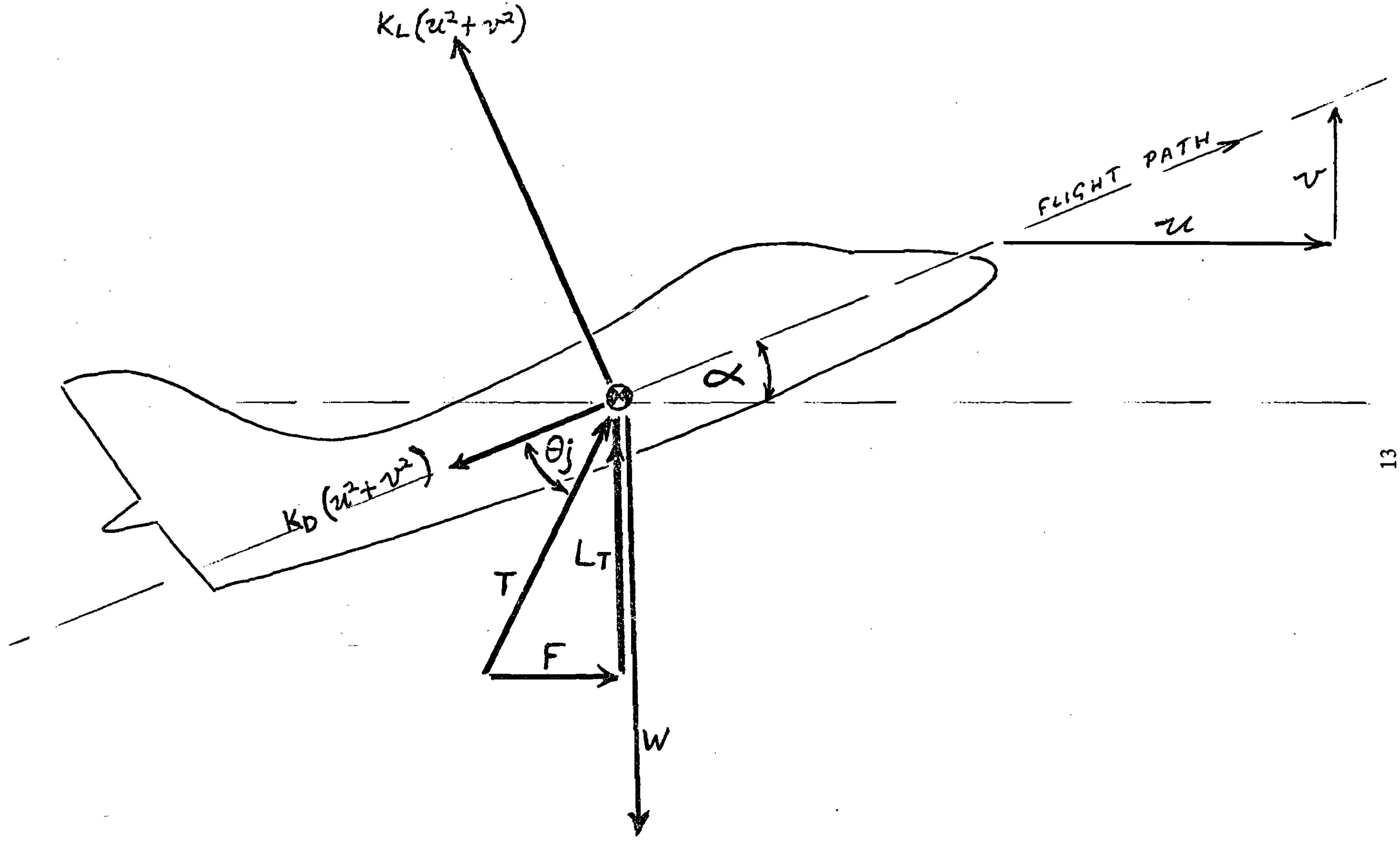


FIG 2 FORCES ACTING ON AIRCRAFT



The symbols have the following definitions:

T = Engine thrust

θj = Thrust deflection angle relative to airframe.

α = Angle between flight path of the aircraft and horizontal.

θ_T = $(\theta j + \alpha)$ = Thrust deflection angle relative to horizon.

L_T = $T \sin \theta_T$ = Vertical component of thrust.

F = $T \cos \theta_T$ = Horizontal component of thrust.

W = Weight of aircraft.

K_L = Lift constant $(= \frac{1}{2}\rho S C_L)$

K_D = Drag constant $(= \frac{1}{2}\rho S C_D)$

u = Horizontal component of velocity.

v = Vertical component of velocity.

$\dot{u}$ = Horizontal acceleration.

$\dot{v}$ = Vertical acceleration.

g = Acceleration due to gravity.

t = Time from launch.

The acceleration of the aircraft may be expressed as follows:

Horizontally,

$$\frac{W\dot{u}}{g} = F - K_L(u^2+v^2)\sin\alpha - K_D(u^2+v^2)\cos\alpha$$

Vertically,

$$\frac{W\dot{v}}{g} = (L_T - W) + K_L(u^2+v^2)\cos\alpha - K_D(u^2+v^2)\sin\alpha$$

Since:

$$\operatorname{Tan}\alpha = \frac{v}{u}, \quad \operatorname{Cos}\alpha = \frac{u}{\sqrt{u^2+v^2}}, \quad \operatorname{Sin}\alpha = \frac{v}{\sqrt{u^2+v^2}}$$

Hence:

$$\dot{v} = \frac{g}{W}\left[L_T - W + \sqrt{u^2+v^2}\,(K_L u - K_D v)\right] \quad - \quad (1)$$

and,

$$\dot{u} = \frac{g}{W}\left[F - \sqrt{u^2+v^2}\,(K_L v + K_D u)\right] \quad - \quad (2)$$

These equations are not conveniently integrable so for initial exploratory work some results were obtained by manual integration methods with the aid of a desk calculator. An explanation of the method used together with some results is in Appendix 1.

1.3.2 The only fixed wing VTOL aircraft in operational use is the Harrier and it is thus appropriate to choose as a basis for calculation a hypothetical aircraft having characteristics broadly similar to the Harrier for later comparison with actual Harrier performance. The basic data are:

Net installed thrust, T = 19,200 Lb

Maximum weight, W = about 25,000 Lb

Lift to drag ratio in take-off conditions with deflected thrust, 5:1

It is also assumed that the aircraft can sustain unaccelerated level flight at W = 22,000 Lb and $\theta_T = 60^\circ$ at a speed of 120 knots (200 ft/sec).

From this it can be calculated that

$K_L = 0.1343$ and $K_D = 0.0269$. These values have been used in all calculations unless otherwise stated.

1.3.3 Having established the order of launch velocity required and the time taken for completion of the semi-ballistic trajectory, further results were obtained by applying the manual integration method to a WANG 3300 digital computer using very small time intervals for successive calculations. For convenience in using the teleprinter the symbols used were amended as follows:

T = time interval for successive calculations.

W = time from launch.

V = vertical component of velocity.

U = horizontal component of velocity.

L = vertical component of thrust.

F = horizontal component of thrust.

Z = vertical acceleration.

A = horizontal acceleration.

Y = vertical distance from launch point.

B = angle between flight path of aircraft and horizontal.

C = thrust deflection angle relative to horizon.

A typical program written in BASIC for a launch at a weight of 25,000 Lb and a launch velocity of 106.5 ft/sec with $\theta j = 55^{\circ}$ is as follows:

```
1    REM PROGRAM
5    Y = 0: V = 53.25: U = 92: F = 1672: L = 19100
6    N = 1
9    T = 0.1
10   P = U↑2 + V↑2
15   Q = 0.1343*U - 0.0296*V
20   R = 0.1343*V + 0.0269*U
25   Z = 0.00129*(L-25000+Q*SQR(P))
30   A = 0.00129*(F-R*SQR(P))
40   V = V+Z*T
45   U = U+A*T
50   B = ATN(V/U)
55   C = 0.9599+B
60   F = 19200*COS(C)
65   L = 19200*SIN(C)
66   Y = Y+V*T
67   W = N*T
68   N = N+1
69   IF W = INT(W) THEN 75
70   GO TO 80
75   PRINT "W=";W;  "Y=";Y;  "C=";C
80   IF W < 30 THEN 9
```

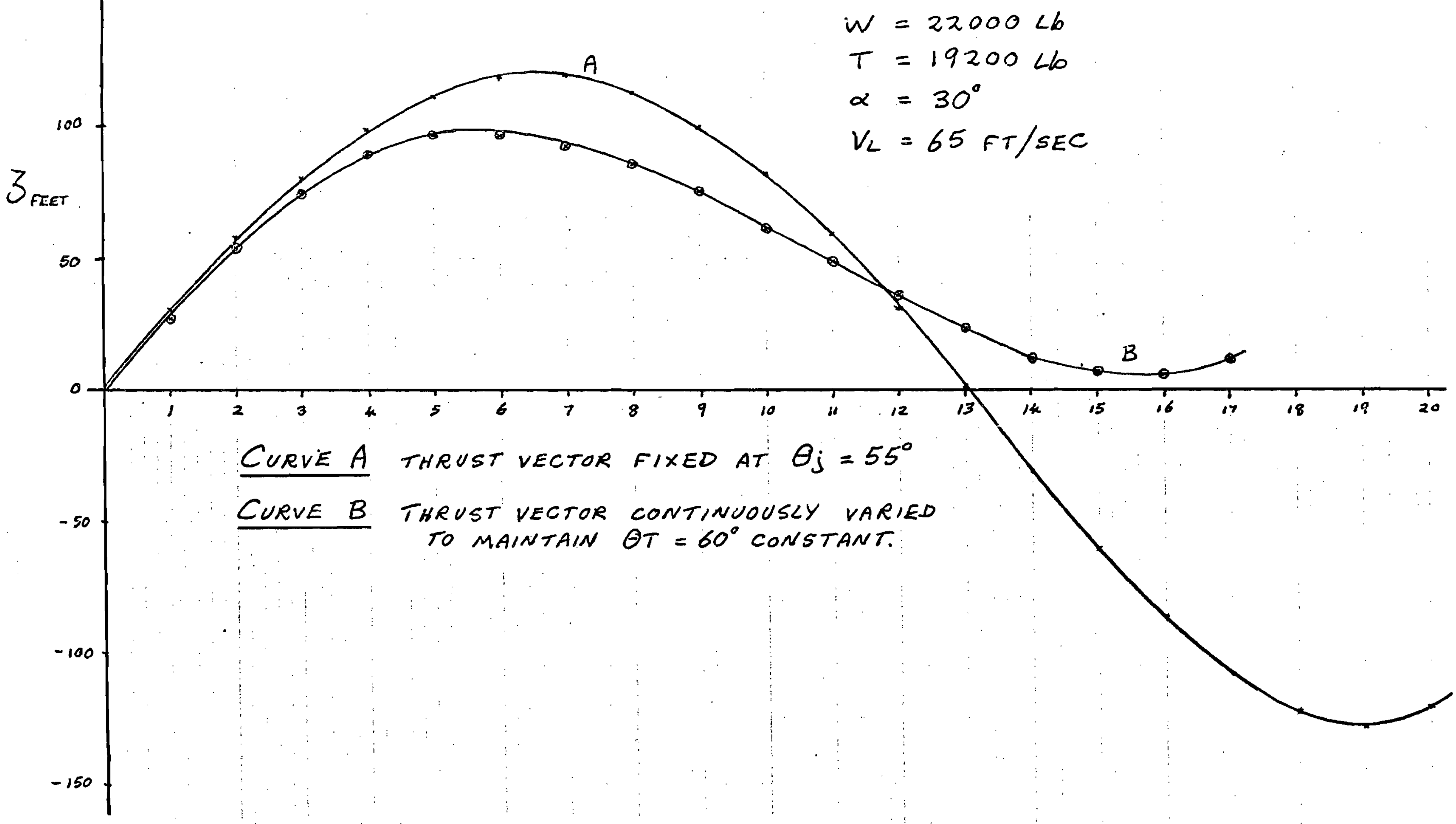
1.3.4 It soon became apparent that keeping θj fixed during the semi-ballistic flight resulted in a less than optimum performance and that an improvement could be obtained by continuously varying θj as the aircraft attitude changed, to maintain a constant value of θ_T. The reason for this is that, with θj constant θ_T has high values during the first part of the semi-ballistic flight which results in a high trajectory but little or no gain in velocity and hence aerodynamic lift, before the peak of the trajectory is reached. After the peak relatively high downward velocities are attained before the increased horizontal acceleration provides a counter-balancing aerodynamic lift.

If θ_T is kept constant the horizontal acceleration is continuous and virtually constant from the moment of launch and aerodynamic lift keeps vertical velocities relatively small.

1.3.5 The difference between the two thrust variation techniques is illustrated in Figs 3, 4 and 5. If θj is kept constant an 11% increase in launch velocity is required and the time in semi-ballistic flight is increased by 20%. With increase in weight the change in launch velocity is slightly increased and the time difference slightly decreased. It should also be noted that in the example the fixed value of $\theta j(55^{o})$ has been chosen so that the average value of θ_T is 60^{o} in order to give a direct comparison. The reduction of launch velocity is valuable and the reduction of time in semi-ballistic flight is also an important feature since this is time with the engine at full power so that engine life and fuel consumption are affected. For these reasons it is concluded that variation of θj to maintain constant θ_T during the period of semi-ballistic flight results in significant and worthwhile improvements in performance.

FIG 3

COMPARISON OF THRUST VARIATION TECHNIQUES



COMPARISON OF THRUST VARIATION TECHNIQUES

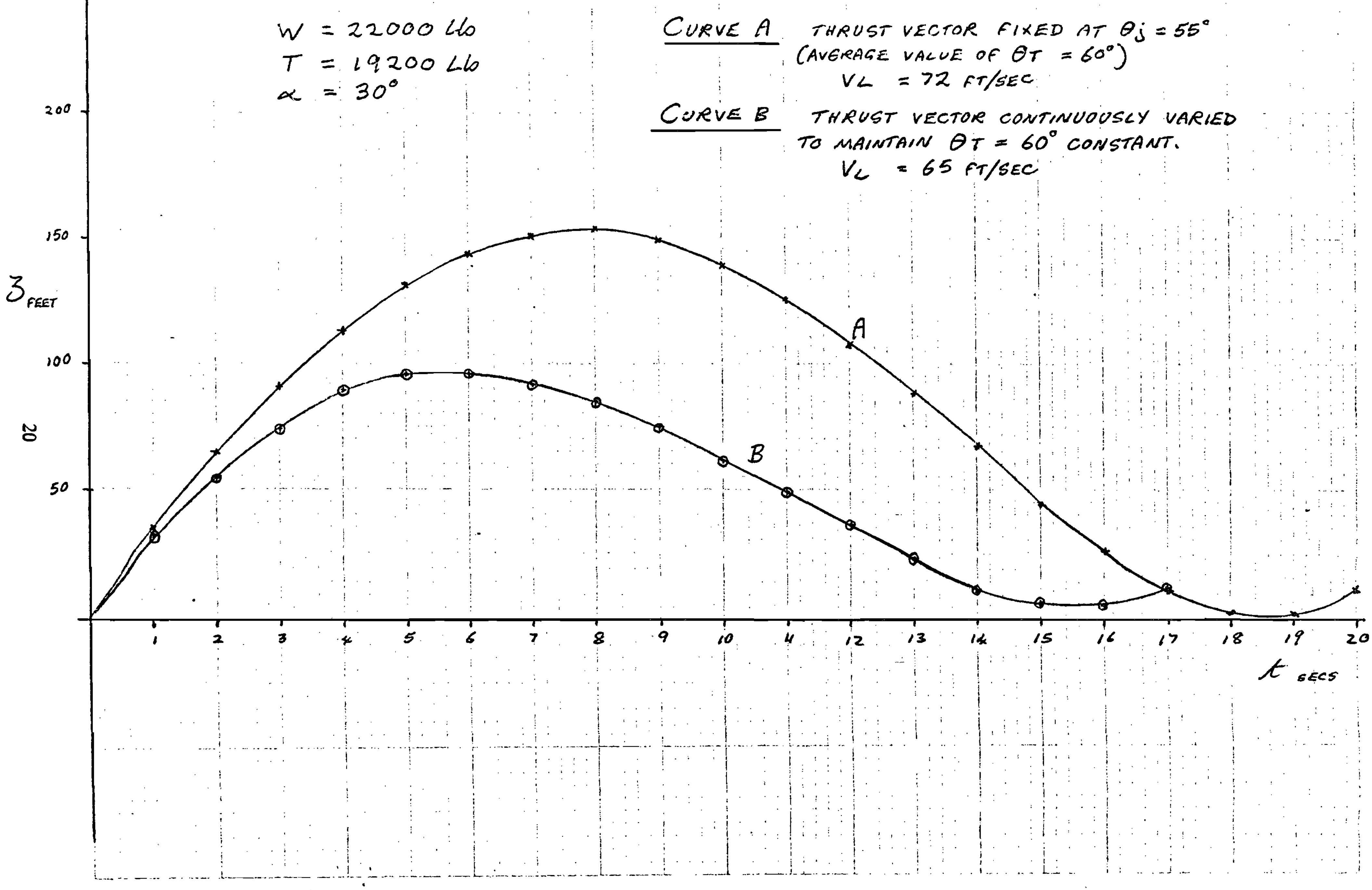


FIG 5

COMPARISON OF THRUST VARIATION TECHNIQUES

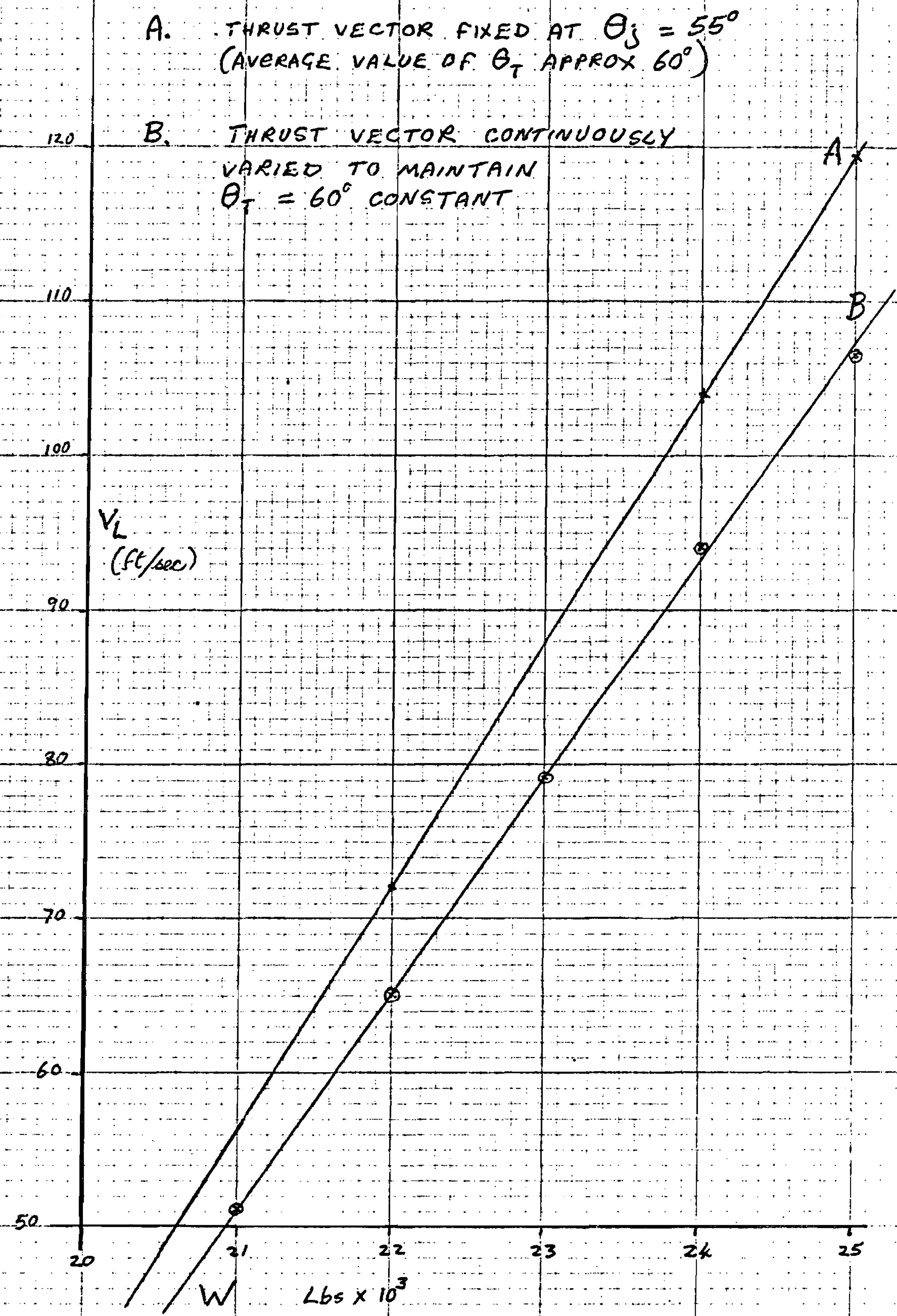
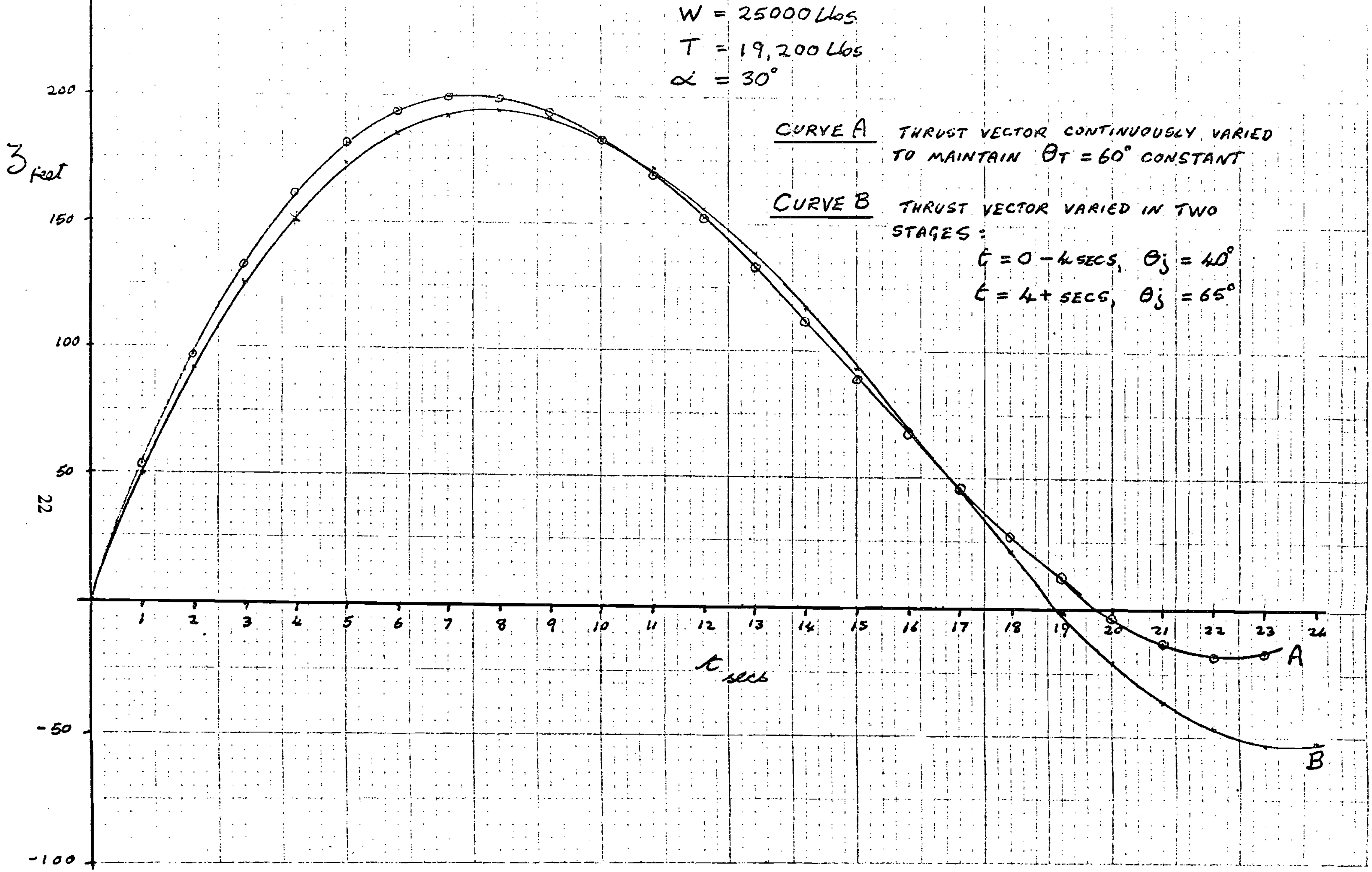


FIG 6



W = 25000 Lbs
T = 19,200 Lbs
α = 30°
CURVE A THRUST VECTOR CONTINUOUSLY VARIED TO MAINTAIN θT = 60° CONSTANT
CURVE B THRUST VECTOR VARIED IN TWO STAGES:
t = 0 - 4 secs, θj = 40°
t = 4+ secs, θj = 65°
z feet
200
150
100
50
-50
-100
t secs
1 2 3 4 5 6 7 8 9 10 11 12 13 14 15 16 17 18 19 20 21 22 23 24
A
B

1.3.6 To achieve a constant value of θ_T, θj must either be varied continuously by means of a servo system connected to a horizontal reference or be varied in stages by the pilot. The first method introduces some complexity although it is not difficult technically. The second method requires the pilot to make accurate selections with a fair degree of accuracy in timing. Fig 6 compares continuous variation of θj with a two stage variation.

As far as the trajectory is concerned there is no significant difference between the two methods. The two stage method does however place an additional workload on the pilot at a time when accurate flying is essential and it may well be that this additional load is unacceptable.

It is considered that the advantages of varying θj justify the complexity and expense involved. All calculations therefore assume that this is the method used unless otherwise stated.

1.3.7 The use of constant θ_T makes it possible to simplify the equations of motion subject to the following assumptions:

(i) That the horizontal acceleration is uniform. For practical purposes this is a valid assumption for the speeds under consideration if a value of 0.9 (Acceleration neglecting drag) is used.

(ii) That the vertical velocity, v is small compared with the horizontal velocity, u. This is certainly not true during the first few seconds after launch but is valid for the rest of the period of semi-ballistic flight when aerodynamic effects have become significant.

Applying the above assumptions:

where, a = horizontal acceleration

z = vertical distance from launch point

then,
$$\frac{d^2z}{dt^2} = \frac{g}{W}\left(L_T - W + K_L u^2\right)$$

$$u = u_o + at$$

$$\therefore\ u^2 = u_o^2 + 2u_o at + a^2t^2$$

$$\frac{d^2z}{dt^2} = \frac{gL_T}{W} - g + \frac{gK_L u_o^2}{W} + \frac{2gK_L u_o at}{W} + \frac{gK_L a^2t^2}{W}$$

Integrating:

$$\frac{dz}{dt} = v_o + \frac{g}{W}\left(L_T t - Wt + K_L u_o^2 t + K_L u_o a t^2 + \frac{K_L a^2 t^3}{3}\right) \qquad - \ (3)$$

Integrating again:

$$z = v_o t + \frac{g}{W}\left(\frac{L_T t^2}{2} - \frac{Wt^2}{2} + \frac{K_L u_o^2 t^2}{2} + \frac{K_L u_o a t^3}{3} + \frac{K_L a^2 t^4}{12}\right) \qquad - \ (4)$$

Using these equations the initial launch conditions may be calculated as shown at Appendix 2.

A comparison of results achieved by manual integration of equation (1) and the use of simplified equations (3) and (4) is shown in Fig 7.

1.4 Determination of Trajectory by Analogue Computer

1.4.1 A thorough study of the effects of the important variables upon the trajectory requires the calculation of a large number of trajectories. In addition it would be useful to check the figures obtained by manual integration methods and by the use of equations (3) and (4) which are the results of approximations. The problem was therefore applied to an EAI 580 Analogue Computer with solutions presented as an oscilloscope display or by pen recorder as required.

Due to limitations of the computer only the trajectories involving constant θ_T could be produced in this way.

As a further cross-check the equations of motion were formulated with reference to the aircraft flight path instead of the horizontal and vertical axes used hitherto. Fig 8 shows the forces acting upon the aircraft and its motion, the symbols, where not previously defined are as follows:

v = velocity along flight path

γ = angle of flight path to horizontal

$\dot{v}$ = acceleration along flight path

$\dot{\gamma}$ = rate of change of γ

$\dot{z}$ = vertical velocity

The motion of the aircraft along the flight path may be expressed as follows:

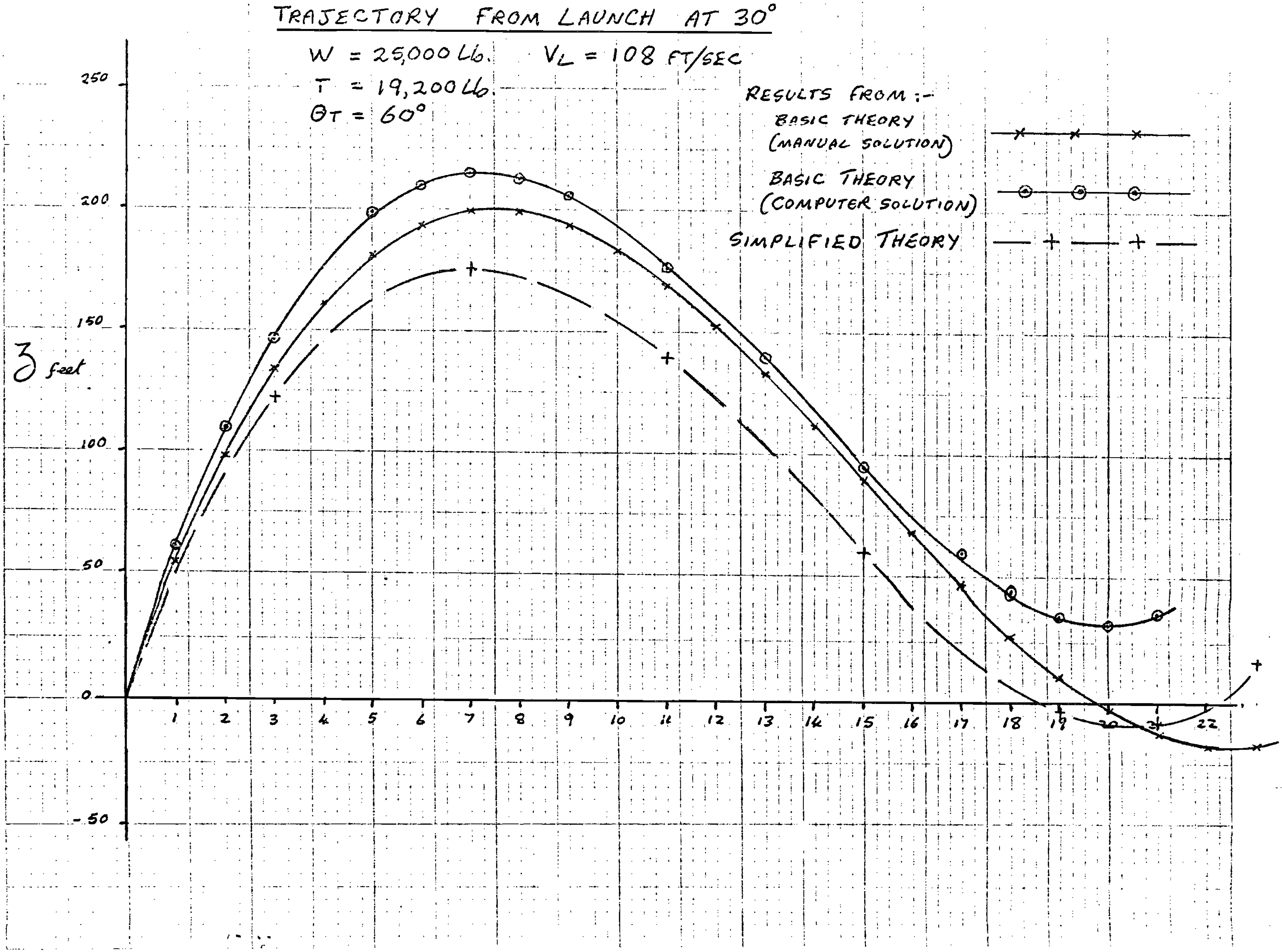
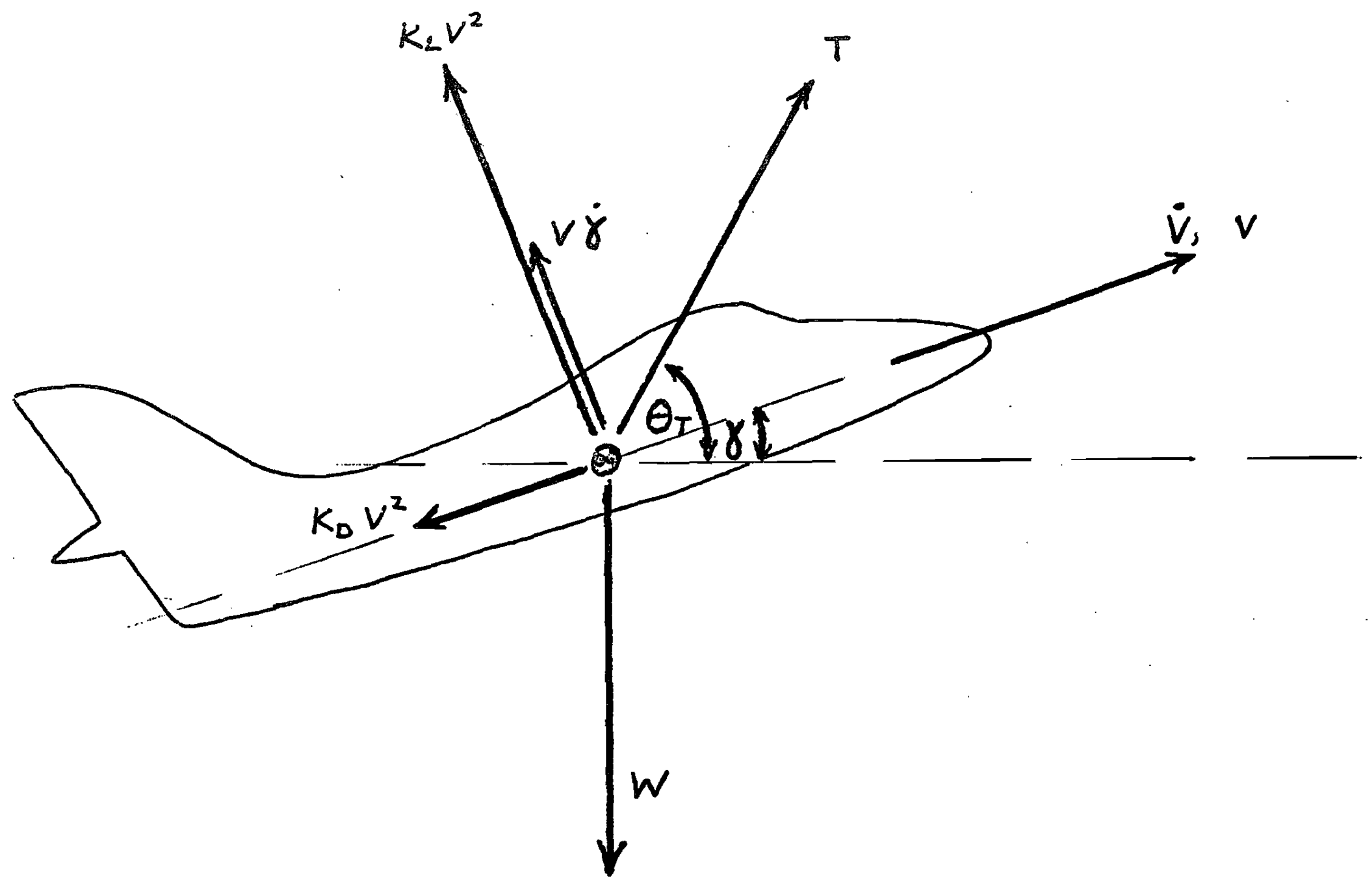

TRAJECTORY FROM LAUNCH AT 30°
W = 25,000 Lb.
V_L = 108 FT/SEC
T = 19,200 Lb.
θ_T = 60°
RESULTS FROM :-
BASIC THEORY (MANUAL SOLUTION)
BASIC THEORY (COMPUTER SOLUTION)
SIMPLIFIED THEORY
z feet
250
200
150
100
50
0
-50
1 2 3 4 5 6 7 8 9 10 11 12 13 14 15 16 17 18 19 20 21 22

FIG 8 APPLIED FORCES AND MOTION OF AIRCRAFT



$$m\dot{v} = T\cos(\theta_T - \gamma) - W\sin\gamma - K_D v^2$$

$$\therefore\ 0 = -m\dot{v} + T\cos\theta_T,\ \cos\gamma - (W - T\sin\theta_T)\sin\gamma - K_D v^2 \qquad - (5)$$

$$mv\dot{\gamma} = K_L v^2 - W\cos\gamma + T\sin(\theta_T - \gamma)$$

$$\therefore\ 0 = -mv\dot{\gamma} - (W - T\sin\theta_T)\cos\gamma - T\cos\theta_T\sin\gamma + K_L v^2 \qquad - (6)$$

$$\dot{z} = v\sin\gamma$$

$$\therefore\ 0 = -\dot{z} + v\sin\gamma \qquad - (7)$$

These equations were used in obtaining the computer solutions for the aircraft trajectory.

The program and scalings with some typical results are at Appendix 3.

1.4.2 The computer results matched well with those obtained by manual integration and by the use of equations (3) and (4) as is illustrated in Fig 7. In all cases the computed result was slightly more favourable, ie requiring a lower launch velocity, v_L.

It therefore seems reasonable to conclude that results obtained by any of the three methods are interchangeable for most purposes. This is valuable because manual methods have to be used, for instance if θj is fixed or if conditions are changed during the course of semi-ballistic flight.

1.5 Optimum launch angle

This was determined by finding the launch velocity required for zero height loss (ie point P at launch height) at each launch angle, for two weights, 22,000 Lb and 25,000 Lb. The results

Fig 9

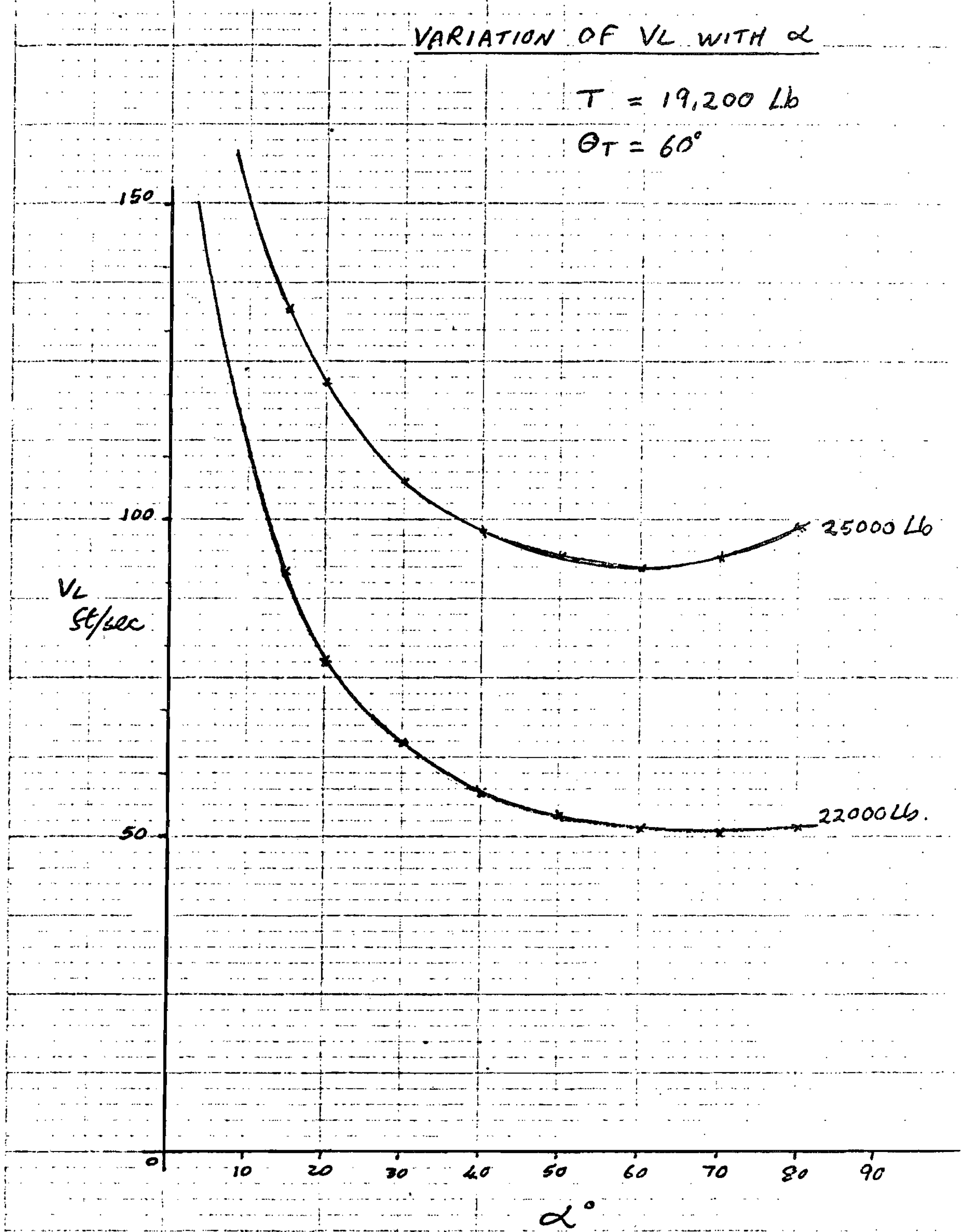

VARIATION OF VL WITH α
T = 19,200 Lb
θT = 60°
150
100
50
0
VL
ft/sec
25000 Lb
22000 Lb.
10
20
30
40
50
60
70
80
90
α°

Fig 10.

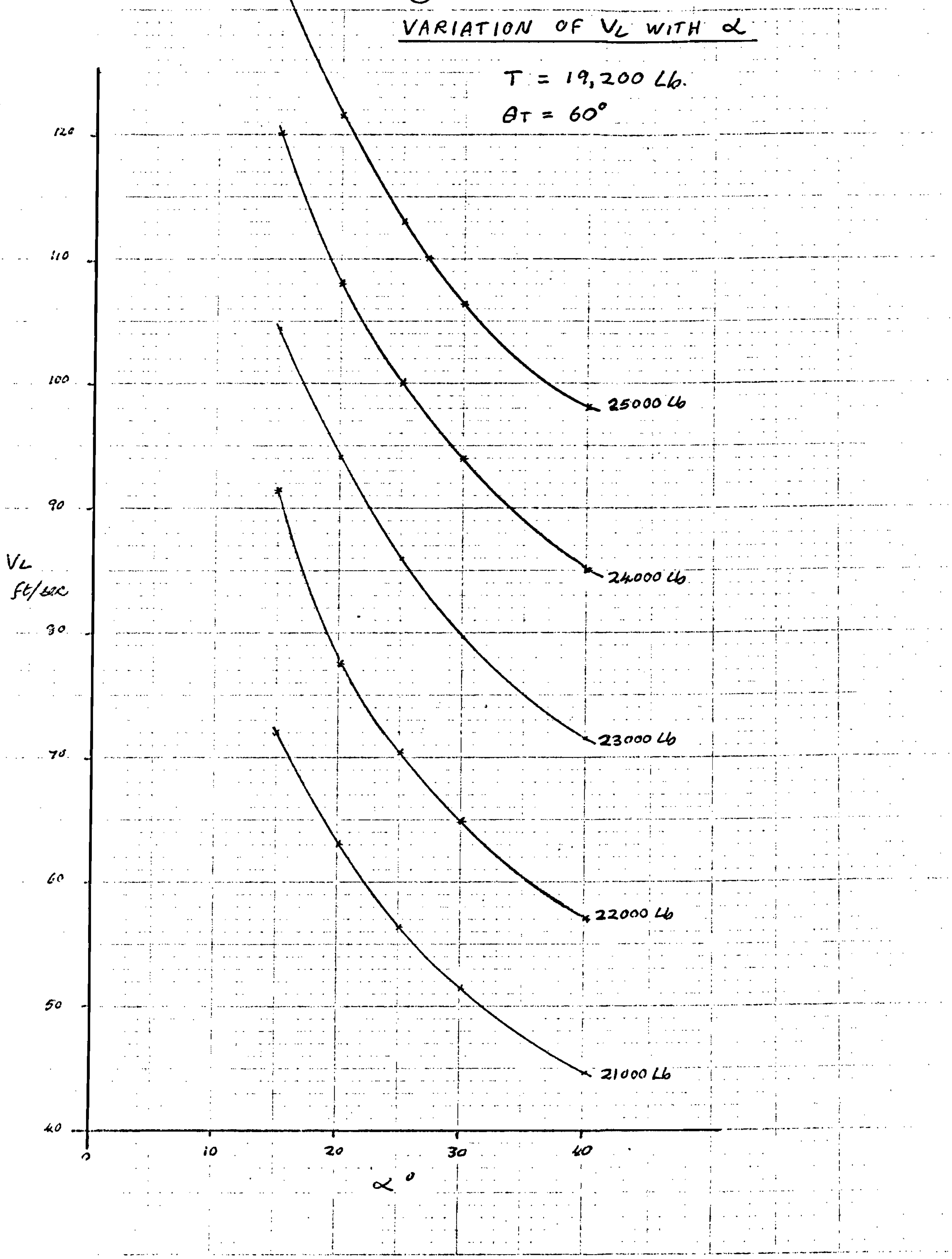

VARIATION OF V_L WITH α
T = 19,200 Lb.
θ_T = 60°
25000 Lb
24000 Lb
23000 Lb
22000 Lb
21000 Lb
V_L
ft/sec
120
110
100
90
80
70
60
50
40
0
10
20
30
40
α°

Fig 11

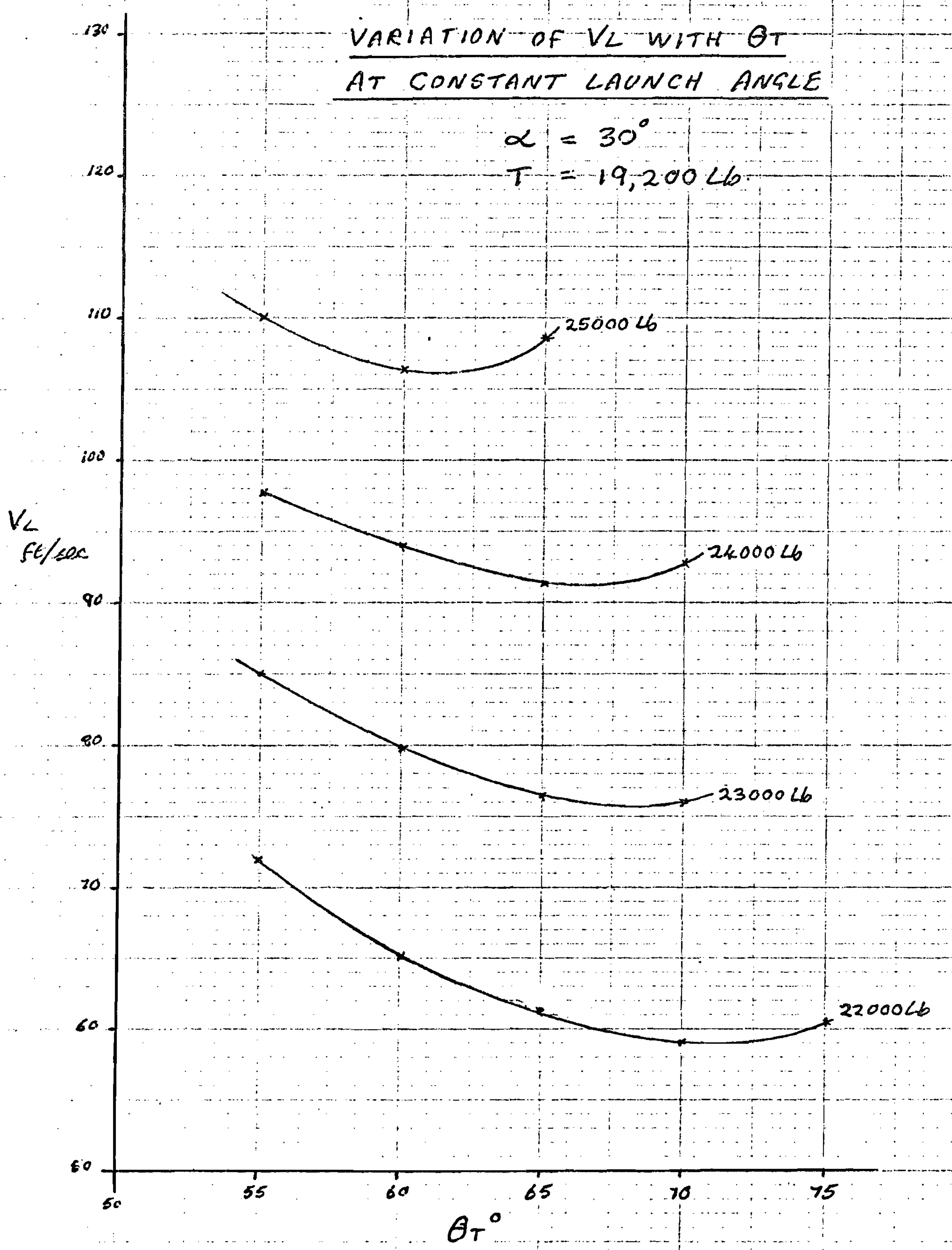
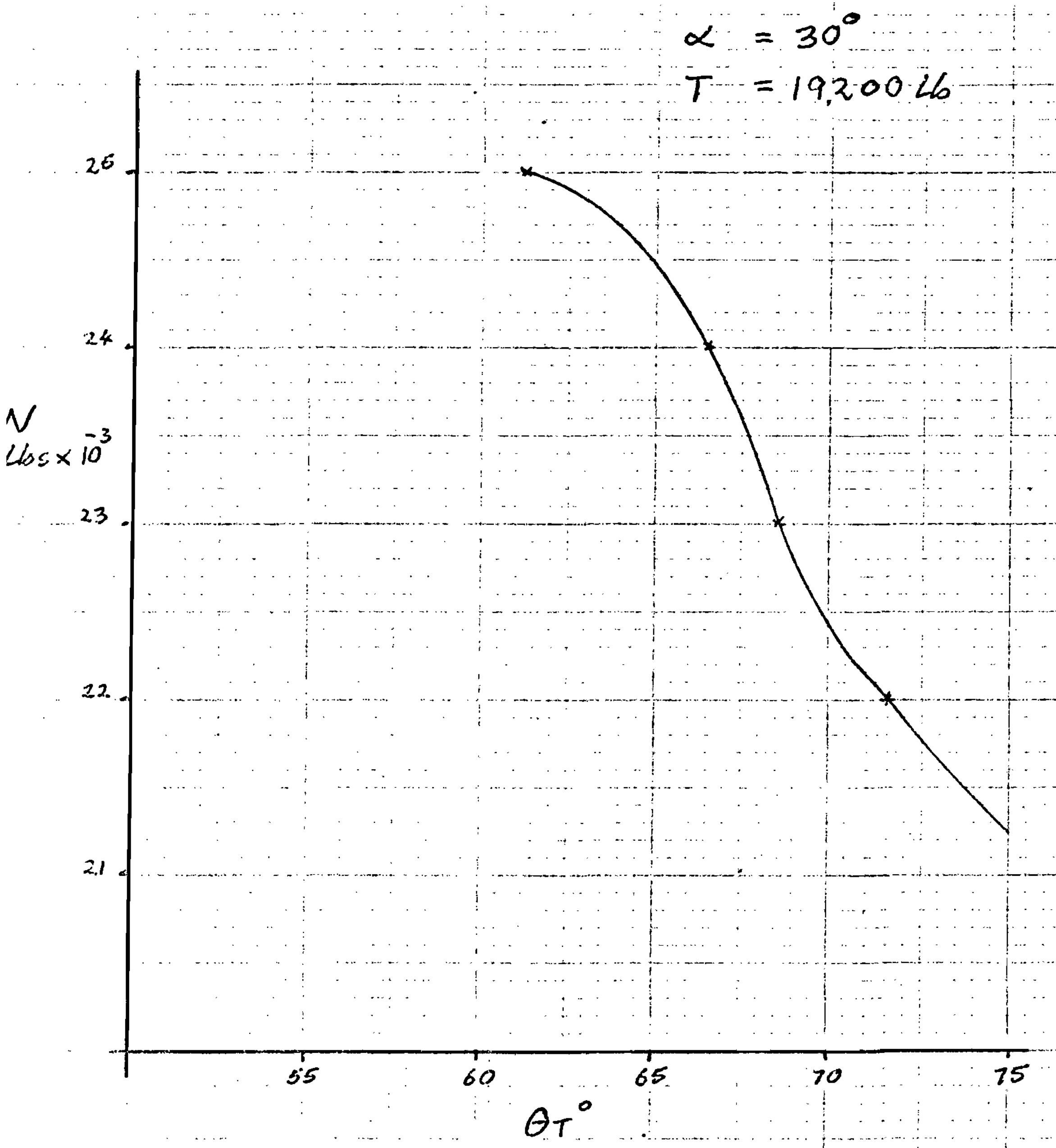

VARIATION OF VL WITH θT
AT CONSTANT LAUNCH ANGLE
α = 30°
T = 19,200 Lb
VL ft/sec
θT°
25000 Lb
24000 Lb
23000 Lb
22000 Lb
130
120
110
100
90
80
70
60
50
50
55
60
65
70
75

FIG 12

OPTIMUM VALUES OF θ_T
AT CONSTANT LAUNCH ANGLE



are shown graphically in Fig 9.

Clearly there is no advantage in increasing the launch angle beyond 60° and the greatest reduction in v_L occurs between $\alpha = 0^{\circ}$ and $\alpha = 30^{\circ}$.

30° is therefore probably the optimum launch angle although if other factors permit it may be worthwhile increasing the angle to 40° in some cases.

Launch velocities for a range of weights at useful values of α are plotted in Fig 10.

1.6 <u>Optimum Thrust Deflection Angle</u>

The effect of changing θ_T at constant launch angle was found by a similar method to that of 1.5 above. The results are shown in Figs 11 and 12. As might be expected there is an optimum value for each weight, most sensitivity being shown at the higher weights.

For most practical purposes it would probably be advantageous to fix a value of θ_T of 60° for use with any launching system designed around a maximum launch weight of 25,000 Lbs. For the majority of launches this reduces the time spent at full power and provides a built-in margin to cater for errors and emergencies.

1.7 <u>Variation of Launch Velocity</u>

Launch velocity was increased in two increments of 5 ft/sec at constant α and θ_T from the value of v_L required for zero height loss. Results are shown in Fig 13. It is clear that an increment of 10 ft/sec should be adequate to cope with most inaccuracies in launch conditions. It also highlights the importance of achieving the correct launch velocity although in this respect this method of launching is no more sensitive than conventional methods.

In this diagram Z_P is the height of point P above the launch point.

1.8 <u>The Effect of Wind</u>

Natural wind or wind caused by the motion of the ship is added to u_o if the launch is into wind and has the effect of increasing the weight which may be launched or of permitting a decrease in v_L for the same weight.

The method used to calculate wind effects is shown in Appendix 4 and results in Fig 14.

1.9 <u>Effect of Delaying Thrust Deflection</u>

One of the methods of launching into semi-ballistic flight, which will be considered later, requires the aircraft to accelerate under undeflected thrust and then to deflect the thrust to a pre-selected value of θ_T at the instant of leaving the ship. Deflecting the thrust is not an instantaneous process and in practise the pilot may, for various reasons, delay selecting the thrust deflection. The first part of the semi-ballistic trajectory may therefore be at a value of θ_T well below the optimum. This question will be examined in more detail later. For present purposes it is assumed that delay in pilot reaction will be a

Fig 13

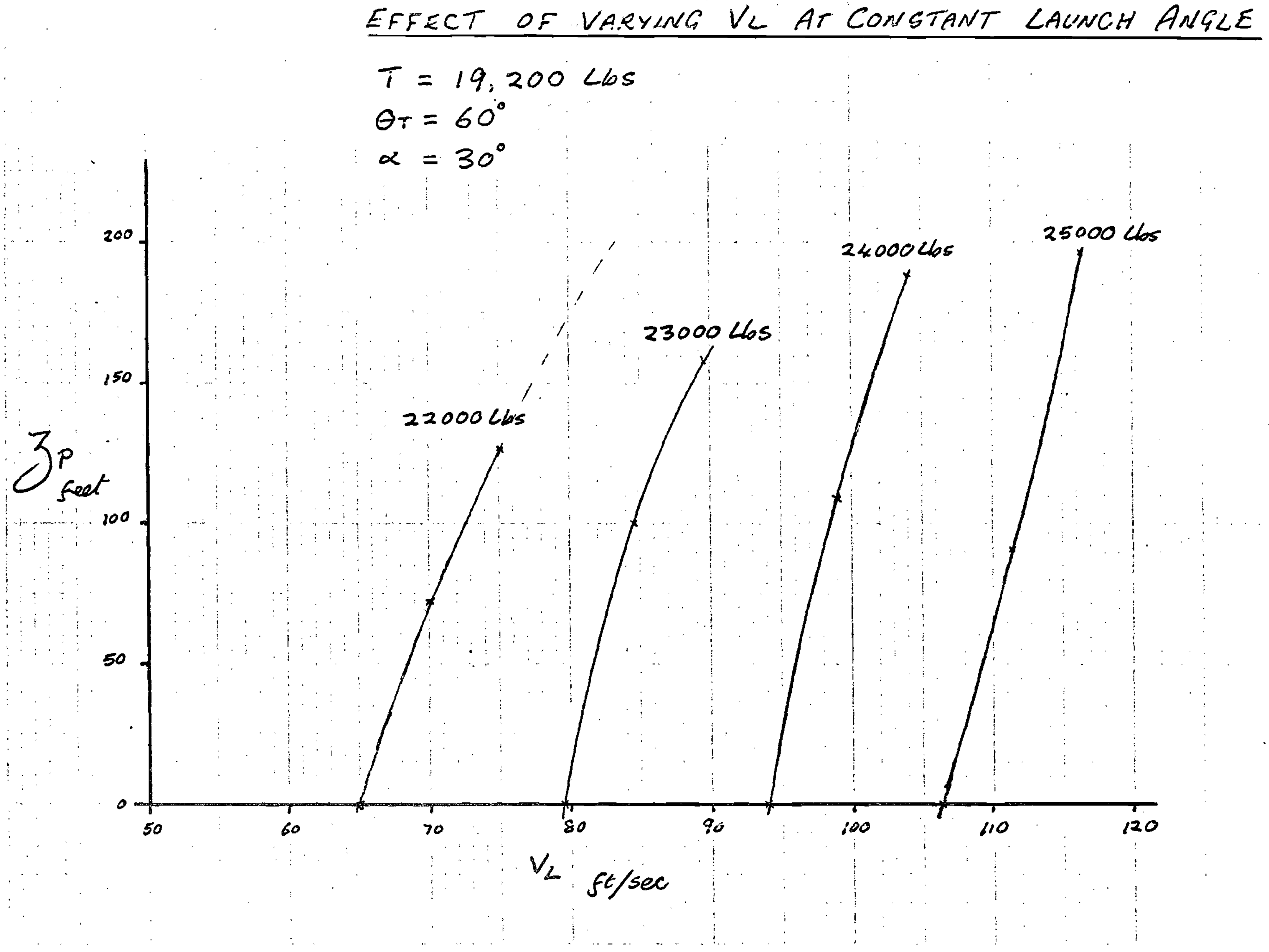

EFFECT OF VARYING VL AT CONSTANT LAUNCH ANGLE
T = 19,200 Lbs
θT = 60°
α = 30°
22000 Lbs
23000 Lbs
24000 Lbs
25000 Lbs
Zp feet
0
50
100
150
200
50
60
70
80
90
100
110
120
VL ft/sec

FIG 14

VARIATION OF V_L WITH WIND OVER DECK

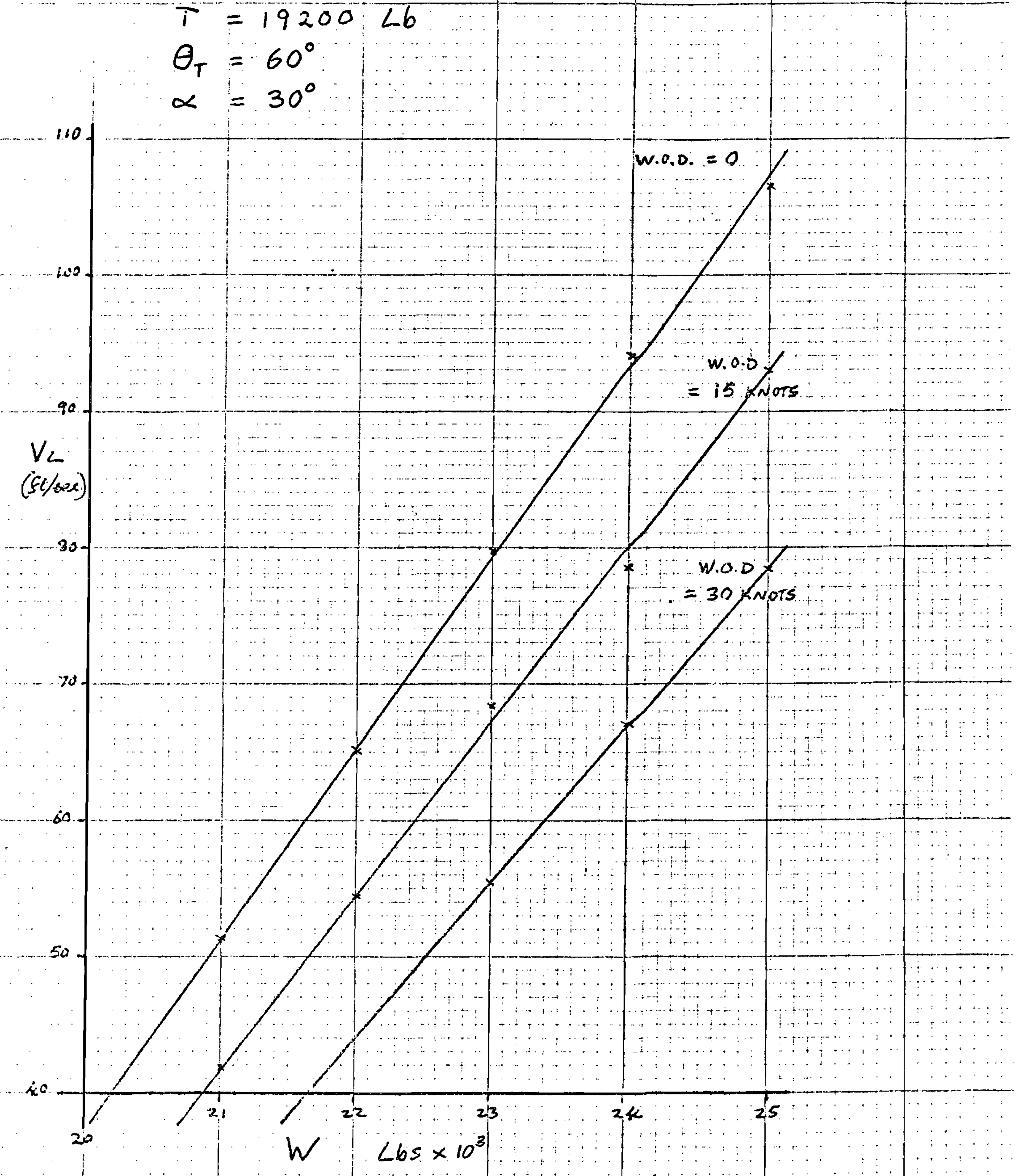
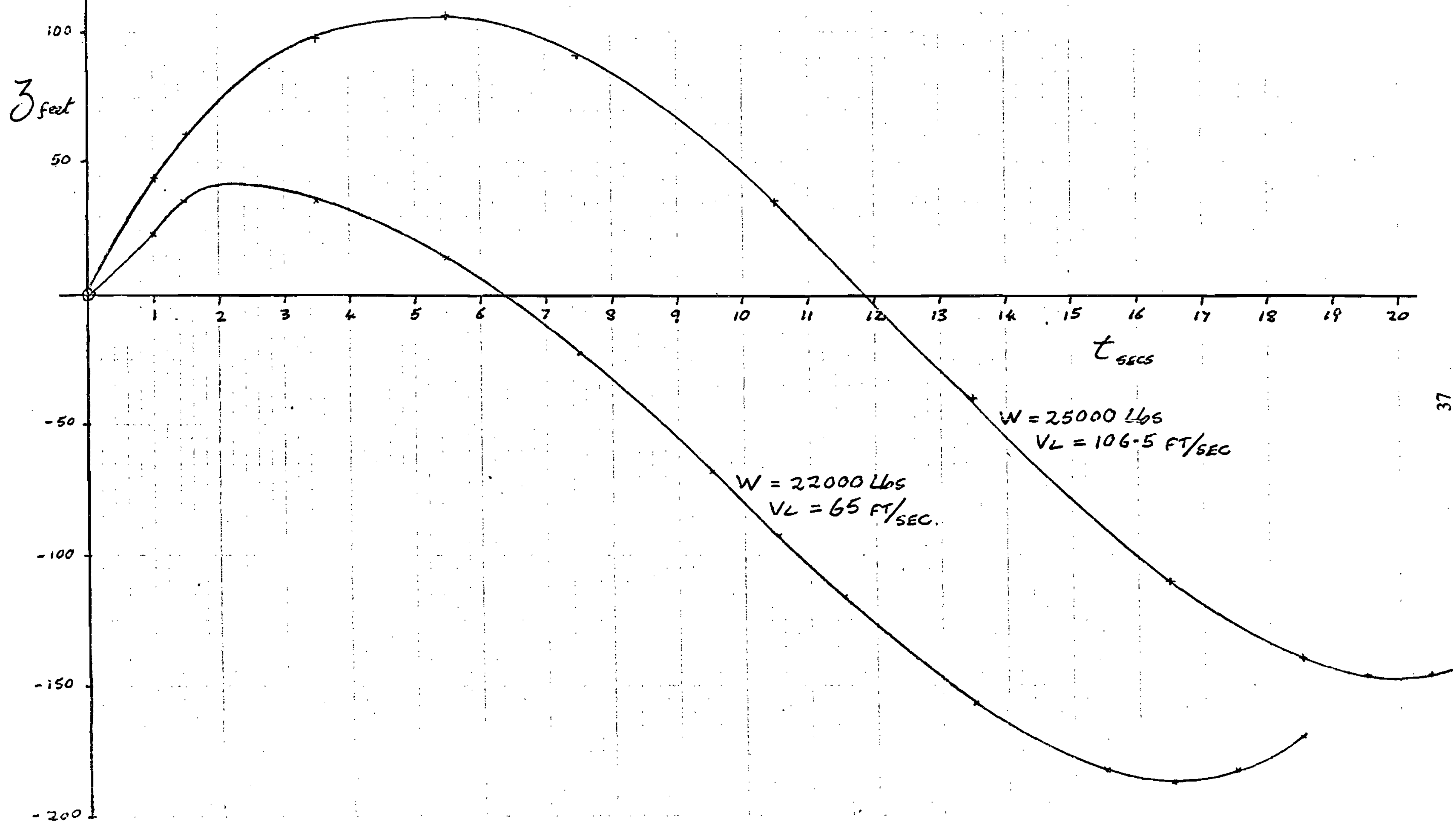


FIG 15 EFFECT OF DELAYING NOZZLE ROTATION

$\alpha = 30^\circ$, $T = 19{,}200$ lbs. $\theta_j = 0^\circ$ UNTIL $t = 1$ SEC. NOZZLES THEN ROTATED TO GIVE $\theta_T = 60^\circ$ AT $t = 1.5$.



maximum of 1 second and that the jet nozzles will reach the selected value of θ_T in 0.5 seconds.

The resultant trajectories have been calculated for weights of 22,000 and 25,000 Lbs. Method of calculation is shown in Appendix 5 and results in Fig 15. The effect is more marked at the lower weights resulting in a nett height loss of 187 ft at 22,000 Lbs and 148 ft at 25,000 Lbs. From any ship these launches would be disastrous.

To ensure a successful outcome under these conditions requires an increase in v_L, the amount of which can be found by reference to Fig 13. Thus at 22,000 lbs, v_L must be increased by about 17 ft/sec and at 25,000 Lbs by about 7 ft/sec.

1.10 Aircraft Incidence at Instant of Launch

1.10.1 In zero-wind conditions the incidence of the aircraft at the start of its semi-ballistic flight will be approximately zero (depending on undercarriage characteristics) and it may therefore require a small initial rotation in the nose-up sense to achieve an optimum incidence of about 10-15°. In practise this is a somewhat unrealistic condition. In most cases it is likely that there will be a wind over the deck of at least 15 knots. The incidence may be deduced in this case by simple resolution of velocities as shown in Fig 16.

1.10.2 At the lighter weights and higher windspeeds it is likely that the aircraft incidence may be close to or exceeding the stalling angle, particularly at launch angles greater than 30°. However this is a very transitory phase lasting one or two seconds at most and provided the aircraft does not pitch or roll uncontrollably it should have little effect on the trajectory. Under the conditions likely to result in stalling incidence ie light weight, the launch velocities are low and consequently aerodynamic forces are small. The aircraft is being controlled by its jet reaction controls whose power is independent of airspeed. Provided that the stall does not result in pitching or rolling moments exceeding those of the reaction controls its effect need not be serious. Stalling incidence may be avoided entirely, of course, by increasing v_L above the minimum required.

1.11 Aircraft Rotation in Pitch

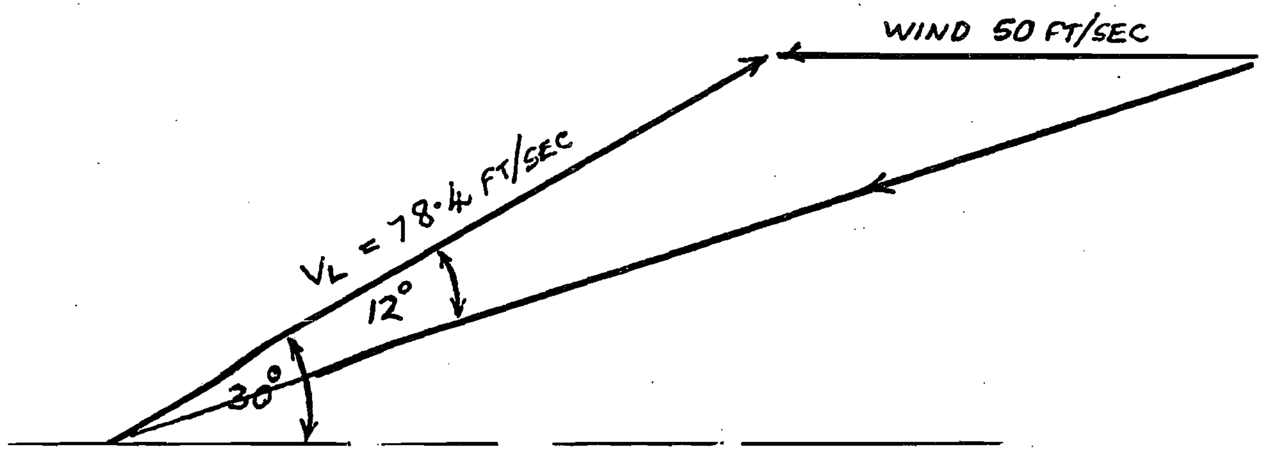
Immediately after launch the aircraft attitude in space may be 30° or more nose-up. At the peak of its trajectory it must be in an attitude of about 10° nose up.
Having passed the peak of trajectory the aircraft attitude is unlikely to greatly exceed 5° nose down. The greatest rates of rotation are therefore required early in the trajectory when control is entirely dependent on the jet reaction controls. The inertia of the aircraft in the pitching plane is large and rates of rotation in excess of 10° per second may be difficult to control. It is

FIG 16

INCIDENCE AT LAUNCH

$W = 25000$ lbs

W.O.D. $= 30$ KNOTS $= 50$ FT/SEC



desirable therefore not to exceed about half this value. Since the peak of the aircrafts trajectory occurs between 4 and 8 seconds after launch the pitching rates are well within this limit.

1.12 Instrumentation

1.12.1 The semi-ballistic technique will introduce some new experiences and sensations to the pilot, some more akin to space flight than to aerodynamic flight.
Following a conventional take-off or a horizontal catapult launch, speed and height normally increase progressively and the aircraft's situation can be continuously monitored by visual reference or by instruments and either will provide instant warning of the development of a dangerous situation. To a large extent this is true also of a transition of a VTOL aircraft from hovering to conventional flight.

1.12.2 Following a launch into semi-ballistic flight the pilot must achieve and maintain accurately a given angle of incidence. This is a routine function so far as pilots of fixed wing naval aircraft are concerned and presents no unusual problems. The subjective visual impressions however are quite unusual. Initially the impression will be virtually one of hovering - but at an apparently highly dangerous angle. The attitude soon returns to more normal values but then the aircraft begins to descend and the impression of increasing speed is reinforced by the decreasing height above the sea. Instruments confirm these impressions but give no clear indication as to whether the situation is safe or not. The position is analagous to that of the pilot of a space craft which is about to re-enter the earth's atmosphere. In fact, of course, if the launch angle and speed were correct and there have been no serious excursions from the required angle of incidence, all of which can be known to the pilot there should be no cause for concern. It is probable that, with experience, pilots will soon develop the ability to detect any unusual development and there is the great advantage in this method of launch that the time taken and hence time available for escape is very much greater than in conventional take-offs. All the same it is possible that the pilot will require some unequivocal indication as to whether or not the outcome of the launch is likely to be a happy one. This would cover, for instance, the case of an undetected power loss.

1.12.3 Ideally, perhaps, the indication should come from a continuous monitoring of the horizontal and vertical accelerations. Any serious discrepancy will, however, very soon show up in terms of airspeed or vertical velocity. Any given point on the trajectory at a given weight should show a certain value of airspeed and vertical speed. The third parameter required is either time from launch or height. The height measurement would have to be by radio altimeter whose accuracy would be affected to some extent by sea state. Time from launch could be provided by a simple and reliable mechanism and is therefore probably the best choice for the third parameter.

The instrument required would therefore have a pilot's input, before launch, of aircraft weight. Thereafter it would take information from the ASI, VSI and a clock. The presentation of its correlated information could take the form of some kind of graduated scale indicating the margin of safety in hand or a GO/NO GO indication by a green or red light for example. The latter would probably be quite satisfactory since the only remedial action open to the pilot, assuming his flying has been accurate, is a possible increase in θ_T.

1.13 The foregoing work has shown that by launching a V/STOL aircraft at an angle to the horizontal into a ballistic trajectory very considerable reductions in launch velocity are possible. At a launch angle of 30° the launch velocity of a 22,000 Lb aircraft at a thrust/weight ratio of 0.87 is reduced from 120 kts to 39 kts - a reduction of nearly 66%. At a thrust/weight ratio of 0.77 the reduction is 58%. These figures apply to the hypothetical aircraft considered so far. The question of how closely a real aeroplane - the Harrier - is likely to match this performance must now be examined

1.14 Performance of the Harrier in Semi-Ballistic Flight

1.14.1 Two aspects must be considered. In the first place the trajectory must be determined. Then the question of stability and control of the aircraft has to be examined in case the trajectory is likely to be affected by limitations arising therefrom. In what follows, much of the data used was supplied by Hawker Siddeley Aviation Ltd.

1.14.2 The Trajectory

The same launch conditions were postulated as for the hypothetical aircraft, ie that the pilot maintains constant incidence throughout the semi-ballistic flight and that θ_T is held constant at 60°. (It should be noted that existing aircraft have no facility for maintaining θ_T constant and in the absence of this a two stage variation in nozzle angle will be necessary as detailed at 1.3.6.)

Simulated launches of Harriers were run on the HSA Ltd computer at launch speeds and weights which, for the hypothetical aircraft previously considered, would have yielded trajectories in which the flyaway point was at launch height. The computer results have been translated into the curves shown in Figs 17 and 18. These should be compared with Figs A 3.2 and 3.3.

1.14.3 In every case the aircraft falls below the original launch height and therefore requires an increase in launch velocity to bring the flyaway point up to launch height.

The increases required are:

At 22,000 Lbs	2.6 ft/sec
23,000 Lbs	1.2 ft/sec
24,000 Lbs	1.4 ft/sec
25,000 Lbs	3.6 ft/sec

The differences in launch velocity between the hypothetical aircraft and the Harrier therefore lie between 1.5 and 4% of the hypothetical prediction. This is a gratifyingly close result and the differences probably arise from the guessed values of the lift and drag constants used for the hypothetical aircraft.

From these results the effects of wind over deck were calculated as before and the plotted results are shown in Fig 19.

1.14.4 The Harrier launches were simulated only at launch angles of 30° due to shortage of computer time. The fact that the differences between the Harrier model and the hypothetical aircraft were so small gives confidence that the behaviour of the Harrier will be close to that predicted under other conditions. As a general rule an addition of 3 ft/sec to the launch velocity indicated for the hypothetical aircraft should give results for the Harrier which are sufficiently accurate for present purposes.

1.14.5 <u>Stability and Control</u>

This presents a very complex and ever-changing picture. At the moment of launch the airspeed may be very low and there will be little aerodynamic control power or damping.

FIG 17.

HARRIER PERFORMANCE PREDICTION

($\alpha = 30°$, $T = 19{,}200$ lb, $\theta_T = 60°$)

A. $W = 22000$ lb, $V_L = 65$ FT/SEC

B. $W = 23000$ lb, $V_L = 79.7$ FT/SEC

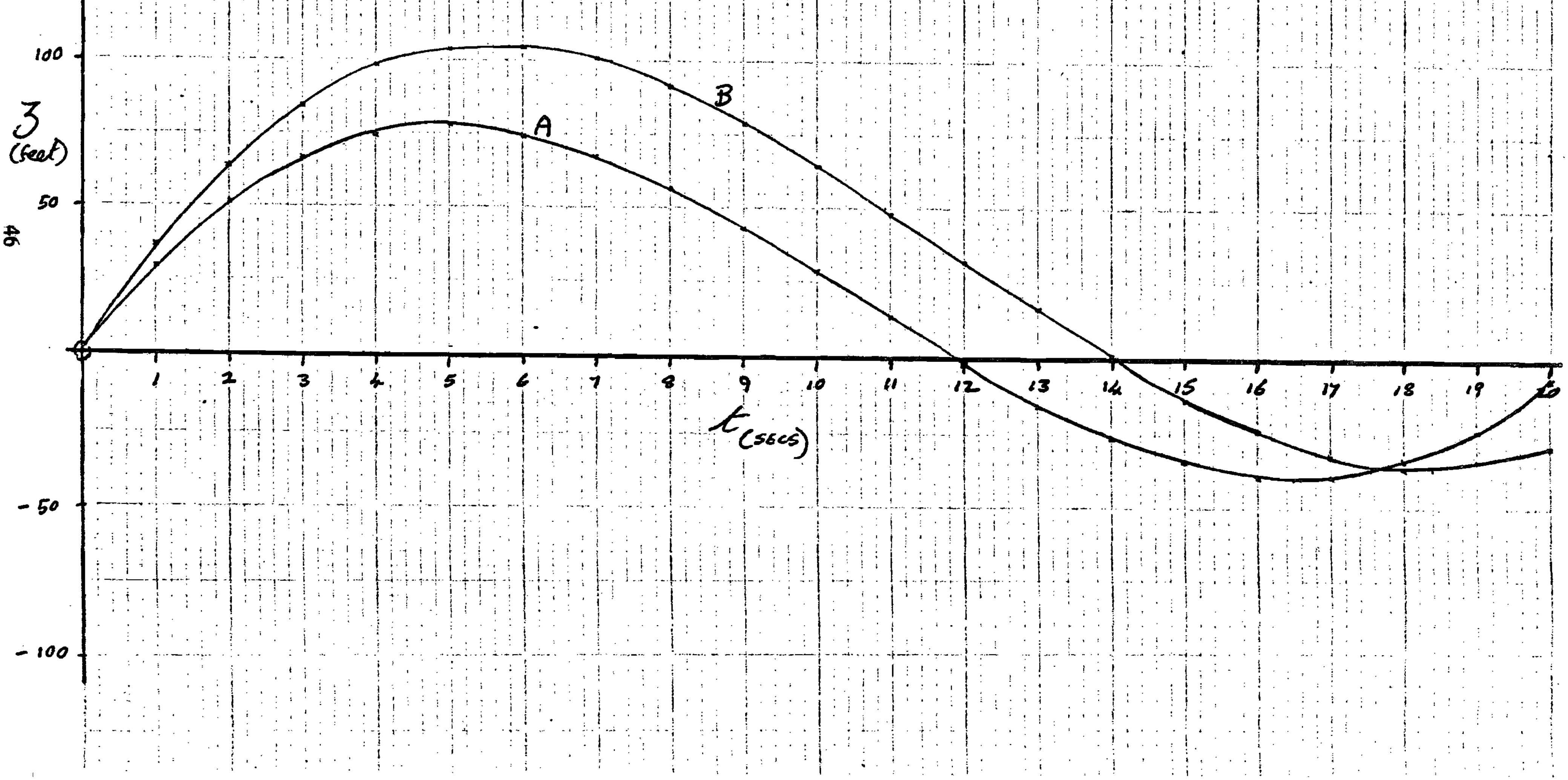
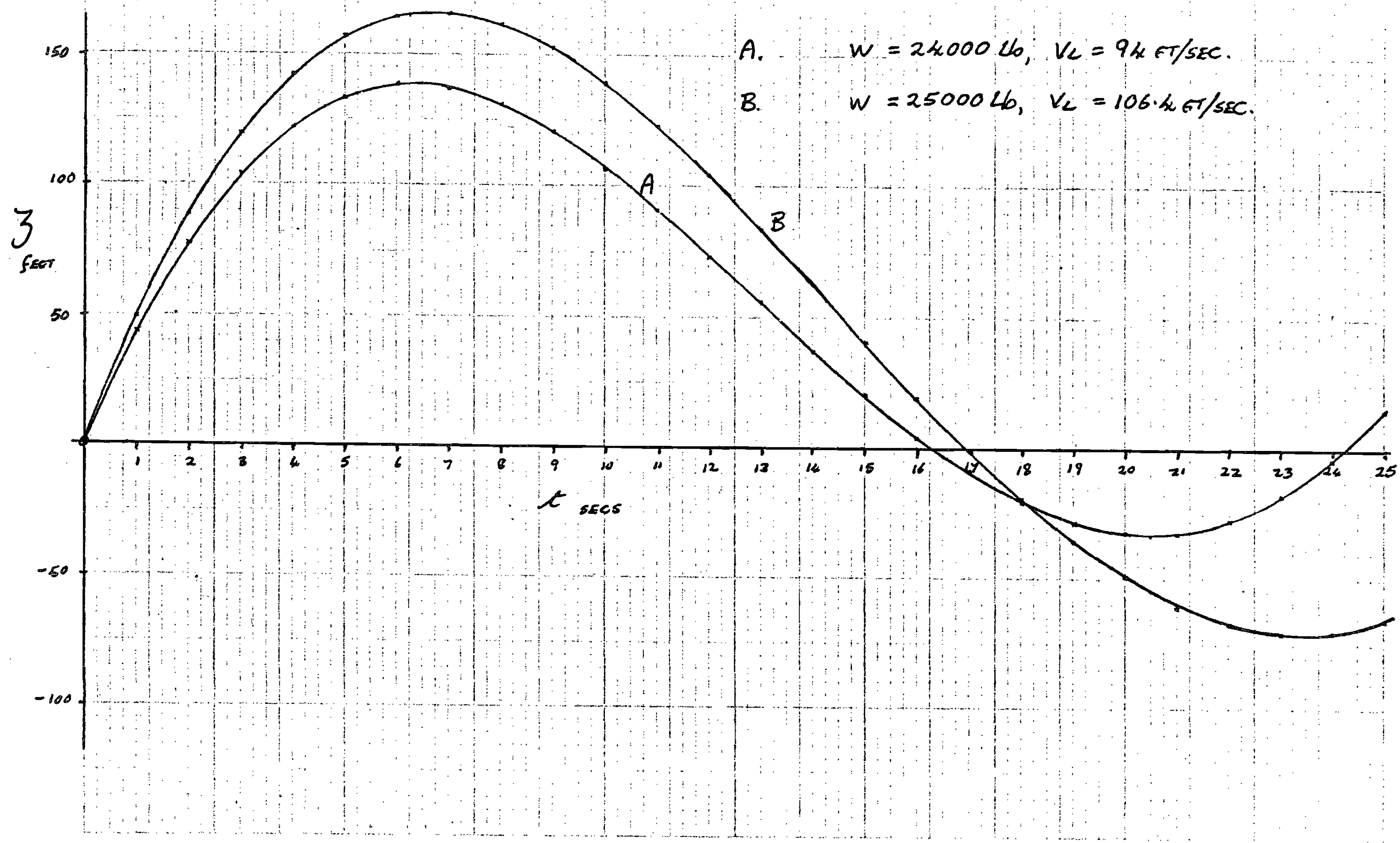


FIG 18. HARRIER PERFORMANCE PREDICTION

($\alpha = 30°$, $T = 19200$ lb, $\theta_T = 60°$)



E.R.

FIG 19

HARRIER — V_L/W

$T = 19200$ lbs

$\theta_T = 60°$

$\alpha = 30°$

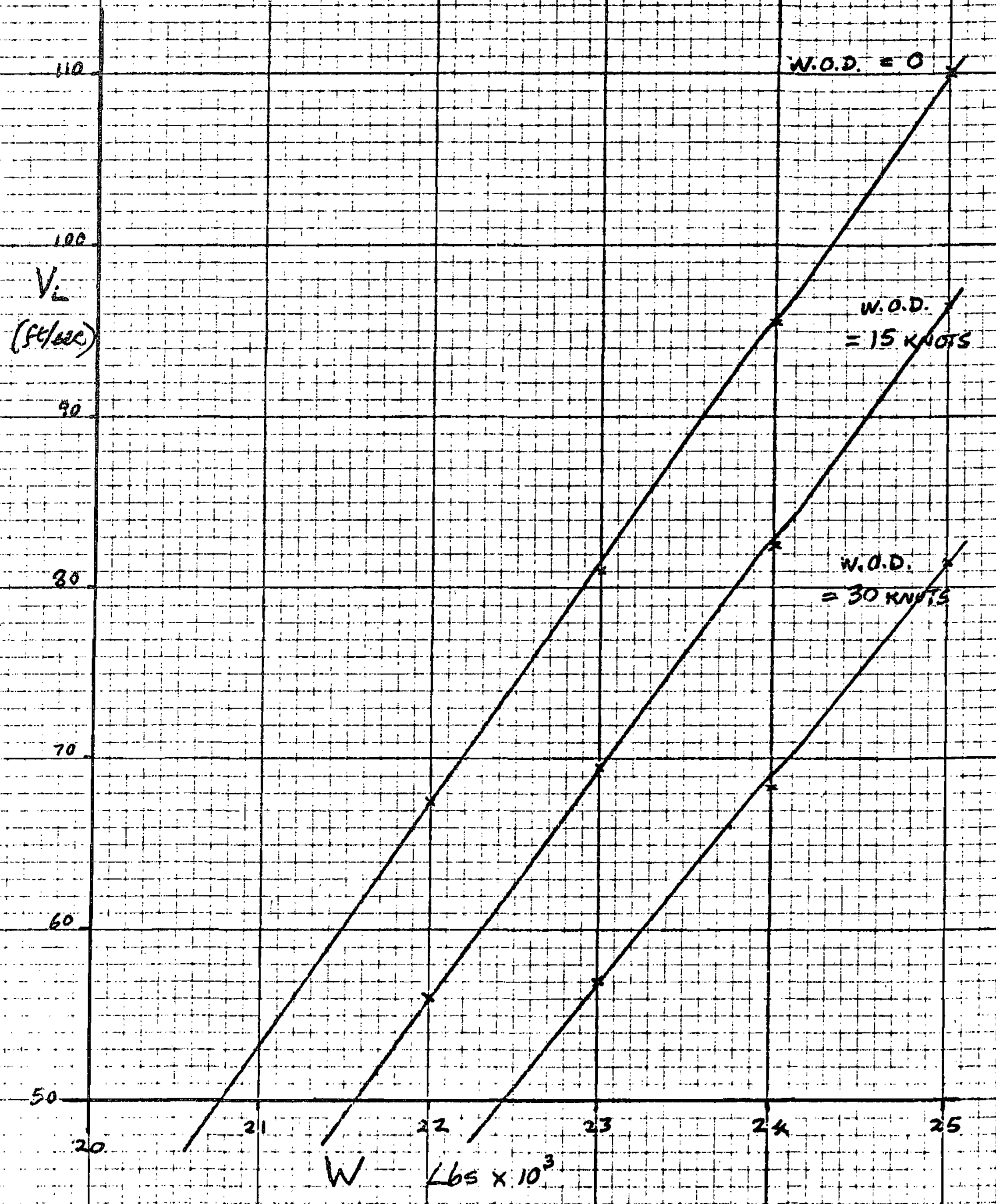
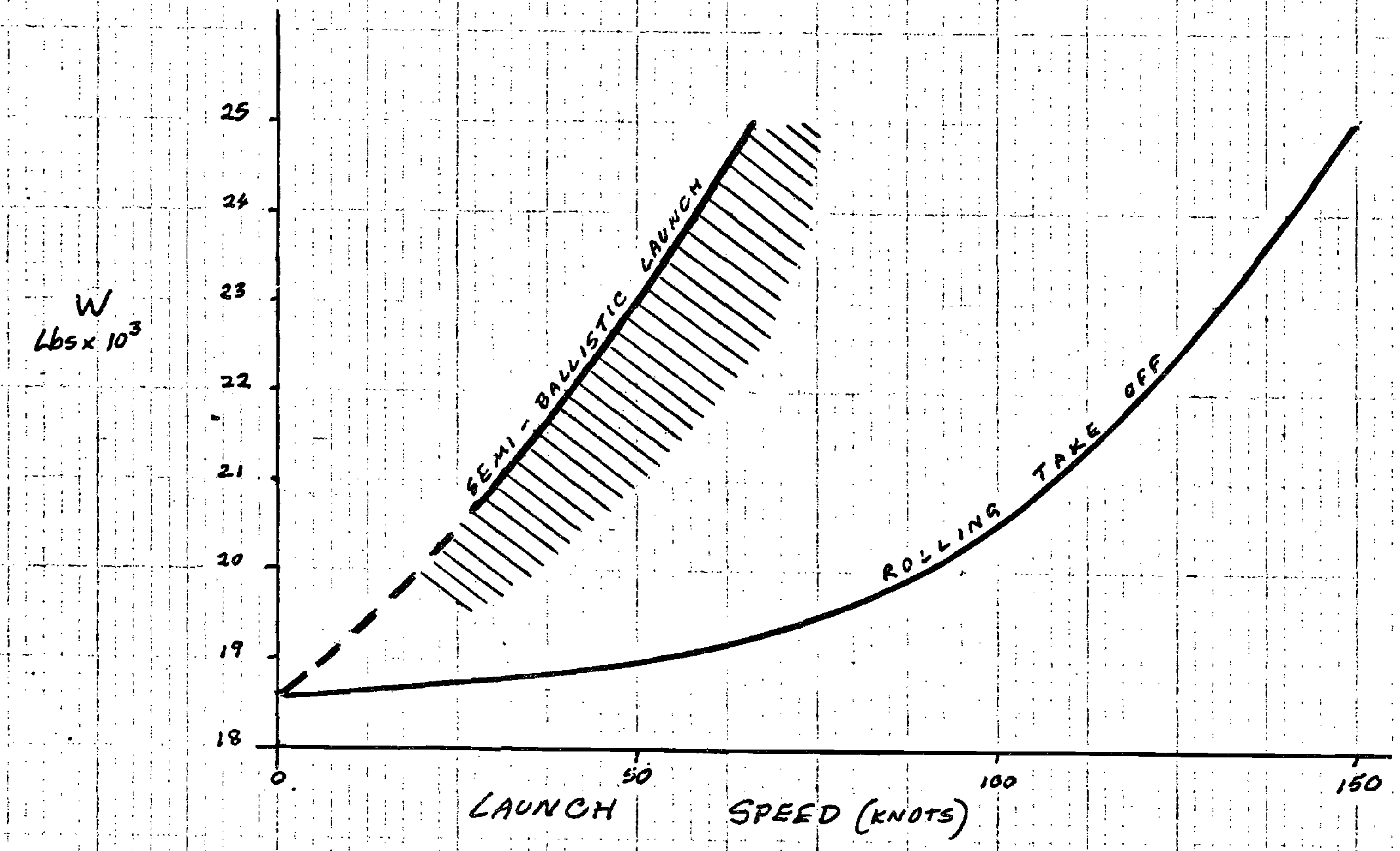


Fig 20


W
Lbs x 10^3
18
19
20
21
22
23
24
25
0
50
100
150
LAUNCH SPEED (KNOTS)
SEMI - BALLISTIC LAUNCH
ROLLING TAKE OFF

47

At this time control forces are provided by the jet reaction controls and the aircraft may be considered to be virtually neutrally stable about all axes. As airspeed increases aerodynamic control power is added to reaction control power and generally the aircraft achieves positive stability though there can be a transient period when the aircraft is unstable from aerodynamic causes. The above conditions apply whenever the aircraft is in transition between hovering and forward flight and no serious control or stability problems arise.

1.14.6 In the semi-ballistic flight which is now postulated the situation is rather different. The difference is evident by reference to Fig 20. The lower curve represents approximately the variation of aircraft take-off weight with lift-off speed from hovering flight to rolling take-off. The curve and the area below it encompass normal Harrier operations and is well explored. The limits of semi-ballistic flight from a launch angle of 30° are represented by the upper curve. We now propose to operate in the area between the two curves and particularly in the shaded area. Put simply, for any given weight the airspeed is likely to be much lower and the likely effects of this on stability and control must be considered. The airflow around the Harrier in this situation and especially during the first few seconds of semi-ballistic flight will be considerably modified by the four columns of deflected jet efflux. The complete picture requires model and full scale flight testing but a reasonably good prediction can probably be obtained by examination of the data available.

Considering the three axes in turn the situation is as follows. (Calculations are at Appendix 6)

1.14.7 Yaw. The aircraft is unstable in yaw at the speeds under consideration for launching and remains so, though decreasingly, for most of the semi-ballistic flight. This instability arises as a result of intake momentum drag. The air intakes are situated well forward of the aircraft CG, and in them a very large change in momentum of the incoming air occurs creating high drag forces. With the jets deflected downwards at large angles there are no counterbalancing forces aft of the CG until forward speed is great enough for the fin to become effective. At low forward speeds therefore any small disturbance in yaw results in a yawing moment being applied to the aircraft in the same sense and this moment increases with increasing yaw angle. Whether this instability imposes serious limitations or not depends on the rate of divergence following the initial disturbance and the control forces available to correct it. The condition is potentially most serious immediately after launch when airspeed is low.
A representative case is the situation following the launch of the aircraft at a weight of 23,000 Lbs and a speed of 50 knots. It is assumed that the aircraft is launched in a yawed condition and the yaw allowed to develop for a period of two seconds. The pilot then takes control and, using an

average of 0.6x the maximum jet reaction yaw control available to him, restores the aircraft to an unyawed condition. The result may be tabulated as follows:

Initial Yaw Angle	Yaw Angle After 2 seconds	Time to Achieve Zero Yaw (secs)
5°	7.47°	1.36
10°	14.82°	2.03
15°	22.26°	2.61
20°	29.82°	3.14

These results indicate that the increase in weight over the VTOL condition results in a low rate of divergence due to the high inertia of the aircraft in yaw. While this also gives lower control response times the jet reaction control power remains adequate. However launching in a yawed condition is by no means an unlikely occurrence and if the initial yaw is not immediately corrected, large angles of yaw can develop.

At speeds above 50 knots the stabilising influence of the fin will become appreciable and will tend to outweigh the increase in destabilising intake momentum drag as the aircraft accelerates. Published accounts of flight experience indicate that the divergence may be appreciably less than that calculated which ignores fin damping. It therefore seems probable that the aircraft can be launched with about 15° of yaw or a cross-wind component of up to 12 knots though it is obviously desirable to avoid this particularly where local wind effects may produce concurrent adverse rolling moments.

1.14.8 Roll. Immediately after being launched the aircraft may be considered to have neutral stability in roll with stability becoming increasingly positive throughout the period of semi-ballistic flight. Purely rolling disturbances are likely to be fairly small and transitory and the jet reaction control power for their correction is ample. However, roll must be considered in association with yaw because the mainplane has a pronounced anhedral which will generate a rolling moment when the aircraft is sideslipping. Unless lateral velocity is kept small there is a very real danger that the consequent rolling moment will exceed the control power available particularly if near maximum yaw control is being applied at the same time.

This rolling effect is more marked at high angles of incidence and so launches at the lower weights (and hence low launch velocity) in a strong wind over deck, which results in high incidence on launch, are likely to be particularly affected.

1.14.9 Pitch. The aircraft is almost neutrally stable at launch with positive stability increasing steadily throughout the period of semi-ballistic flight. Although damping in the pitching plane will increase rapidly the reaction control forces available are powerful and there is thus a risk of over-controlling during the first few seconds of semi-ballistic flight if large and rapid corrections to aircraft attitude are required. Any divergence from the optimum angle of incidence is bound to affect the trajectory, nevertheless a certain amount of over-controlling immediately after launch is tolerable because aerodynamic forces are small and the resultant gains or losses in lift, even at quite large divergencies from the optimum are not significant.

In the last part of the semi-ballistic trajectory, serious divergencies are much less likely since the aircraft has positive stability with substantial damping and the pilot will have had time to correct any early disturbances or oscillations due to over-controlling. There is however a period at, or just after the peak of the trajectory where the pitching rate of the aircraft is required to change considerably and it is at this point that divergencies are most likely to occur with significant effect. The situation is probably no worse than that of many a conventional aircraft requiring to be rotated accurately in pitch immediately after take-off at low airspeed; all the same the likely effects are worth examining briefly.

1.14.10 Consider first a case in which the aircraft has established a nose-down pitch rate of 6^{o}/sec to conform with the early stage of the trajectory. At the peak of the trajectory at, say, $t = 5$ secs, the pitch rate required reduces, let us assume, to zero, although in fact the reduction is likely to be less abrupt. Normally changes in pitch rate can be expected to be made smoothly, progressively and with anticipation but we will further assume that the pilot

does nothing until his Airflow Direction Detector (ADD) shows a noticeable reduction in attitude below the optimum. Then he takes corrective action using near maximum reaction control power initially, reducing quickly as the ADD indicates that the attitude is returning to the correct value, such that the average control power used is 0.6x maximum.
From Appendix 6 we see that the result is a maximum divergence from optimum attitude of -1.006^{o} and that the divergence is corrected in 0.39 seconds. The effect on the trajectory is a total height loss of 0.83 feet which is negligible.

1.14.11 Considering next a more severe case, the situation is as before but the nose-down pitching is allowed to continue for one second after falling below the optimum before the pilot takes action. This assumes that the pilot's attention has been distracted from that vital instrument, the ADD, so that a large divergence has developed before correction is applied. As before, an average of 0.6x maximum control power is used. In this case the divergence from the optimum attitude is large, -7^{o}, and there is a total time lapse of 2.22 seconds before the divergence is corrected. The resultant loss of height at the flyaway point is 32.3 feet and depending on initial launch height this could bring the aircraft dangerously close to the sea. This is an extreme case, however, and this simple treatment has ignored the likelihood that a large divergence of this nature would be countered by a more prolonged use of maximum nose-up control power resulting in an overswing which will reduce the loss of height. The changes in drag do not greatly affect the trajectory.

1.14.12 To summarize, it seems that the launching technique and the increase in weight as compared with VTOL operations introduce no fundamentally new control or stability problems. The weight increase is, if anything, beneficial in the Yaw/Roll case. In the pitching plane the importance of accurate flying is self-evident; maintaining an accurate attitude should be no great problem to the Harrier pilot as he will be able to devote most of his attention to the ADD while in semi-ballistic flight. The landing of an aircraft such as the Buccaneer on an aircraft carrier is a much more difficult task by comparison as in that case the pilot must maintain attitude equally accurately while simultaneously monitoring airspeed, making precise power adjustments and following the indications of the Deck Landing Sight. There appears then to be no obvious objection to the use of the semi-ballistic method for launching Harriers. We must therefore now examine ways and means of achieving semi-ballistic flight.

CHAPTER 2

Design of a Launching System

2.1 The system is required to launch V/STOL aircraft into a semi-ballistic trajectory to enable them to operate effectively from ships at sea at thrust/weight ratios less than unity.

2.2 This broad specification may be expanded and made more definitive by listing the characteristics of an ideal launching system and placing them in order of priority. The main characteristics will be determined by the performance required and by physical limitations. Their order of priority is to a large degree a matter of individual opinion.

The list which follows therefore reflects the author's views which place a high value on small size, simplicity and reliability, even at some cost in performance, for reasons which are outlined in 5.2.3.

2.3 Ideal Characteristics of a Launching System

(i) Small size

Ideally it should occupy no more deck space than is required for the aircraft itself.

(ii) Reliability

It must be robust, tolerant of minor damage, easily repaired and maintained. It is therefore likely to be simple.

(iii) Usable in all but extreme sea states.

(iv) Independent of wind strength and direction.
The need, in calm conditions, to work up to a high ship speed or to turn into the natural wind is a tactical limitation. See 2.13 however for qualifying remarks.

(v) Capable of launching specified aircraft at their maximum operational flying weights.

(vi) Imposing no special modifications or limitations on the aircraft.

(vii) Capable of launching aircraft at short intervals.
The desirability of this feature is dependent on the number of aircraft and launchers in the ship. At one extreme, if only one aircraft is carried then clearly this is of low priority.

(viii) Require no complex piloting techniques.

(ix) Quick Reaction time
Must require no lengthy preparation or warming-through before being brought into action.

(x) Must be readily adaptable to various types of fixed wing V/STOL aircraft.

(xi) Must be readily applicable to all ocean going surface ships.

(xii) Loading drill must be swift and simple.

Inevitably a compromise will be necessary.

2.4 Types of Launching System

2.4.1 Although the extensive use of fixed wing V/STOL aircraft at sea is an inevitable, if long delayed development, the Harrier is the only aircraft likely to become widely operational in the immediate future.

In practical terms the problem therefore resolves itself into a requirement to impart to this aircraft vertical velocities of up to about 50 ft/sec together with horizontal velocities of up to 100 ft/sec. The aircraft's weight at present is unlikely to exceed 25,000 Lb but in the longer term 30,000 Lb is likely to be achieveable. There are a variety of ways of meeting the requirement, either by adaptation of tried and proven methods or by entirely novel means. A number of methods have been examined and the most promising methods investigated at some depth.

As a basis of comparison, the two most tried and proven conventional methods of launching aircraft are considered first. They are:

(i) Horizontal, free rolling take-off.

(ii) Horizontal catapult launch.

2.5 Horizontal, free rolling Take-off

2.5.1 The technique for this is summarized at 1.1.1. This is the method currently employed for ship-borne Harriers.

If the speed required for level flight at a given weight and nozzle deflection angle is known and the acceleration is assumed to be uniform then a simple formula gives reasonable accuracy in calculating take-off distance. Some allowance must be made for drag and rolling friction. In the following calculations this is done by applying a factor of 1.05 to the speed required.

If:

Speed required	=	v
Acceleration	=	a
Installed thrust	=	T
Aircraft weight	=	W

Then $a = \frac{Tg}{W}$ and,

Take-off distance $= \frac{v^2W}{2Tg}$

For example at a weight of 22,000 Lbs, thrust of 19,200 Lbs a Harrier doing a rolling take-off requires a speed of 200 ft/sec.

$\therefore v = 200 \times 1.05 = 210\text{ft/sec}$

and

Take-off distance $= \frac{210 \times 210 \times 22{,}000}{2 \times 19{,}200 \times 32.2} = 784$ feet

in no wind.

Horizontal, free-rolling take-off distances calculated by this method are:

WEIGHT (Lbs)	WIND OVER DECK (knots)	DISTANCE (feet)
22,000	0	784
22,000	15	602
22,000	30	442
25,000	0	1,222
25,000	15	990
25,000	30	765

2.5.2 This type of take-off meets many of the requirements of the ideal but completely fails on characteristics (i) and (iv). Since the aircraft must take-off over or near the bow where pitching motion is large the method is likely to be limited to moderate sea states and at the smallest end of the scale - a ship about 500 feet long - it is unlikely that condition (iii) could be met.

In addition to its sensitivity to ship pitching this method is sensitive to any delay in nozzle rotation. In practice steps are taken to ensure that nozzle rotation occurs while the aircraft is still on the deck and this adds significantly to the deck space required. As a result the method is unlikely to be applicable in ships of less than 15,000 to 20,000 Tons.

2.6 Horizontal Catapult Launch

2.6.1 Here it is assumed that the aircraft is accelerated, by a catapult, at 4g to flying speed. So at 22,000 Lb the take-off distance required is: $\frac{v^2}{2a} = \frac{200 \times 200}{2 \times 4 \times 32.2} = 155$ feet

Similarly the following table of take-off distances can be constructed:

WEIGHT (Lbs)	WIND OVER DECK (knots)	DISTANCE (feet)
22,000	0	155
22,000	15	119
22,000	30	88
25,000	0	214
25,000	15	171
25,000	30	133

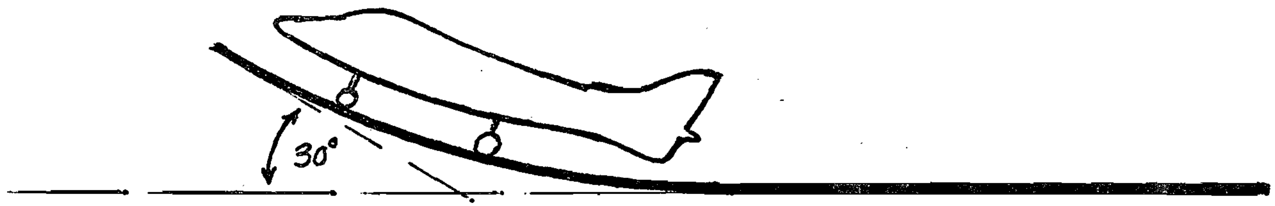
2.6.2 This approaches the ideal in terms of deck space but is complex and requires specialised aircraft. A major shortcoming is the size, weight and complexity of the catapult machinery. Probably a steam catapult would be necessary in which case 5 feet must be added to the distances given above for deceleration of the pistons and shuttle. It suffers the same sensitivity to ship pitching as the horizontal free take-off. It is therefore unlikely to be a practical proposition in ships of less than 15,000 tons.

2.7 Clearly neither of the above two methods can be applied to small ships but the tables can be used as a useful yardstick in assessing the performance of other methods using the semi-ballistic launch technique. As this technique involves launching the aircraft at an angle of around 30° to the horizontal and since, at launch the aircraft must be approximately aligned with the relative airflow a complication arises, by comparison with horizontal launches, in that before or during the launching process the aircraft must be rotated by about 30° in pitch. Ideally this rotation should occur during the launching process to meet condition (xii) of the required characteristics.

The solutions which come closest to the ideal and are most readily feasible in the short term are considered in the next section.

FIG 21

THE SKI - JUMP



2.8 The Ski-Jump

2.8.1 The principle is illustrated in Fig 21. The aircraft accelerates under its own power with thrust undeflected, ($\theta j = 0^o$) horizontally at first and then up a curved ramp until its velocity is at 30^o to the horizontal. On reaching the end of the ramp the pilot vectors the thrust to a pre-selected angle and then maintains constant incidence while the aircraft follows the semi-ballistic flight path to the flyaway point.

The aircraft accelerates faster on the horizontal portion of its run than on the curved ramp. The latter therefore should be of as small a radius as possible. It must not however be so small that the tail of the aircraft strikes the deck or the centrifugal loading exceeds the strength of the undercarriage.

In the case of the Harrier, this latter is the limiting factor.

2.8.2 Undercarriage Strength:

The design vertical loads on the Harrier's undercarriage expressed as multiples of the static load on each unit at a Design Landing Weight of 16,900 Lbs are:

Main Leg	5.4
Nose Leg	4.1
Outrigger Legs	9.0

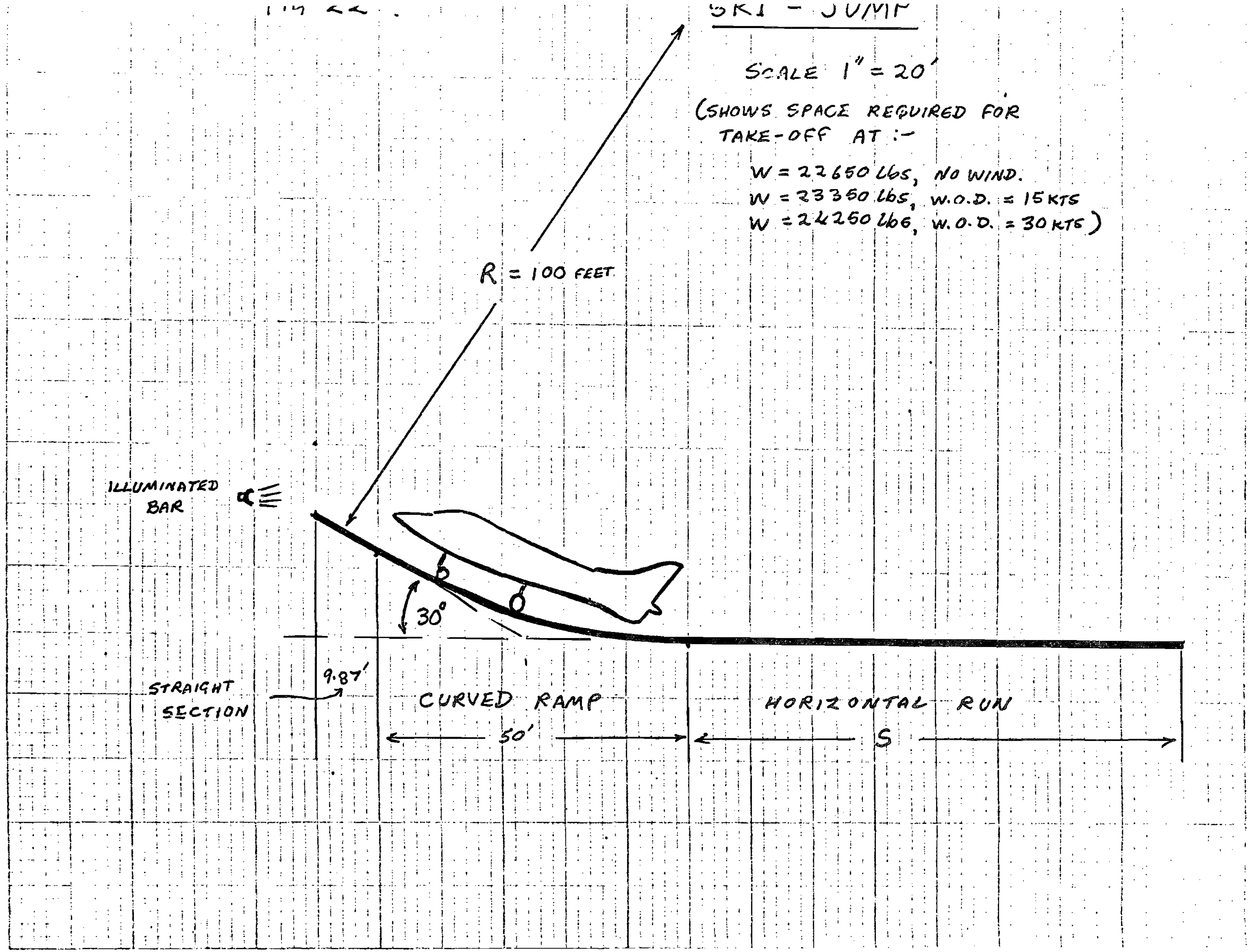
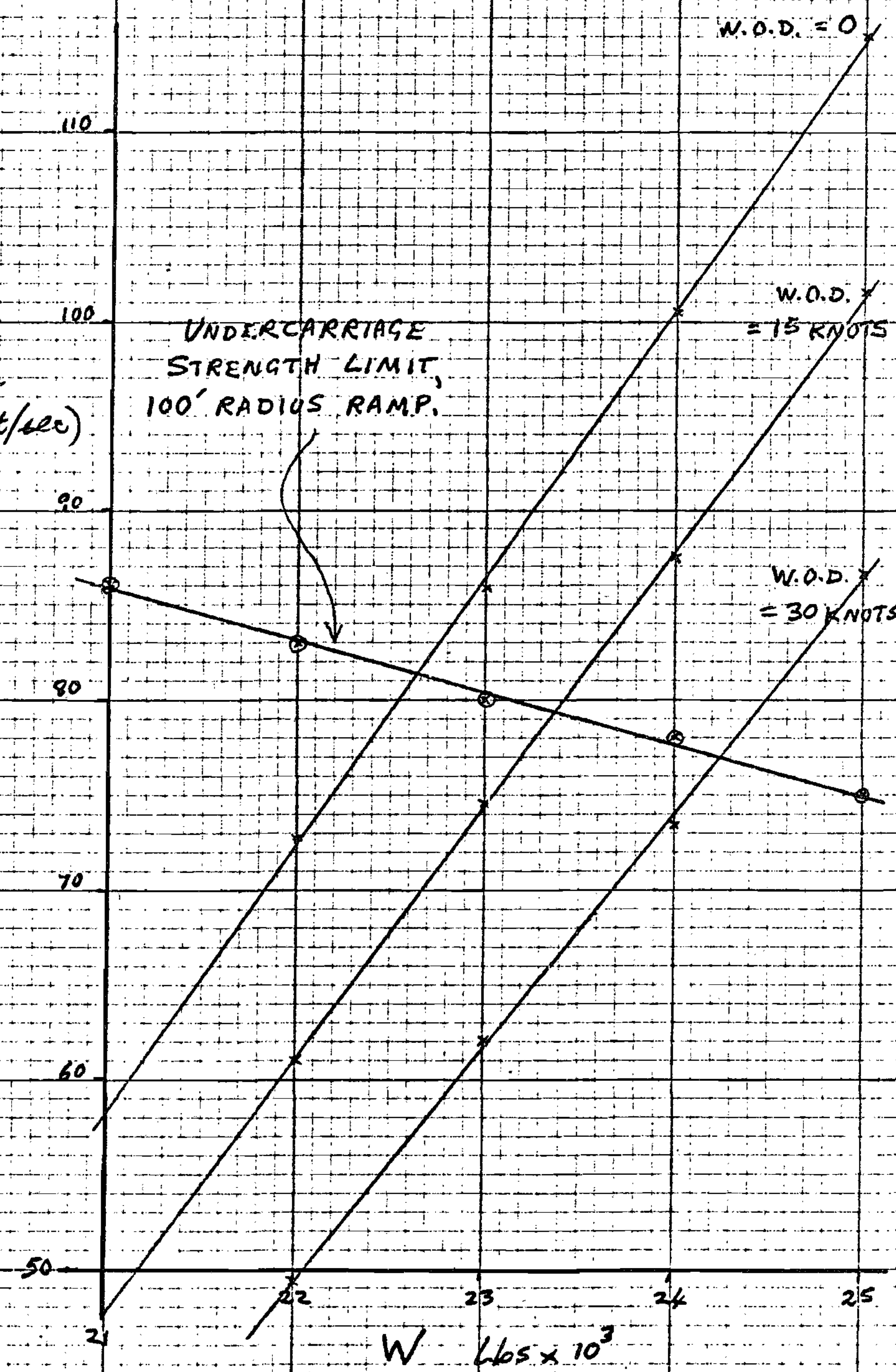


FIG 22.
SKI - JUMP
SCALE 1" = 20'
(SHOWS SPACE REQUIRED FOR
TAKE-OFF AT :-
W = 22650 lbs, NO WIND.
W = 23350 lbs, W.O.D. = 15 KTS
W = 24250 lbs, W.O.D. = 30 KTS)
R = 100 FEET
ILLUMINATED BAR
30°
9.87'
STRAIGHT SECTION
CURVED RAMP
50'
HORIZONTAL RUN
S

FIG 23

HARRIER – LAUNCH SPEEDS FOR UNASSISTED SKI-JUMP TAKE-OFF

$T = 19200$ Lbs
$\theta_T = 60°$
$\alpha = 30°$



E.R.

The strength of the nose leg therefore sets the limit of centrifugal loading. In order to ensure conservative results it has been assumed that a basic loading of 1g applies all the way round the ramp in addition to the centrifugal loading, the limit being 4.1g at Design Landing Weight.

So, maximum reaction force on undercarriage

$= 4.1 \times 16{,}900 = 69{,}350$ Lbs

$= 3.15$g at 22,000 Lbs weight

$= 2.76$g at 25,000 Lbs weight

For a given ramp radius the limits are set by weight and speed. Armed with this information a ramp radius can be selected. The selection is inevitably a compromise, in this instance a radius of 100 feet is considered to be the practical minimum. Limiting speeds for various weights on a 100 foot radius ramp are shown in Fig 23.

2.8.3 Calculation of Take-Off Distance

The symbols used have the following definitions:

V_L = velocity on leaving ramp

v_r = velocity on entry to ramp

v = average speed on curved ramp

R = ramp radius

α = angle between tangent to ramp and horizontal

g = acceleration due to gravity

W = aircraft weight

T = engine thrust

n = centrifugal loading as multiple of g

a = horizontal acceleration

a_2 = acceleration on curved ramp

S = horizontal run required

μ = coefficient of rolling friction

ω = aircraft rate of pitching

$\dot{\omega}$ = pitching acceleration

M = pitching torque

I = aircraft pitching movement of inertia about C of G

I_2 = aircraft pitching moment of inertia about main wheels

l = distance between aircraft main and nose wheels

The launch velocity required is known and the take-off distance can be calculated by working back from this. The take-off is then in two distinct stages:

acceleration round the ramp and,

horizontal acceleration.

(i) Acceleration round the ramp.

Force causing acceleration $= T - W \sin \alpha$

$$= \frac{W}{g} a_2$$

$$a_2 = \frac{g}{W}(T - W \sin \alpha) \qquad (8)$$

Variation of a_2 with α is almost linear between

0^{o} - 30^{o} so that an acceptable result can be obtained by finding an average value of a_2 and treating the problem as one of motion under uniform acceleration.

$$\text{ie} \quad v_r = \sqrt{V_L^2 - 2a_2R\alpha} \qquad \text{——— (9)}$$

At the speeds being considered aerodynamic drag is negligible but rolling friction is significant, particularly on the ramp.

As the aeroplane will be running over a smooth steel surface a value of μ = 0.025 has been assumed.

$$\text{Friction force on ramp} = \mu W(n + \text{Cos}\,\alpha)$$

$$n \doteq \frac{V^2}{gR}$$

$$\therefore \text{Friction force on ramp} = \mu W\left(\frac{v^2}{gR} + \text{Cos}\,\alpha\right)$$

Since v is not known an iterative process becomes necessary and:

$$a_2 = \frac{g}{W}\left[T - W\,\text{Sin}\,\alpha - \mu W\left(\frac{v^2}{gR} + \text{Cos}\,\alpha\right)\right] \qquad \text{——— (10)}$$

(ii) Horizontal Acceleration.

$$\text{Friction force} = \mu W$$

$$a = \frac{g}{W}(T - \mu W)$$

$$S = \frac{v_T^2}{2a}$$

$$= \frac{v_T^2 W}{2g(T - \mu W)} \qquad \text{——— (11)}$$

In travelling around the curved ramp a pitching motion has been imposed upon the aircraft $= \dfrac{V_L}{R}$ radians/sec.

When the nosewheel clears the end of the ramp a pitching moment in the opposite sense is applied whilst the main wheels are still on the curved ramp.

This moment is:

$$M = W\left(\frac{V_L^2}{gR} + \cos\alpha\right) \text{ x distance between main}$$

wheel point of contact and the aircraft C.G.

(= 4.25 feet)

$$\therefore M = 4.25W\left(\frac{V_L^2}{gR} + \cos\alpha\right)$$

But $M = I_2\dot{\omega}$

$\therefore \dot{\omega} = \dfrac{M}{I_2}$ and is applied for a time of

$\dfrac{l}{V_L}$ seconds.

$$\therefore \omega = \frac{Ml}{I_2 V_L} = \frac{4.25Wl\left(\frac{V_L^2}{gR} + \cos\alpha\right)}{I_2 V_L}$$

So the resultant pitching motion of the aircraft is:

$$= \frac{V_L}{R} - \frac{4.25\,Wl\left(\frac{V_L^2}{gR} + \cos\alpha\right)}{I_2 V_L} \qquad \text{——— (12)}$$

In all cases investigated the resultant motion was in the negative, ie nose-down sense. A maximum permissible pitching rate of 6°/second was chosen and in most cases this was exceeded. It is therefore necessary to reduce the rate of pitching to acceptable levels before the aircraft starts its semi-ballistic flight. This can be done by adding a straight section equal in length to l (which for the Harrier is 11.4 feet) to the curved ramp.

The total space required for take-off is therefore:

$$S \; + \; \text{horizontal length of ramp} \; + \; 11.4 \cos 30^{\circ}$$

A typical ramp layout is shown to scale at Fig 22.

2.8.4 In common with the horizontal free rolling take-off the Ski-Jump method requires timely and accurate thrust vectoring. Accuracy can be assured by a pre-positioned stop on the thrust vector control lever, timing is entirely in the pilot's hands and without some help is liable to an unacceptable degree of scatter. The standard aid for a horizontal take-off is a line painted on the deck near the bow. The pilot operates the nozzle lever as the line disappears from his sight at the bottom of the windscreen. With the Ski-Jump system it may well be that the cessation of 'g' loading and the nose down nod of the aircraft on reaching the end of the curved portion of the ramp will be a sufficient cue to the pilot. This can only be discovered by trial but in any case a visual cue is, at the very least, highly desirable since it can be so devised as to allow for pilot reaction time and nozzle rotation time. The use of physical sensation as a cue can only result in action <u>after</u> the event.

2.8.5 The best form of visual cue for the Ski-Jump system is likely to be an adaptation of the existing painted line. Since there is no deck at the end of the ramp and the pilot will be looking slightly upwards towards a background of sky the painted line is probably best replaced by an illuminated bar. A further refinement could be a suitably positioned switch, operated by the nose wheel passing over it, which would cause a bright flash of the illumination just before it disappeared below the pilot's line of sight. This would give the pilot three calls for action in rapid succession: the flash, the disappearance of the illuminated bar and the cessation of 'g' loading. The exact positioning of the illuminated bar must be largely a matter of experiment. At the moment under consideration the undercarriage legs would be compressed by the centrifugal loading on the aircraft and in those circumstances the bar would disappear from the pilot's sight at a point about 30 to 35 feet ahead of the nosewheel. The best position of the bar would therefore probably be 10 to 20 feet beyond the end of the ramp and 2 to 6 feet below the extended line of the ramp. The position is indicated in Fig 22.

2.8.6 With such positive cues it seems not unreasonable to expect a high degree of accuracy in timing of the thrust vectoring and therefore only a small allowance need be made for the residual inaccuracies. For this purpose an addition of 5 ft/sec is made to the launch speeds given in Fig 19 and take-off distances are calculated on this basis. Typical calculations are at Appendix 7. The results are tabulated below and shown graphically in Fig 24.

Harrier Take-Off Performance - Unassisted

Ski-Jump

(T = 19,200 Lbs, θ_T = 60°, LAUNCH ANGLE 30°)

AIRCRAFT WEIGHT (Lbs)	WIND OVER DECK (knots)	LAUNCH SPEED		TAKE-OFF SPACE (ft)	
		FT/SEC	KNOTS		
21,000	0	59	35.4	85	
22,000	0	72.6	43.6	122	
22,650	0	81.5	48.9	152	⨍
21,000	15	47.5	28.5	63	
22,000	15	61	36.6	93	
23,000	15	74.5	44.7	133	
23,350	15	79.5	47.7	150	⨍
22,000	30	49.5	29.7	69	
23,000	30	62	37.2	100	
24,250	30	77	46.2	150	⨍

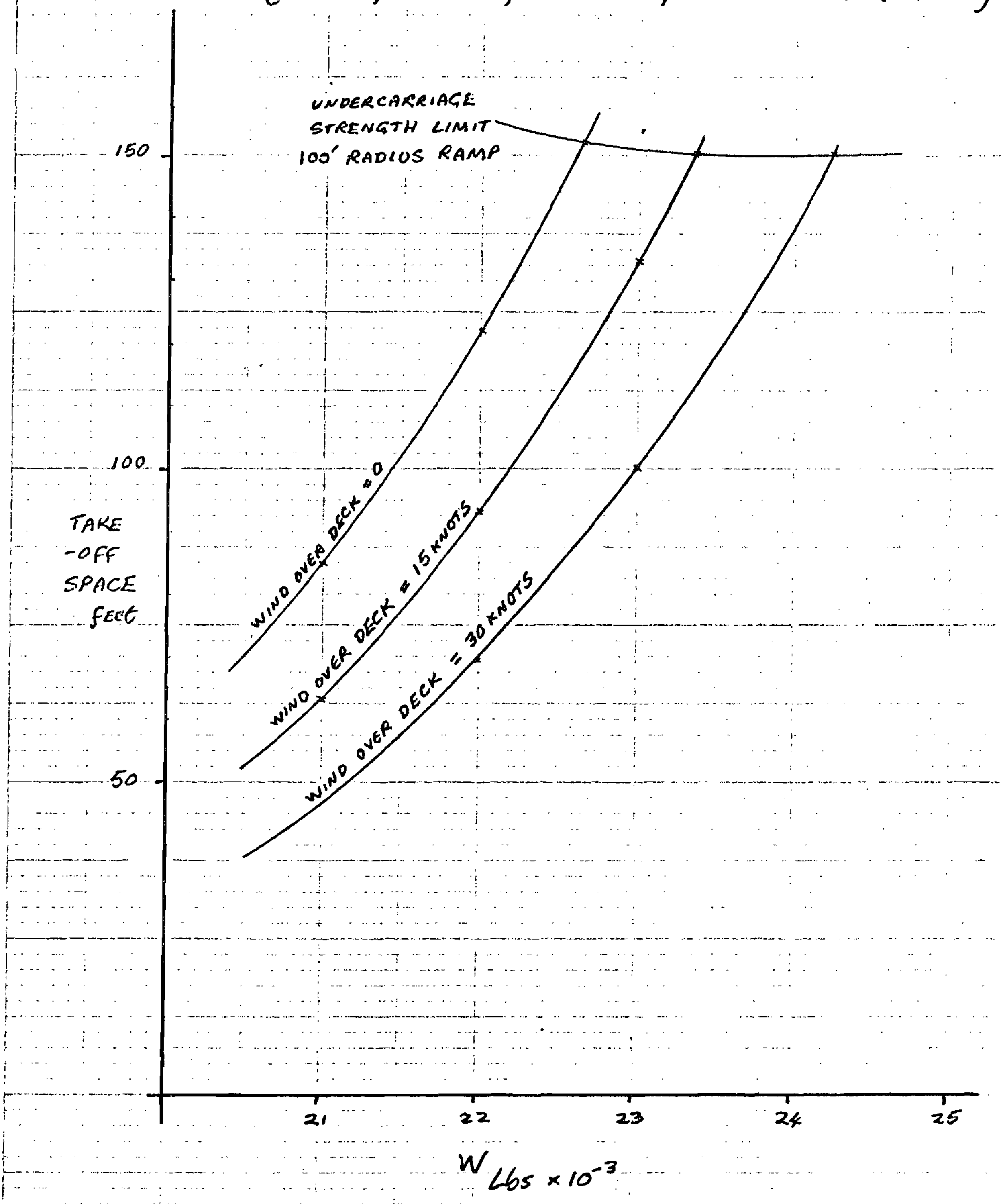
⨍ Undercarriage limit

To operate the Harrier at a weight of 25,000 Lbs within existing undercarriage limitations and with a thrust of 19,200 Lbs would necessitate an increase in ramp radius to

FIG 24

HARRIER - UNASSISTED SKI-JUMP TAKE-OFF PERFORMANCE

(T = 19,200 Lbs, θ_T = 60°, LAUNCH ANGLE = 30°)



132 feet, assuming wind over deck is 30 knots. The take-off space would be in the order of 200 feet.

2.8.7 These results show that the Ski-Jump produces a take-off performance comparable or better than the horizontal steam catapult but without the complexity, weight, expense and space requirements of the latter. It very nearly meets all the requirements of the ideal launching system, falling short only on items (i) and (iv). In the latter case it could be made independent of wind direction by mounting it on a turntable but the complication and expense seem hardly worthwhile. The speed of operation would be such that with the small number of aircraft to be launched the ship would only have to turn into wind for a very short period. The greatest advantage of the Ski-Jump is its simplicity. There are no moving parts, nothing to go wrong. It is therefore near the ultimate in reliability and should be very tolerant of minor damage and easily repaired. It will also be very cheap.

Other advantages are:

(a) A gain in initial height as part of the launching process since the end of the ramp is 20 feet above flight deck level.

(b) Flexible positioning. A ski-jump launch does not have to be over the bow. This leads to:

(c) less sensitive to ship pitching. The ramp can be placed so that the launch point is near the mid-length of the vessel where pitching motion is least.

(d) less wasted space - the area under the ramp is usable for other purposes.

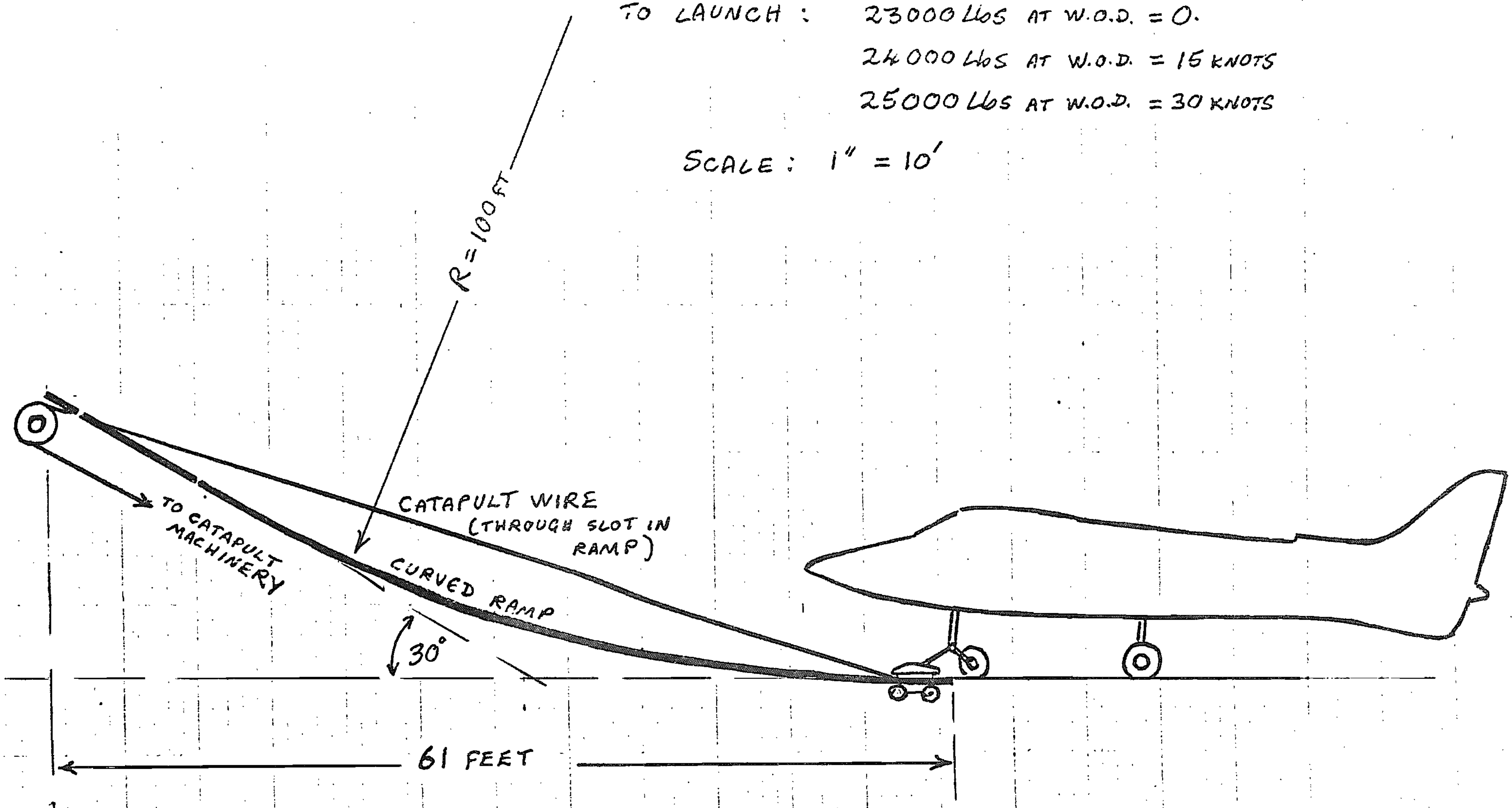
Where the Ski-Jump falls short of the ideal in item (i) - space required; it can be improved at the cost of some loss of simplicity by combining it with a catapult. This is the next launching system to be considered.

FIG 25

CATAPULT ASSISTED SKI-JUMP

TO LAUNCH: 23000 LBS AT W.O.D. = 0.
24000 LBS AT W.O.D. = 15 KNOTS
25000 LBS AT W.O.D. = 30 KNOTS

SCALE: 1" = 10'



2.9 Catapult Assisted Ski-Jump

2.9.1 For this application a steam catapult is not necessary as launch speeds are very low. There are a number of methods which could be used but the hydraulic catapult which was superseded by the steam catapult would be very suitable. In this type of catapult the energy is imparted to the aircraft by means of a wire tow rope. Its application to the Ski-Jump is illustrated in Fig 25.

As the launching energy is supplied by the catapult there is no need for a horizontal run, the aircraft can start from the foot of the ramp. For the same reason the thrust can be vectored before launch which simplifies the pilot's job and eliminates the need for extra launch speed since there is therefore no delay in nozzle rotation.

In calculating the take-off performance and speed required a maximum launch capacity of 25,000 Lb at a wind-over-deck of 30 knots has been selected. Symbols and formulae used are as follows:-

V_L = launch velocity required

R = ramp radius

L = length of ramp round curve

a = acceleration applied to the aircraft by the catapult

v = speed limit on curved ramp due to undercarriage strength

n = maximum centrifugal load factor

d = total distance required to attain

V_L under acceleration a.

Then:

$$v = \sqrt{nRg}$$

and, if the launch angle is 30°,

$$L = \frac{2\pi R \times 30^\circ}{360^\circ}$$

$$a = \frac{v^2}{2L}$$

$$d = \frac{v^2}{2a}$$

If it is assumed that the moving parts of the catapult can be brought to rest in a distance of 3 feet, then:

Total horizontal space required

= horizontal space occupied by the curved ramp

+ $(d - L) \text{ Cos } 30^\circ + 3 \text{ Cos } 30^\circ$

2.9.2 The calculations at Appendix 8 show that it is possible to reduce the ramp radius to the minimum radius giving adequate deck clearance. However it is also clear that there is no real point in doing so as the gain in space is insignificant. By retaining a ramp radius of 100 feet there is a limited launch capacity available from unassisted take-off as a fall back position if the catapult is unserviceable.

2.9.3 The space required to launch a Harrier at a weight of 25,000 Lbs with 30 knots w.o.d. is only 60 feet. From the same space, other maximum launch weights are approximately:

23,000 Lbs at w.o.d. = 0

24,000 Lbs at w.o.d. = 15 knots

2.9.4 The Catapult Assisted Ski Jump comes very close to the ideal; it has all the advantages of the ordinary Ski Jump except extreme simplicity. In addition it needs much less space and requires no action to deflect thrust on leaving the ramp. Compared with the horizontal steam catapult it requires much less space and is simpler. However catapult launching will necessitate modification of the aircraft.

2.9.5 Beyond suggesting the use of a hydraulic catapult (such as B.H.5) no examination in detail of suitable catapults has been carried out. There are a number available almost "off the shelf", which could readily be adapted to this application. Choice of the most suitable is a matter for detailed technical examination and beyond the scope of this work.

2.10 The Ballista

2.10.1 This launching device was invented to fulfil the requirement of item (i) of the ideal launching system characteristics. Its name derives from the ancient siege catapult of Roman times and earlier, which it superficially resembles. There are two basic versions which are shown at Figs 26 A, B, C and D.

In order to meet the requirement it was decided to use the maximum practicable launch angle in order to keep launch speeds as low as possible. A launch angle of 40° was chosen. Greater launch angles can easily be achieved from the Ballista but they give a longer time in semi-ballistic flight at full power and can result in high pitching rates being required of the aircraft to avoid unacceptably large angles of incidence.

The method of operation is as follows:

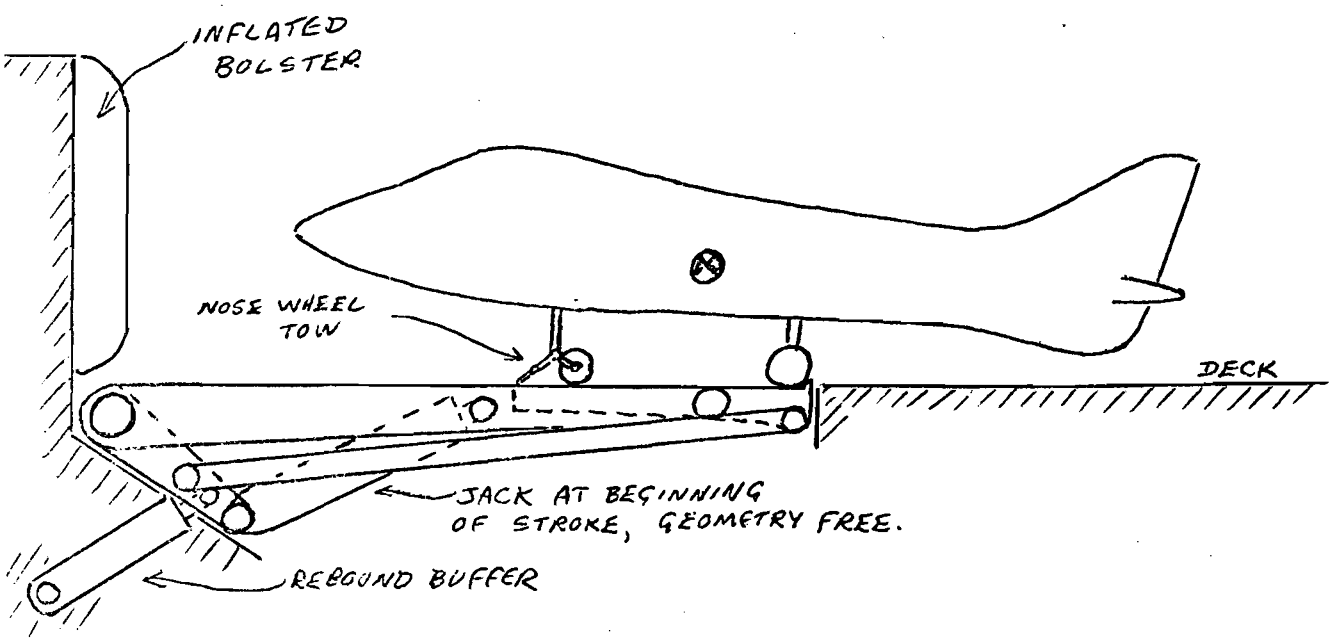
(Refer to Figs 26 A, B and C)

2.10.2 Fig 26A shows the situation at the beginning of the launching process before the mechanism has started to move. The aircraft's wheels rest upon a platform and the nosewheel is attached to the front of the platform by a towing link. (The use of the standard catapult nose tow is envisaged). The platform is pivoted at the end of a swinging arm which is powered by a jack. A linkage is attached to the platform and so arranged that as the arm swings upwards the platform and the aircraft are tilted to a pre-determined angle of about 30°. This angle is reached as the jack comes to the end of its stroke. This situation is shown in Fig 26B.

2.10.3 At this point the geometry of the arm, linkage and jack is fixed by the fully extended jack. The rotating arm has come in contact with the bolster and the ram of the rebound buffer is about to be pulled out by the inertia of the moving parts. The arm therefore decelerates and the aircraft moves forward relative to the platform, disengaging the towing link. The purpose of fixing the geometry is to maintain the angle of the platform relative to the arm so that it remains clear of the aircraft as it leaves.

2.10.4 The moving mechanism is then brought to a halt by a combination of compression of the inflated bolster and the resistance offered by the extending rebound buffer. The weight of the moving parts also assists in the deceleration process. The extended rebound buffer resists the rebound of the mechanism from the bolster and, together with the main jack, allows the mechanism to be lowered for the next launch.

FIG 26


A.
BEGINNING OF LAUNCH
INFLATED BOLSTER
NOSE WHEEL TOW
DECK
JACK AT BEGINNING OF STROKE, GEOMETRY FREE.
REBOUND BUFFER

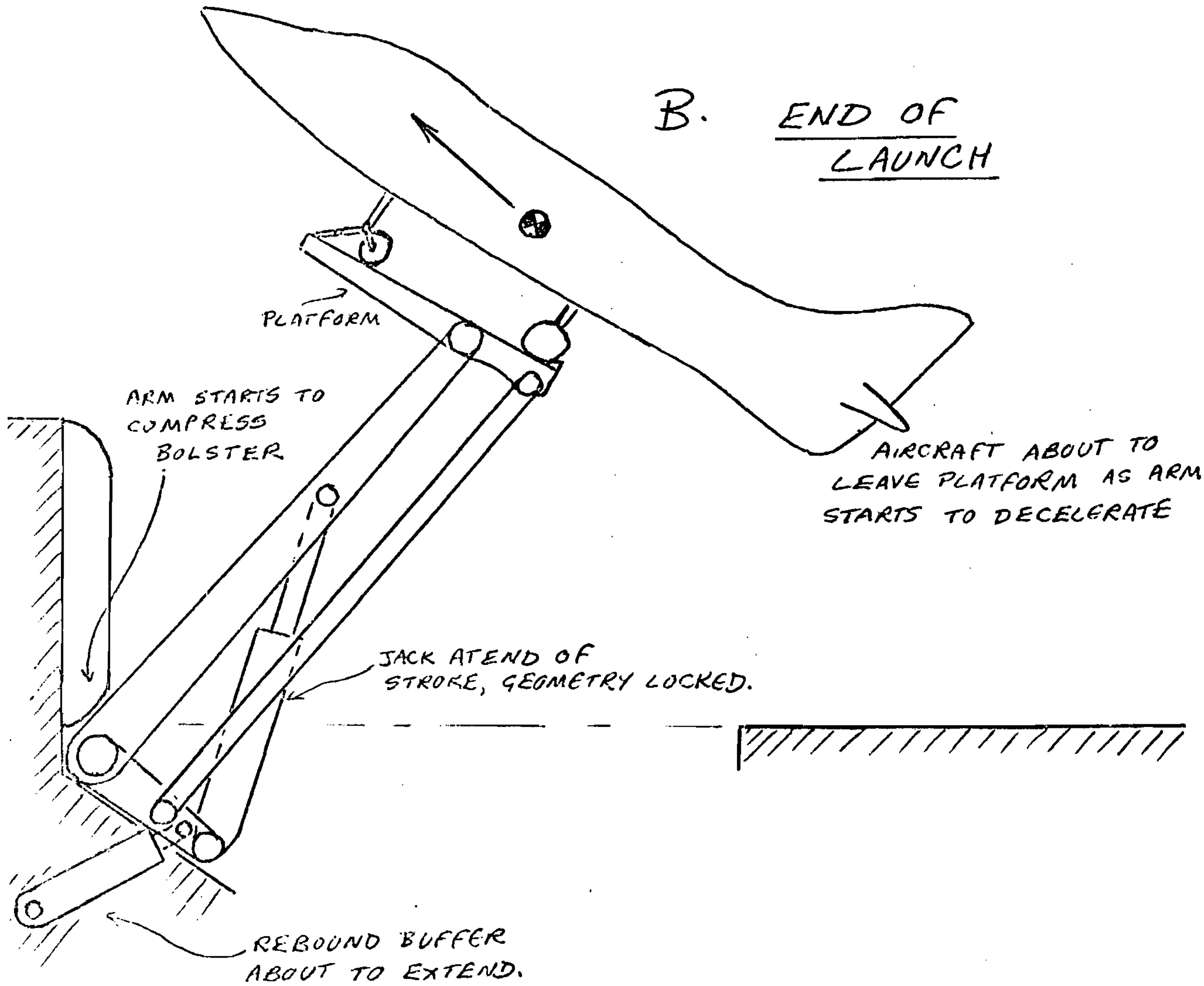

B. END OF LAUNCH
PLATFORM
ARM STARTS TO COMPRESS BOLSTER
AIRCRAFT ABOUT TO LEAVE PLATFORM AS ARM STARTS TO DECELERATE
JACK AT END OF STROKE, GEOMETRY LOCKED.
REBOUND BUFFER ABOUT TO EXTEND.

FIG 26

C. ARRESTING LAUNCHING MECHANISM

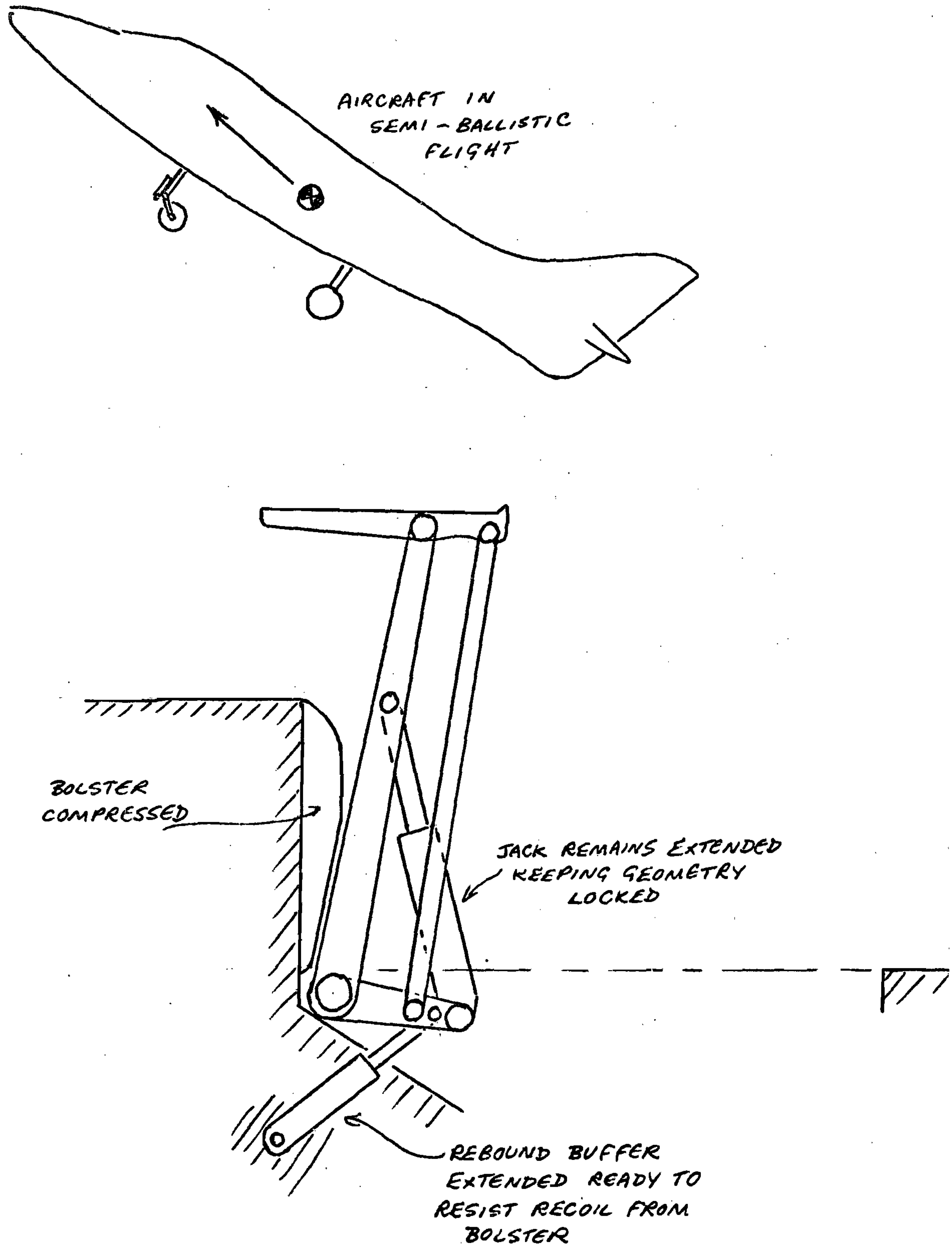
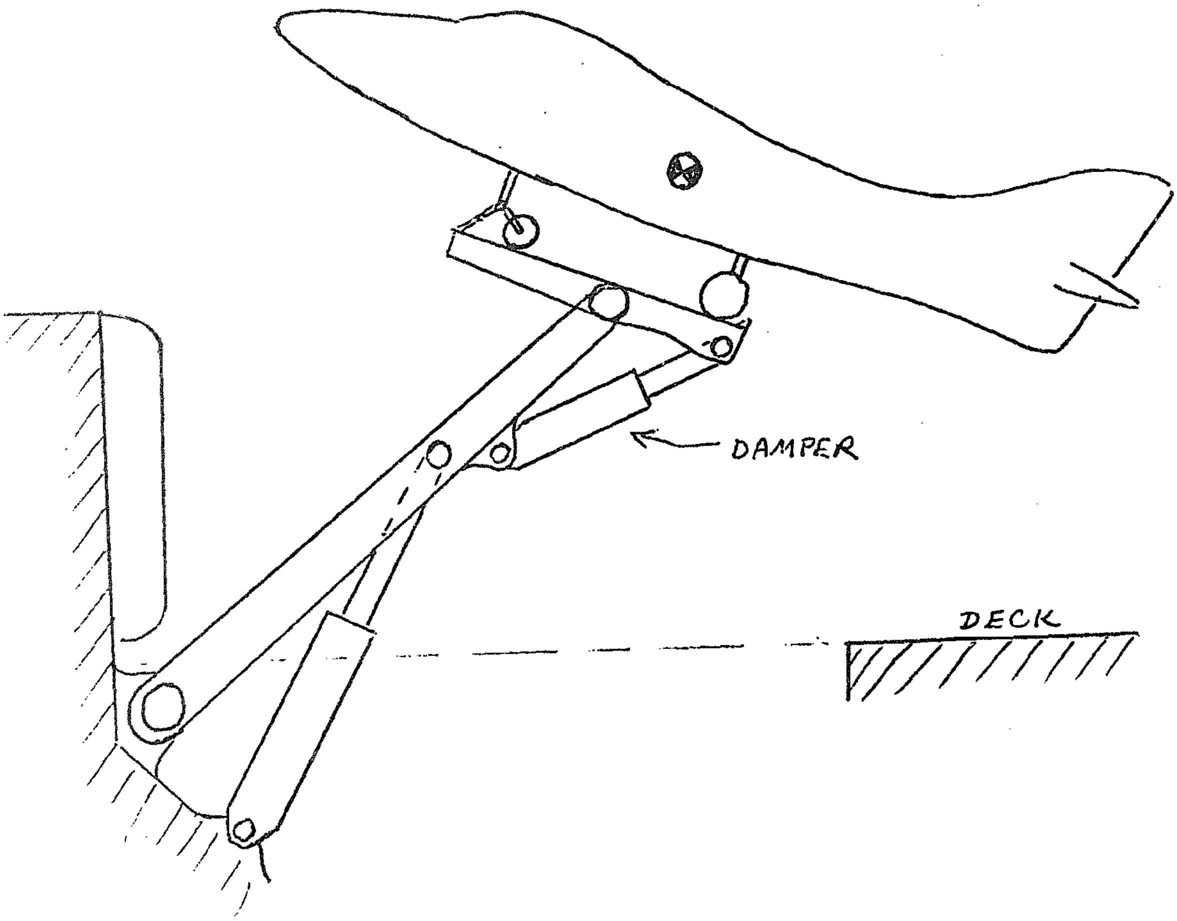


FIG 26

D. ALTERNATIVE METHOD OF AIRCRAFT ATTITUDE CONTROL.



2.10.5 An alternative method of controlling the attitude and rate of rotation of the aircraft is shown in Fig 26D. If the aircraft is suitably positioned its inertia will cause it to pitch nose-up during the launching process. The damper limits the rate and angle of pitch and retains the platform at the desired angle when this has been reached. In this method the functions of the rebound buffer can be combined in the main jack assembly.

2.10.6 Although the aircraft has been shown standing on a platform a further variation of the method is to have the arm and linkage (or damper) engaging in special strong points on the airframe. This would be advantageous if undercarriage strength imposed unacceptable limits.

2.10.7 Over the last 5° of launching motion the loads on the aircraft have a small negative value, ie the aircraft would, if unrestrained, lift clear of the platform. The aircraft is restrained at the front by the nose towing link. To prevent an undesirable pitching motion on the aircraft the other end must also be restrained. This restraint could be simply accomplished by an attachment to the main undercarriage leg which would allow forward motion (relative to the platform) but prevent rearwards or upwards motion.

2.10.8 As the aircraft leaves the launcher its attitude in space will be approximately the same as that of the platform, ie 30°. Its flight path however is at 40° to the horizontal. The resultant small negative angle of incidence (if there is no wind) has no significant effect on the subsequent trajectory due to the low airspeed and the fact that the incidence becomes positive after two seconds. The 30° platform angle is chosen in order to avoid stalling incidence when launching into a 30 knot wind.

2.10.9 The thrust deflection angle is selected before launch and there is therefore no need to make any allowance for pilot reaction time in the launch velocity. The thrust deflection angle relative to the horizon has been assumed to remain constant at 60° during the launching process.

2.10.10 In designing the ballista the object has been to make the launching mechanism as small as possible, ie the arm must be as short as possible. However for a given value of V_L, shortening the arm results in increasing loads on the aircraft and another objective has been to avoid, if possible, exceeding present structural limitations. The maximum acceleration therefore must not significantly exceed 4g and existing undercarriage limits should not be exceeded. Modification of the standard aircraft should be minimal but the use of a specially strengthened nose oleo for nose-towing is unavoidable since practically all the launching loads are carried through it.

For the above reasons the minimum practical lenth of the arm is 31 feet, giving a maximum launch weight (at T = 19,200 Lbs) of 24,000 Lbs with 30 knots wind over deck.

The following symbols have been used in calculation:-
(See Fig 27)

θ = Angle through which arm has moved.

ω = Angular velocity of arm.

α = Angular acceleration of arm.

R = Resultant acceleration of aircraft (in g units).

γ = Angle between arm and R.

β = Angle of R to horizontal.

R_V = Vertical Component of R.

R_H = Horizontal component of R.

L_T = Vertical component of engine thrust (in g units).

F = Horizontal component of engine thrust (in g units).

F_V = Resultant vertical reaction force between aircraft and platform (in g units).

F_H = Resultant horizontal reaction force between aircraft and platform (in g units).

r = Radius of aircraft C.G. from lower pivot.

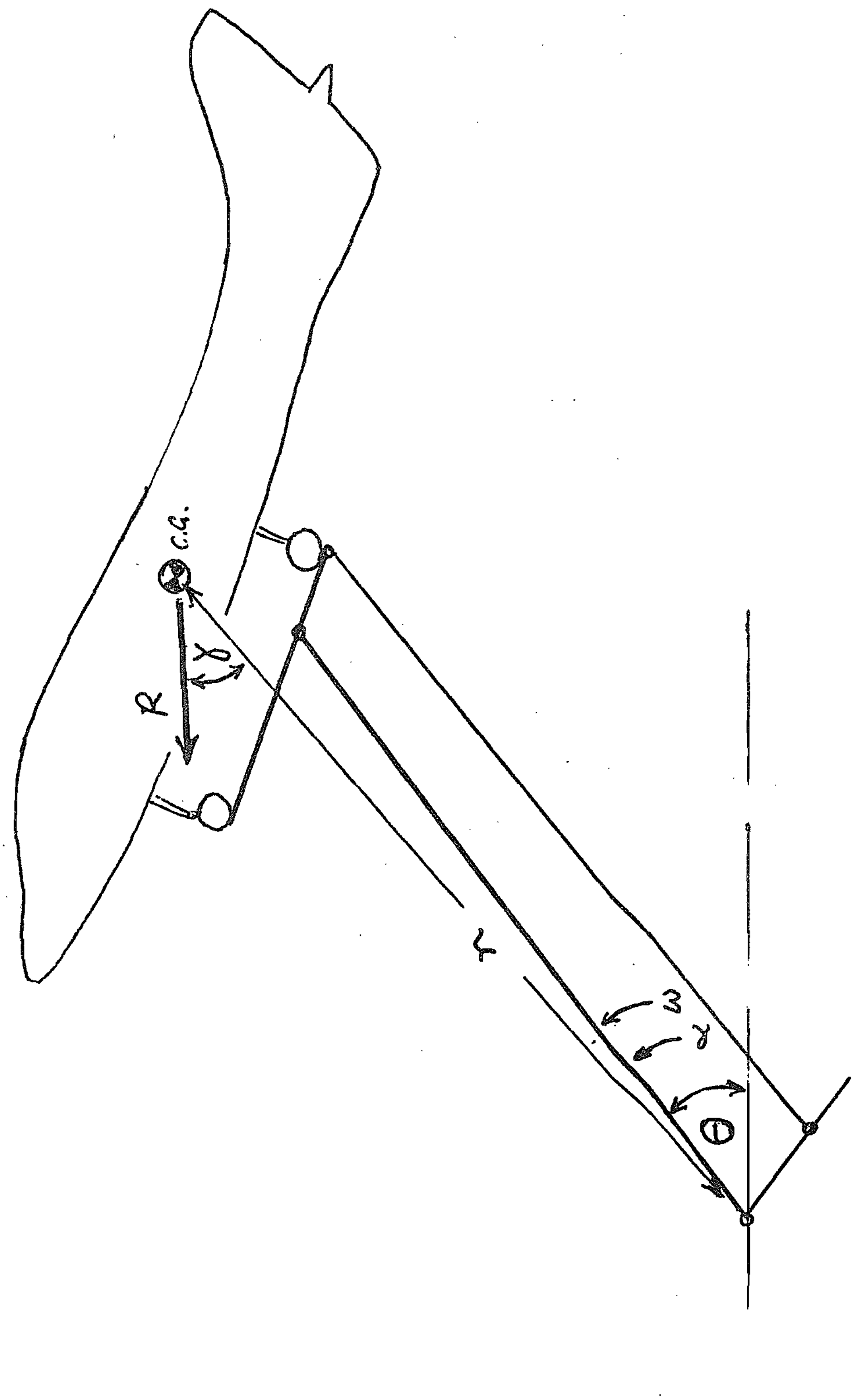


Fig 27

Due to the varying geomentry the value of r varies during the launching process. This variation was determined graphically and is as follows:

θ^o	r feet
10	33
20	34
30	34.5
40	35.4
50	35.8

$$R = r\sqrt{\alpha^2 + \omega^4}$$

and

$$\text{Tan}\,\gamma = \frac{\alpha}{\omega^2}$$

$$F_V = \left[(R_V + 1) - LT\right]$$

$$F_H = R_H - F$$

To launch a Harrier at W = 24,000 Lbs with w.o.d. = 30 knots requires V_L = 64 ft/sec at a launch angle of 40°. V_L = 64 ft/sec occurs when $\theta = 50^o = 0.8727$ radians. At this point, r= 35.8 feet.

$$\therefore \omega = \frac{64}{35.8} = 1.787 \text{ radians/sec.}$$

If the angular acceleration is assumed to be constant then,

$$\alpha = \frac{\omega^2}{2\theta} = \frac{1.787^2}{2 \times 0.8727} = 1.832 \text{ radians/sec}^2.$$

$$L_T = 19{,}200 \text{ Lbs Sin } 60^o = 16{,}627 \text{ Lbs} = \frac{16{,}627}{24{,}000}$$

$= 0.693\ g$

$F = 19{,}200 \text{ Lbs} \cos 60^\circ = 9{,}600 \text{ Lbs} = \frac{9{,}600}{24{,}000}$

$= 0.4g$

From the above, the following table can be calculated:

θ	ω	α	R	γ	β	R_V	R_H	F_V	F_H
0°	0	1.832	1.76	90°	90°	1.76	0	2.067	-0.4
10°	0.799	1.832	1.987	70.7°	60.7°	1.734	0.97	2.041	0.57
20°	1.13	1.832	2.358	55.1°	35.1°	1.357	1.928	1.664	1.528
30°	1.384	1.832	2.84	43.7°	13.7°	0.673	2.759	0.98	2.359
40°	1.599	1.832	3.457	35.6°	-4.4°	-0.263	3.447	0.044	3.047
50°	1.787	1.832	4.095	29.8°	-20.2°	-1.411	3.84	-1.103	3.44

The maximum acceleration of 4.095g is tolerable and the maximum vertical loading of 2.067g is within undercarriage limitations at this weight.

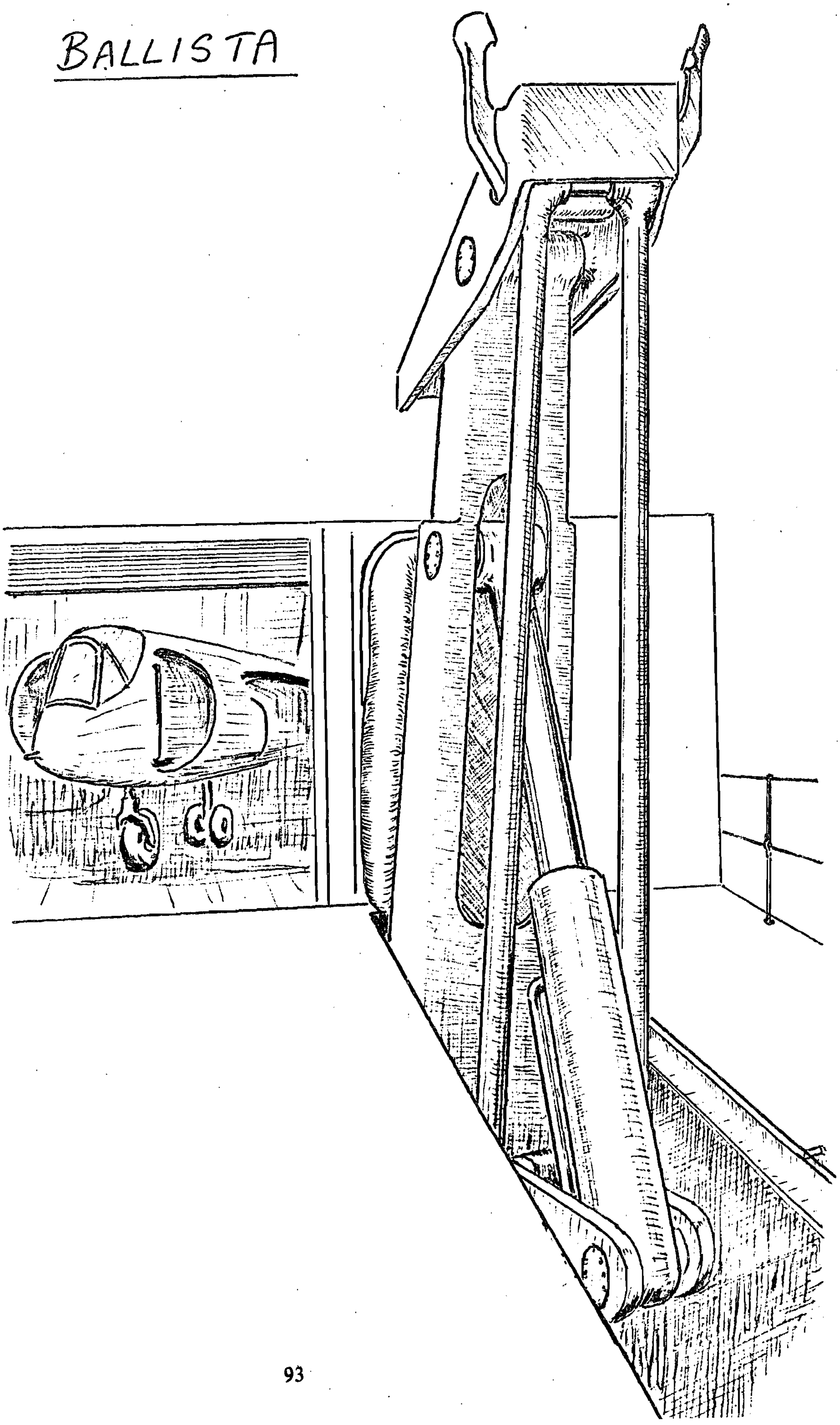
The launching capability of a Ballista with a 31 foot arm is therefore as follows:

24,000 Lbs at 30 kts w.o.d.
23,000 Lbs at 15 kts w.o.d.
22,400 Lbs in no wind

For Harrier with T = 19,200 Lbs at launch angle of 40°

FIG 28

<u>BALLISTA</u>



2.10.11 Stressing and weight calculations are at Appendices 9 and 10. No attempt has been made to design the mechanism in detail or to achieve the most efficient structures.
The object has been to obtain, by the simplest calculation, realistic figures of stress and weight to prove the feasibility of the device. To ensure that the result errs in the conservative direction it has been assumed that the material used is mild steel and a safety factor of at least 2 is applied.
The weight of the moving parts of the mechanism so designed is 5.57 tons. To this must be added the weight of a hydraulic accumulator, pump and prime mover and structural attachments to the ship. A total installed weight penalty of less than 15 tons therefore seems not unreasonable and well within the limitations of a small ship.

2.10.12 The Ballista comes close to being the ideal launching device and where space is an overriding factor it represents an irreduceable minimum. Compared with the steam catapult it is simple and cheap. Its main disadvantages are that it requires the aircraft using it to be specially modified and is perhaps not as readily adaptable to a wide variety of aircraft as the Ski-Jump. At present this is not a pressing requirement.
An impression of the Ballista as it might appear when installed in a ship is at Fig 28.

2.11 The Inclined Catapult

2.11.1 This is nothing more than a conventional catapult inclined at the required launching angle to the horizontal. The steam catapult is not the ideal for this application but almost any other conventional type of catapult could be used.

2.11.2 A Harrier of 24,000 Lbs weight launched at 30^{o} into a 30 knot wind requires a launch velocity of 69 ft/sec. If the catapult provides an acceleration of 4g then the launch velocity can be achieved in a catapult stroke of 18.5 feet. Allowing 3 feet for deceleration of moving parts gives a total catapult stroke of 21.5 feet.
The catapult can be in two forms:

(i) A permanent inclined structure.
This poses problems in loading the aircraft to the catapult. The aircraft would have to be hauled up the slope to the catapult. There would also have to be a curved approach ramp to avoid striking the deck with the tail of the aircraft. In addition a horizontal landing space would have to be provided. The total space required would be in the order of 120 feet. Such an arrangement would offer no significant advantages over the Ski Jump.

(ii) A pivoting structure which can be raised to the required launch angle.
This is the arrangement illustrated in Fig 29. In this case the aircraft is loaded to the catapult in the horizontal position and then the entire assembly is jacked up to the required launching angle.

2.11.3 This scheme introduces some problems with the catapult in that either the entire catapult mechanism must be jacked up with the aircraft and associated structure or else the catapult mechanism remains in the fixed ship's structure and transmits its power via the pivot point of the inclined assembly.
The problems are not insuperable but their solution will inevitably involve an unwelcome increase in complexity and weight.
A further disadvantage is the extra stage in the launching process in jacking the catapult up to the launching angle which will result in increased cycle times.
Despite these disadvantages the Inclined Catapult fulfils many of the ideal requirements and the space required approaches the minimum attainable.
Its advantages are that it can make use of existing equipment and knowledge, it imposes no undesirable pitching motion on the aircraft and is not limited by undercarriage strength.

It is, therefore, listed among the preferred solutions to the launching problem.

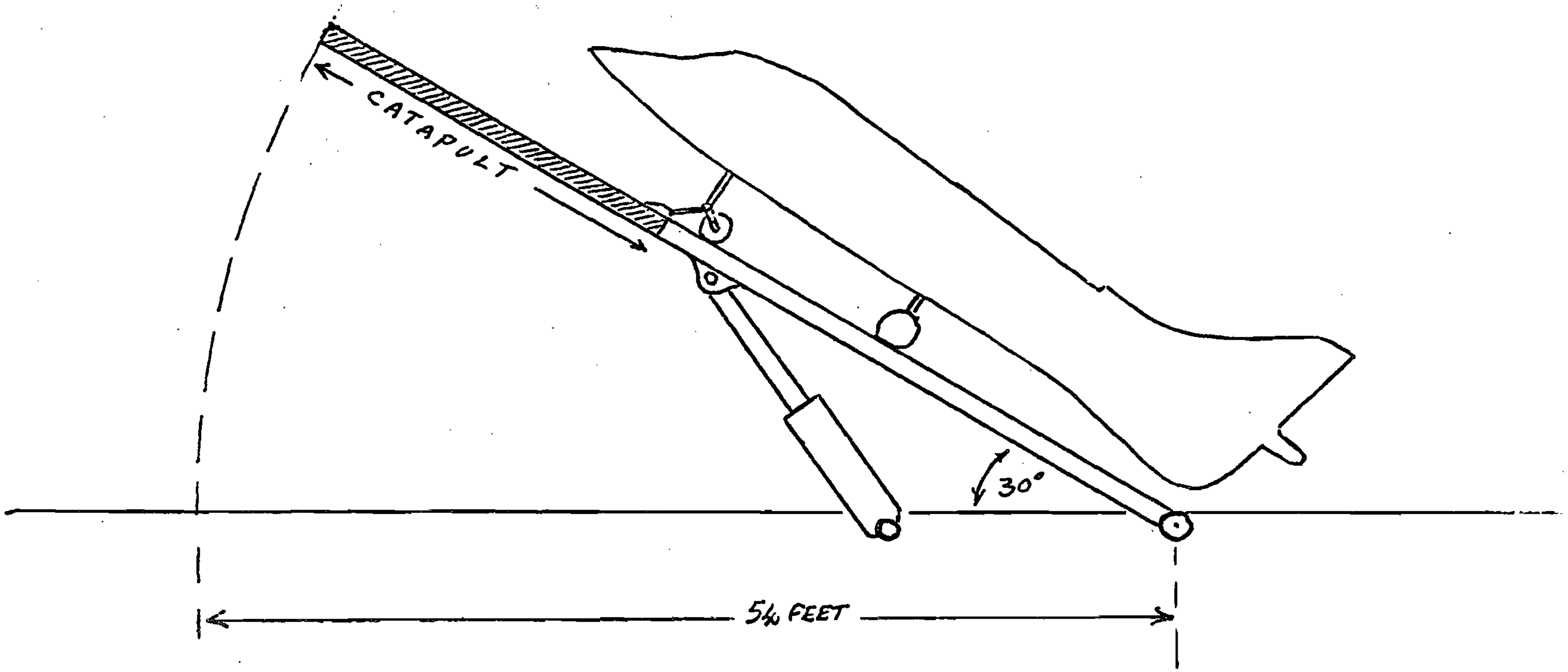
FIG 29

THE INCLINED CATAPULT

SCALE 1" = 10'

TO LAUNCH :-

22000 Lb AT W.O.D. = 0.
23000 Lb AT W.O.D = 15 KTS
24000 Lb AT W.O.D. = 30 KTS.



2.12 Comparison of Launching Methods

2.12.1 The devices examined in the preceding pages, the Ski-Jump, Catapult Assisted Ski-Jump, the Inclined Catapult and the Ballista are those considered the most likely to yield an effective improvement in the performance of V/STOL fixed wing aircraft operating from confined spaces.

No exotic materials or advanced engineering techniques are required for their construction and by comparison with existing launching methods they are cheap and in most cases, simple.

2.12.2 No single one of them emerges as a clearly preferred solution, all achieve dramatic reductions in space required by comparison with the currently established rolling take-off technique.

Of the four, the Ski-Jump by virtue of its extreme simplicity and the fact that no aircraft modification is required can most readily be the subject of experiment and early application.

If minimum space is a paramount requirement then the choice lies between the Ballista and the Inclined Catapult. Of the two the Ballista is, in the long term likely to emerge as the smaller, lighter and simpler; it should be more adaptable and launching intervals will be shorter. On the other hand the Inclined Catapult is the safe solution since it embodies nothing new.

The best solution may well be a compromise, the Catapult Assisted Ski-Jump which combines small space with a limited launching capacity if the catapult is unserviceable.

2.12.3 Ultimately choice depends on the space which can be made available and different methods may be used in different applications. This will be examined in more detail in Chapter 4.

2.12.4 It is again emphasized that take-off weights and distances have been calculated on the basis of an installed thrust of 19,200 Lbs. Performance in semi-ballistic flight is very largely a matter of thrust to weight ratio and this also mainly determines the launch velocity.

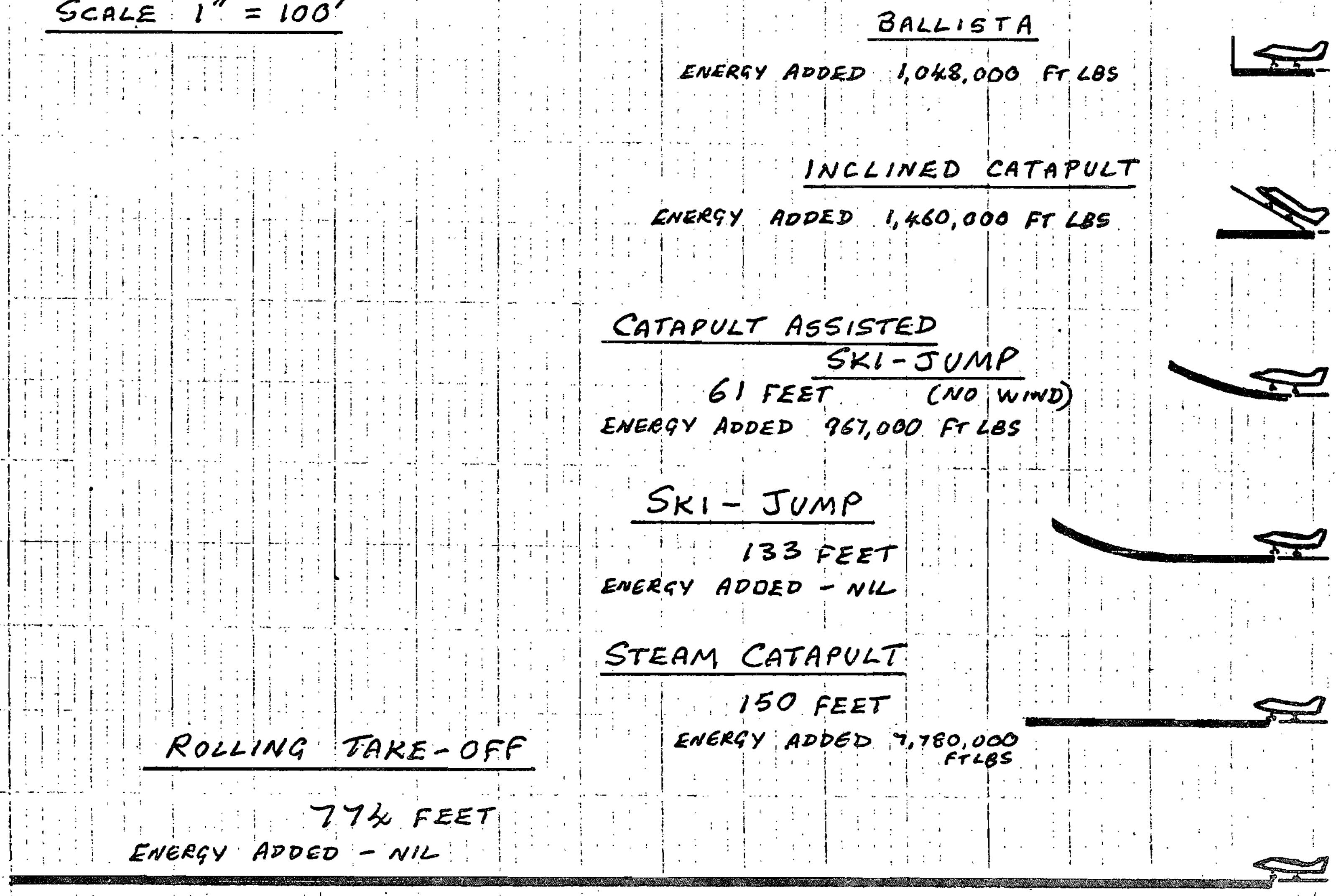
If, then, under given conditions a launch weight of 24,000 Lbs is quoted this represents a thrust/weight ratio of 19,200/24,000 = 0.8. If the installed thrust were increased to, say, 21,500 Lbs then the launch weight in the same conditions would increase to something approaching 21,500/0.8 = 26,875 Lbs.

2.12.5 Fig 30 provides a pictorial comparison of the performance of the various launching methods. It shows the space required for take-off and the energy which has to be added to the aircraft by the launching device.

HARRIER – COMPARATIVE TAKE-OFF PERFORMANCE

WEIGHT 23000 lbs, WIND OVER DECK 15 KNOTS.

SCALE 1" = 100'



2.13 Independence of Wind Speed and Direction

2.13.1 As stated in 2.3 the need to work up to a high ship speed or turn into the natural wind is a tactical limitation. Natural, or ship generated wind is a fact of life which cannot and in practise, never will be ignored in operating aircraft. The benefits accruing from its proper exploitation are too great. Most of the time, over most of the sea, the wind blows and will be used.

All the devices considered above are capable of launching aircraft at useful weights in conditions of no wind. A measure of independence of wind strength has therefore been achieved.

2.12.3 Independence of wind direction requires that the launching device be capable of rotation in the horizontal plane. This could be done quite easily with the Ballista and Inclined Catapult. Even the Ski-Jump could be arranged to swing over quite a wide arc in larger ships. The question is; is the additional complication of a rotatable mounting worthwhile? Those who base their judgement on operational experience in large carriers are likely to give an unhesitating affirmative. In a conventional aircraft carrier launching and recovery must be conducted with the ship heading into the natural wind and with ship speed sufficient to give about 30 knots of wind over the deck or more. Launching and recovery are usually consecutive operations; typically a ship may launch 10 aircraft and recover 10 on each occasion. With two catapults and well drilled crews launching may be completed within 6 minutes. Recovery will have started while the last few aircraft were being launched and is unlikely to take less than a further 10 minutes. During this period the ship might have been doing 20 knots and could therefore be up to 5 miles from the mean line of advance.

By contrast a small ship is unlikely to launch more than four aircraft at a time and assuming it has only one launching device this will occupy only 6–8 minutes at most. Its speed through the water need not be high so that its deviation from the line of advance is small. The landing of V/STOL aircraft is not dependent on wind speed or direction so that aircraft may be recovered as the ship returns to its planned course. Furthermore the ship operating V/STOL aircraft is more flexible, launch and recovery need not be consecutive and may more readily be varied to suit the tactical situation.

It therefore seems not unreasonable to conclude that, for V/STOL aircraft in small ships, the ability to rotate the launching devices is something of a luxury, scarcely justifying the additional complication and expense.

CHAPTER 3

Launching Methods considered but rejected

3.1 RATOG (Rocket Assisted Take-Off).

3.1.1 The use of rockets to assist the take-off of a heavily loaded aircraft or to reduce the take-off distance is a well proven method and was at one time regularly used in aircraft carriers. It could readily be applied to a V/STOL aeroplane such as the Harrier to achieve effectively a vertical take-off at low thrust/weight ratios.

3.1.2 To achieve a payload increase of, say, 2000 lbs over the payload for an unassisted vertical take-off would require a rocket of 3500 Lbs thrust with a thrust duration of about 20 seconds. Loads of the order achievable by the methods already considered would require correspondingly larger rockets.

3.1.3 Although it is technically feasible there are a number of practical difficulties associated with the use of RATOG, these are:

(i) Cost.

A solid fuel rocket of the performance required would cost close to £1000 per round. Arrangements could be made to reduce the cost by recovering the rocket cases and nozzles but this would introduce either undesirable complication or unacceptable limitations (eg stopping the ship or launching the helicopter). The alternative of retaining the rocket case in or on the aircraft introduces weight, volume and drag penalties which would almost certainly be unacceptable.

(ii) Rocket efflux.

The exhaust gases from the rocket are at a much higher temperature and of much greater velocity than the jet efflux of the aircraft. There is therefore more danger to personnel, light structures, aerials and inflammable material in the take-off area. In a small ship it would be difficult to provide adequate protection.

(iii) External Stores.

The fitting of RATOG in or on the aircraft will occupy space which could be used to carry external stores. The loss of the centre line store position for instance on the Harrier would impose an operational limitation.

(iv) Storage Space.

If RATOG were to be used regularly a large number of spare rocket motors would have to be carried in the ship in magazine conditions. In a small ship this requirement alone is likely to be prohibitive.

3.1.4 A partial solution to the problem may lie in the use of a water rocket. This consists of a pressurized container filled with water at high temperature and pressure. When the contents are allowed to exhaust to atmosphere through a nozzle the water flashes off into high velocity steam. This type of rocket would reduce the efflux problem since temperatures would be lower. If the containers could be recovered the storage problem would be almost eliminated and it is possible that sea water could be used as the propellant. The cost is therefore likely to be much less than that of a solid fuel rocket. Its size and weight however is likely to be considerably greater than the equivalent solid fuel rocket. A severe disadvantage is the amount of preparation required before the rocket can be used.

3.1.5 Although the water rocket appears to be quite a promising method it has no significant advantages over competing methods. The use of RATOG as a launching method is therefore rejected.

3.2 The Flexible Launcher

3.2.1 This device was invented to meet the requirement of minimum space. If the launching device is to occupy no more space than is required for a vertical landing it is almost inevitable that the launcher must exert its thrust on the aircraft from behind and below. There are many ways of providing the thrust but any launching device which remains attached to the ship comes up against the fundamental difficulty of stopping the moving parts on completion of the launching stroke.

This is an especial difficulty if hydraulic power is used as the weight of the fluid is added to that of the mechanical parts. If compressed gas is the source of motive power the weight of a piston of the required length and rigidity is still sufficient to make retardation a severe problem.

3.2.2 The solution is to make the gas provide both the propellant force and the rigidity by the use of a flexible inflated structure to transmit the propulsive force.

The principle on which the Flexible Launcher operates is illustrated in Fig 31. A tubular bag passes between two rollers and at this point it is flattened, the two sides of the bag being pressed tightly together to form a gas tight seal. One end of the bag is sealed and carries a gas generator. When compressed gas is admitted to the driving bag it inflates and the inflated portion then becomes a rigid strut. The pressure in the bag turns the rollers which allows the flattened portion of the driving bag to pass between them and become inflated. The bag thus becomes in effect a continuously extendable rigid strut. If gas pressure is maintained constant during extension, the bag will also exert a longitudinal thrust approximately equal to the internal pressure multiplied by the cross section area of the bag. Since every part of the bag is in tension it can be very light, the only rigid structure required is at the end of the bag where the thrust has to be transmitted to the aircraft. Due to the light weight of the moving parts, their retardation is a much more amenable problem, the bag itself has ample strength to withstand being brought to rest in, say 5 feet.

3.2.3 Fig 32 shows the device as it might be used in practise to achieve a launch at 30^o to the horizontal. In the form shown the Flexible Launcher could launch a Harrier weighing 24000 Lbs given 30 knots wind-over-deck., 23000 Lbs in 15 knots wind-over-deck and 22000 Lbs in no wind. If a bag of 5 feet diameter is used an internal pressure of 28 Lbs/in^2 is required to provide the thrust for 4g acceleration and a launch stroke of 18.5 feet gives the required launch velocity of 69 ft/sec. This situation at the end of the launching stroke is shown dotted in the diagram. A further 5 feet of stroke brings the

FLEXIBLE LAUNCHER

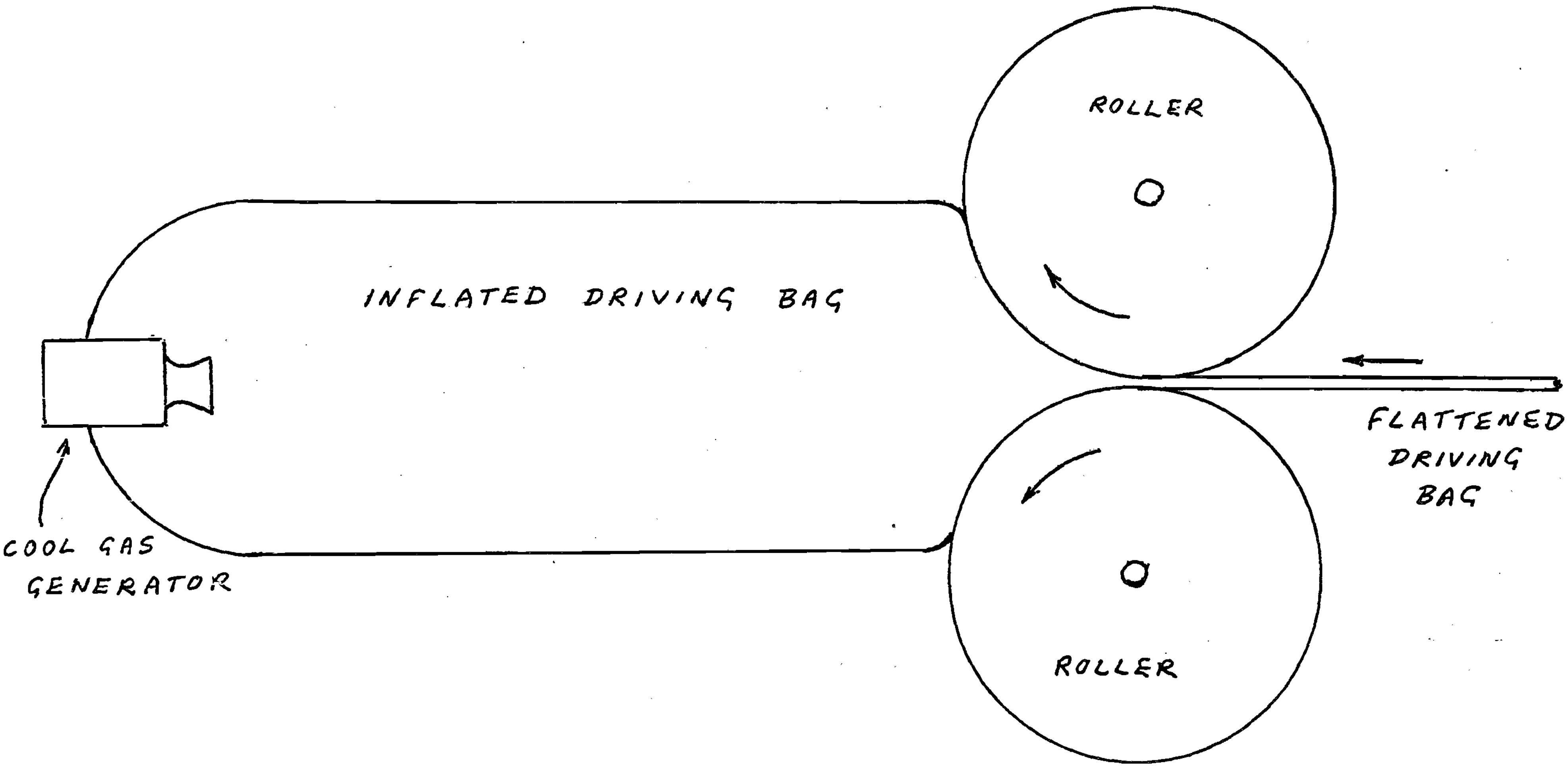
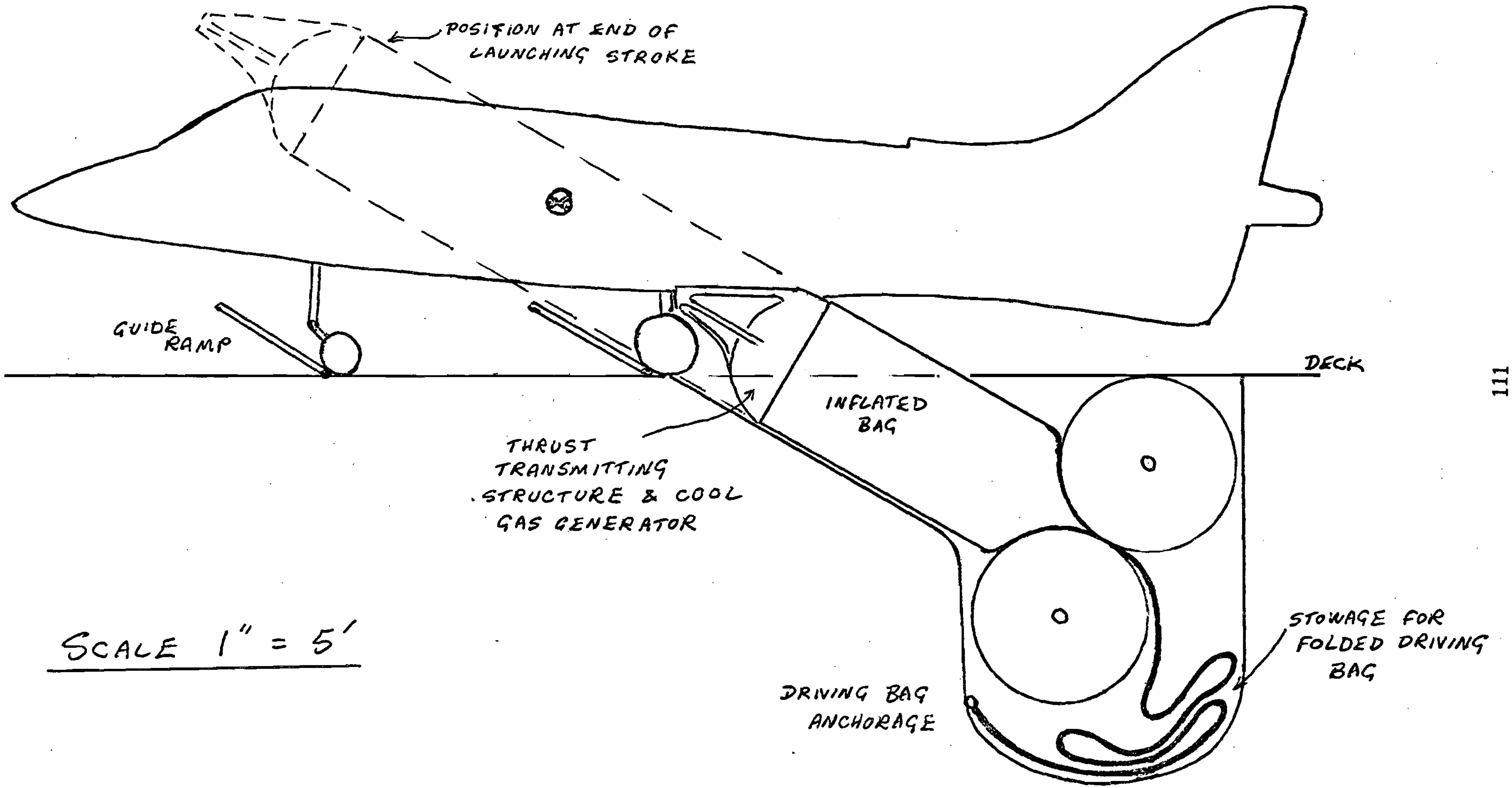
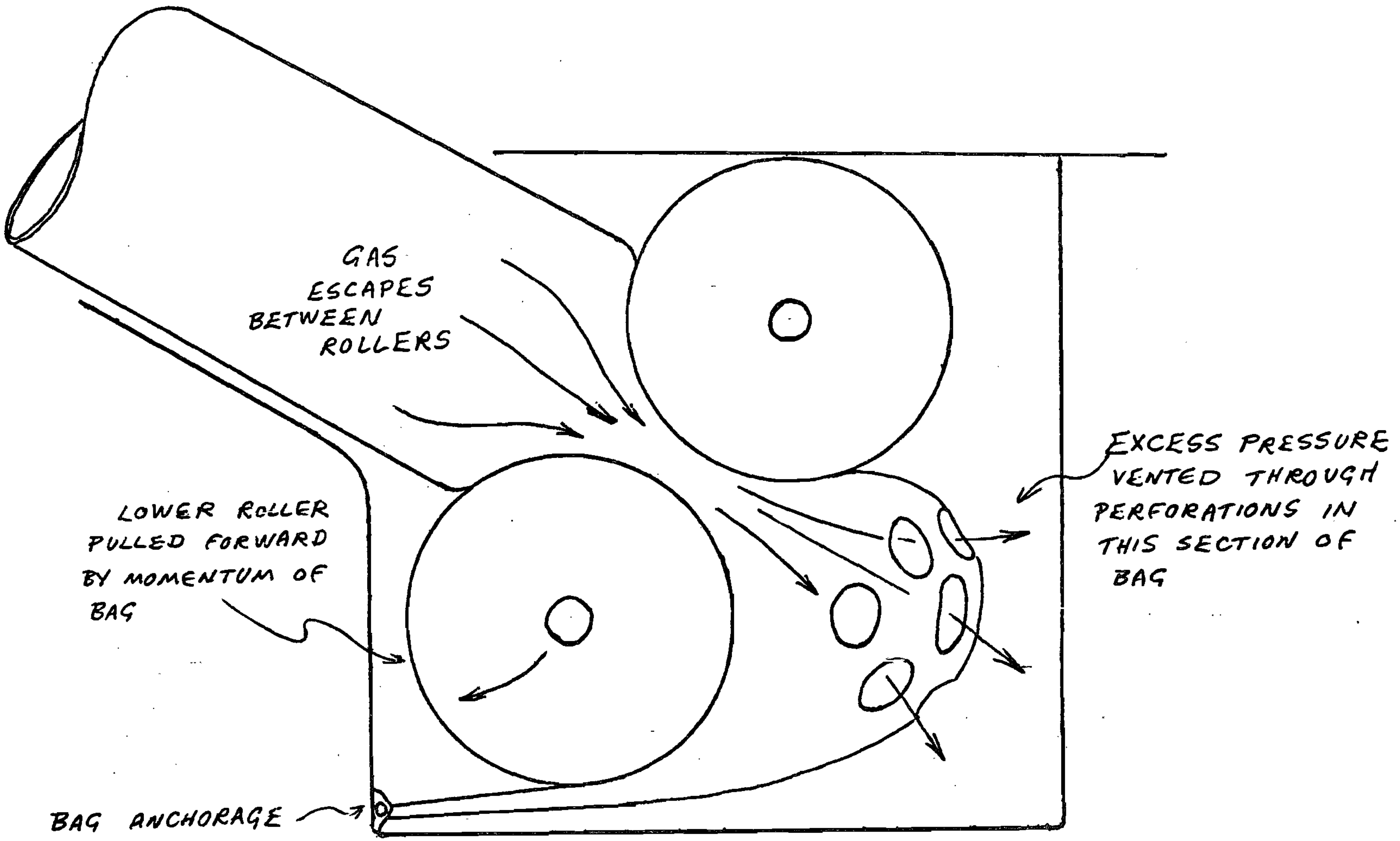


FIG 32 FLEXIBLE LAUNCHER



SCALE 1" = 5'

FLEXIBLE LAUNCHER - DECELERATION OF MOVING PARTS



moving parts to rest. Although 28 Lbs/in^2 is sufficient pressure to provide the thrust the bag pressure must be increased by 12 Lbs/in^2 in order to make the bag rigid enough not to buckle under the weight of the aircraft. Total bag pressure is therefore 40 Lbs/in^2.

3.2.4 This would result in excessive thrust unless the extension of the bag is controlled in some way. This could best be accomplished by a continuous braking of the rollers to control the rate of acceleration of the aircraft.

3.2.5 There are many ways of bringing the moving parts to rest. Perhaps the simplest - and most brutal - method is the use of friction brakes on the rollers. A more subtle method would be to allow the lower roller to swing forward and downward against a shock absorber under the pull of the remaining hitherto uninflated portion of the bag as shown in Fig 33. This would allow the excess pressure in the bag to pass between the rollers and exhaust.

3.2.6 It is envisaged that the bag would be a composite woven structure proofed with neoprene or similar material to make it gas tight. Nylon is a very strong lightweight material but in woven form is somewhat unpredictable in stretching. The bag must be kept straight and it is therefore suggested that nylon should be used to absorb hoop stresses and steel wire for the longitudinal stresses, albeit at some weight penalty. The weight of a bag so constructed has been calculated at 1100 Lbs with a further 300 Lbs added as the thrust transmitting structure and cool gas generator. Calculations of thrust and weight of the bag are at appendix 11.

3.2.7 The Flexible Launcher is a most attractive solution to the launching problem. For a given performance it promises a much lighter weight and smaller dimensions than the Ballista and for these reasons it is considered that the concept should not be abandoned in the long term. For the immediate future however it cannot be recommended for the following reasons:

(a) The production of large flexible structures is, at present, more art than science. There is no background of experience of composite, flexible structures of this size carrying pressures of the order of 40 Lbs/in^2. In addition the bending of the structural fibres through 180° where the bag passes between the rollers, will probably give rise to problems. While none of these problems is insuperable the research and development required is likely to be lengthy and expensive.

(b) The cost of the driving bag in production is likely to be relatively high and its life, in service, may be short.

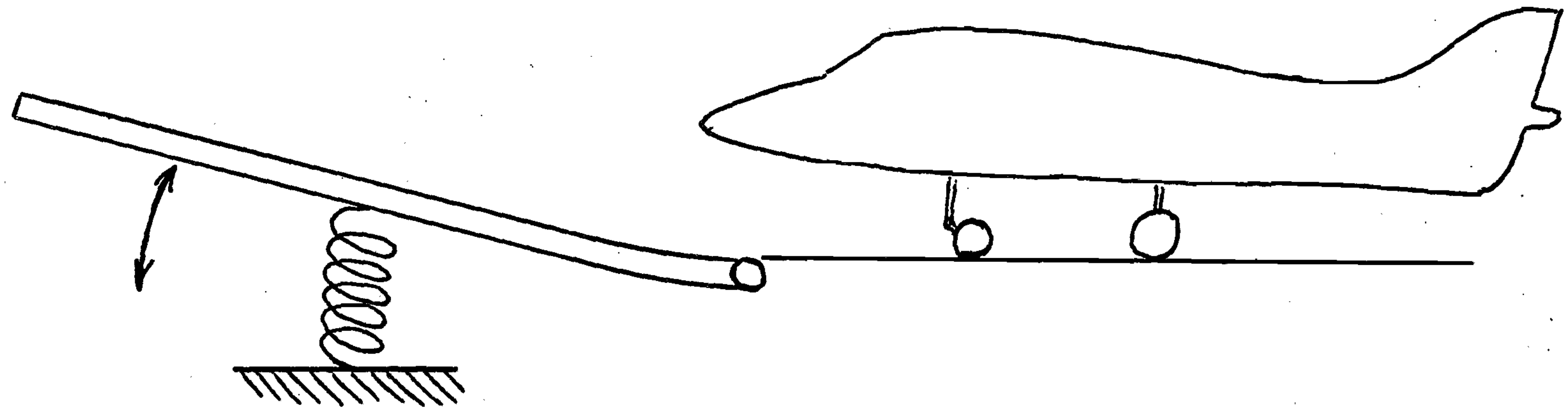
(c) The need to retract and stow the bag, avoiding creases, between launches is likely to result in increased cycle time compared with other launching systems.

(d) Extensive strengthening of the aircraft will probably be necessary around the area through which the thrust is transmitted to the aircraft.

3.2.8 Although not ideal as an aircraft launching device the Flexible Launcher would appear to be immediately applicable in a related field, namely the launching of guided missiles. Here its disadvantages largely disappear. The weight to be launched is smaller, so the bag is much smaller. Higher accelerations may be used. A short bag life is acceptable, indeed one shot only may be adequate. Deceleration of moving parts is either not required or much simpler. As a positive displacement device it is much more efficient than the booster rockets currently employed, so that initial boost rockets can be eliminated or greatly reduced in size with favourable effects on the ships magazine capacity. The attendant problems of rocket efflux, flash etc are thereby eliminated. The advantages to be gained by the employment of the Flexible Launcher as a missile launcher would appear to be such as to justify further research.

FIG 34

THE SPRINGBOARD



3.3 The Springboard and The Powered Springboard

3.3.1 These concepts arose from an original suggestion by Messrs T Jordan and K Causer of Hawker-Siddeley Aviation Ltd. Fig 34 shows the principle of operation of the springboard.

3.3.2 The method of operation is similar in some ways to the Ski-Jump. The aircraft accelerates horizontally under its own undeflected thrust until it strikes the curved entry to a ramp. This ramp pivots vertically about one end and is supported by a spring. When the aircraft strikes it the ramp is deflected downwards compressing the spring. The aircraft continues to accelerate along the ramp which, in due course, commences to move up again as the spring reasserts itself thus imparting a vertical velocity to the aircraft.

3.3.3 The advantage of the method is that if the length of the springboard and the spring rate are correctly chosen the resultant launch angle of the aircraft may be much greater than the original angle of inclination of the springboard. At the same time the aircraft itself has been rotated through a much smaller angle in pitch and pitching rate is relatively modest.

However preliminary examination indicated that the total space required would not be very much less than the Ski-Jump while complexity would be greater. A further disadvantage is that the space under the ramp is not usable. For these reasons the idea was not pursued further in this form.

3.3.4 A logical development of the idea was to supply energy to the aircraft through the springboard so that the aircraft supplied the horizontal component of launch velocity from its own thrust while the vertical velocity was derived from an external power source.

The result was the Powered Springboard illustrated in Fig 35.

This is considerably modified from the original concept. The aircraft starts its run from the pivot point of the board which, initially, is inclined downwards to enhance the acceleration of the aircraft. During the course of the aircraft's run the board is thrust upwards by a jack thus adding the vertical component of velocity.

3.3.5 The following symbols and formulae were used in calculation:

t = elapsed time from start of aircraft's run.

α = angular acceleration of board.

ω = angular velocity of board.

θ = angle through which board has moved.

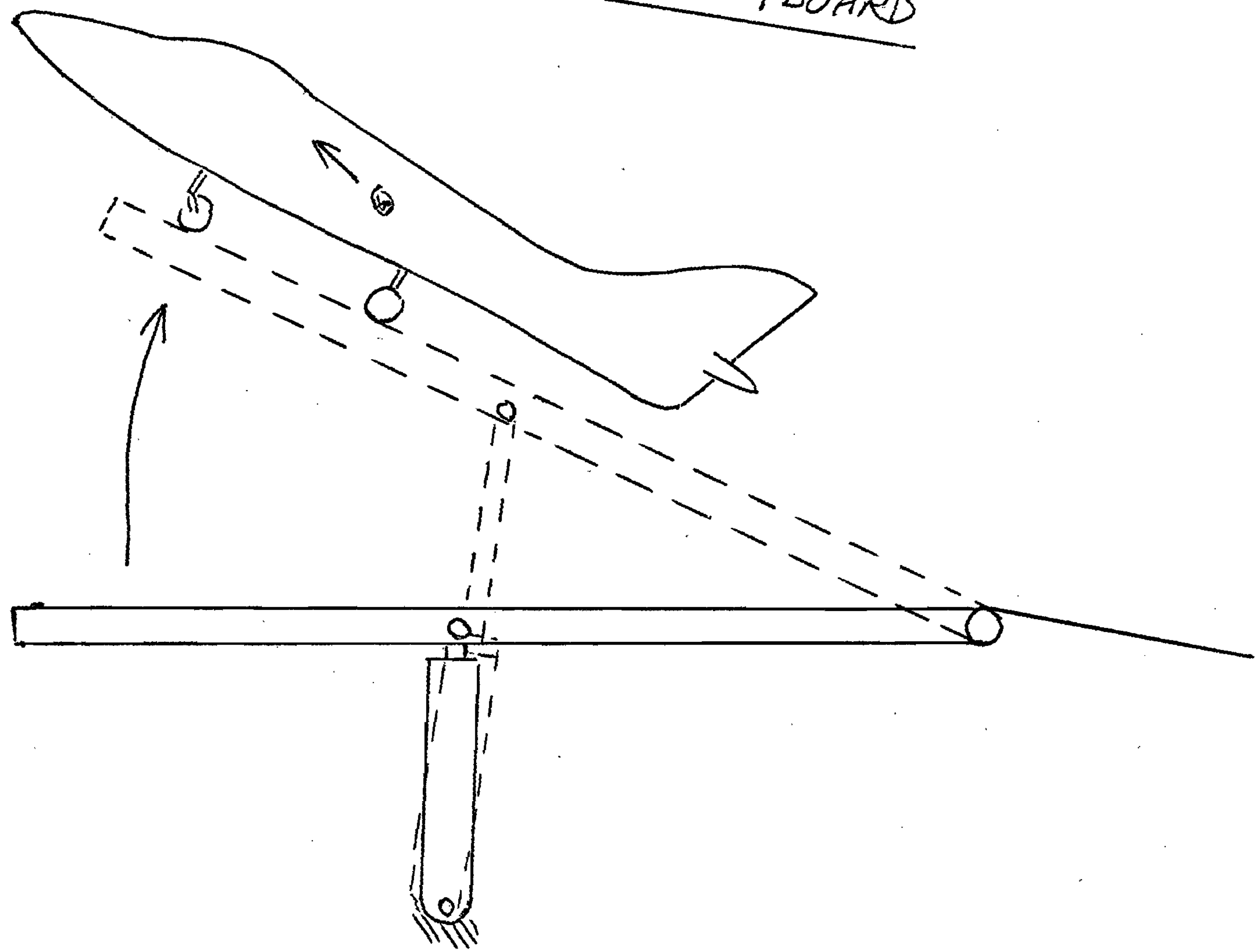
θh = angle of board to horizontal.

a = acceleration of aircraft along the board.

u = velocity of aircraft along the board.

FIG 35

THE POWERED SPRINGBOARD



r = distance travelled along board.

C = coriolis component acceleration of the aircraft.

T = aircraft thrust.

W = aircraft weight.

The acceleration of the aircraft relative to its starting point is the resultant of the coriolis component acceleration and the acceleration along the board.

At any instant;

$$a = \frac{g}{W}\,(T - W \sin \theta h)$$

and

$$C = 2\omega u + \alpha r.$$

3.3.6 There are too many unknowns for a general solution but practical considerations give rise to constraints which, to some extent at least, narrow the range of the unknown quantities and allow a reasoned choice of values.

These constraints are as follows:

(i) θ must be kept small because the vertical components of acceleration and velocity vary as Cos θ. At large positive values of θh (ie above the horizontal) there is a significant horizontal component opposing the horizontal motion of the aircraft. An arbitary limit of 20° was therefore selected for θ.

(ii) Since θ must be small the vertical velocity must be achieved by compressing the rotation of the board into a short interval of time towards the end of the aircraft's run along the board, ie when r is large.

(iii) Values of C must be such that the reaction forces do not exceed the undercarriage strength.

(iv) Between the time when the aircraft nosewheel leaves the end of the board and the time when the main wheels leave, the value of C must be such that the resultant pitching motion of the aircraft lies between 0 and 6^{o}/sec nose up.

(v) From the instant that the main wheels leave the board ω must be less than 0.05 radians/sec to avoid striking the board on the tail of the aircraft.

3.3.7 From (i) and (ii) above it would be best if rotation of the board was delayed until towards the end of the aircraft run. From a practical point of view however it is highly desirable that the first movement of the aircraft initiates the rotation of the board. The alternative therefore is to keep the angular velocity of the board very low until the end of the run.
Another result of (ii) is that it is difficult to keep the loads on the undercarriage within limits during the

brief period during which most of the rotation of the board must occur.

The greatest difficulty arises from (iv) and (v). From consideration of (ii), peak values of ω occur very close to the end of the run and must be followed by a very rapid reduction in ω to meet the requirements of (iv) and (v). The resulting values of α are very large indeed.

The method of calculation ultimately adopted was a trial and error process. First the maximum and minimum values of ω arising from the constraints were determined. Using these as a guide an arbitary curve of ω against t was drawn such that the area under the ω/t curve would result in a maximum value of θ, at the end of the run, of approximately 20°.

α could then be obtained from the slope of the ω/t curve. This gave enough information to calculate a and u could be determined from the area under the resulting a/t curve. Hence, C could be calculated.

3.3.8 A number of combinations was tried. Figs 36 and 37, show some of the best results achieved in attempting to launch a 22000 Lbs aircraft at 30° launch angle with no wind over the deck.

In this instance the aircraft nose wheel leaves the end of the board at t = 1.66 secs. The length of the board is nearly 50 feet.

A take-off distance of 50 feet is attractive but, as shown in Fig 36, very high angular accelerations will be necessary and the imposition of these accelerations on a 50 foot structure capable of carrying the aircraft will require very large forces. The Powered Springboard is therefore not a practical proposition.

Fig 36

POWERED SPRINGBOARD

ANGULAR DISPLACEMENT, VELOCITY AND ACCELERATION.

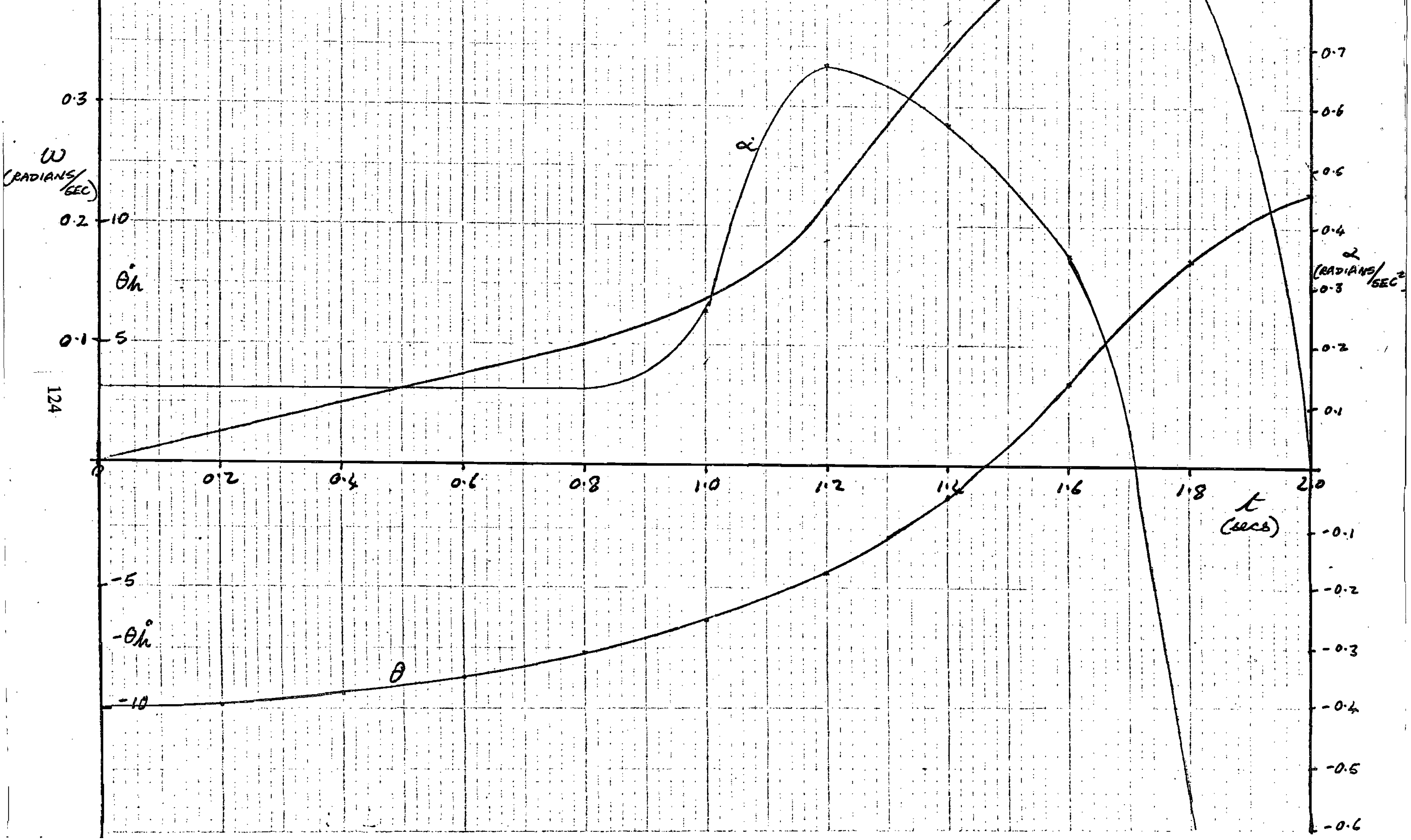
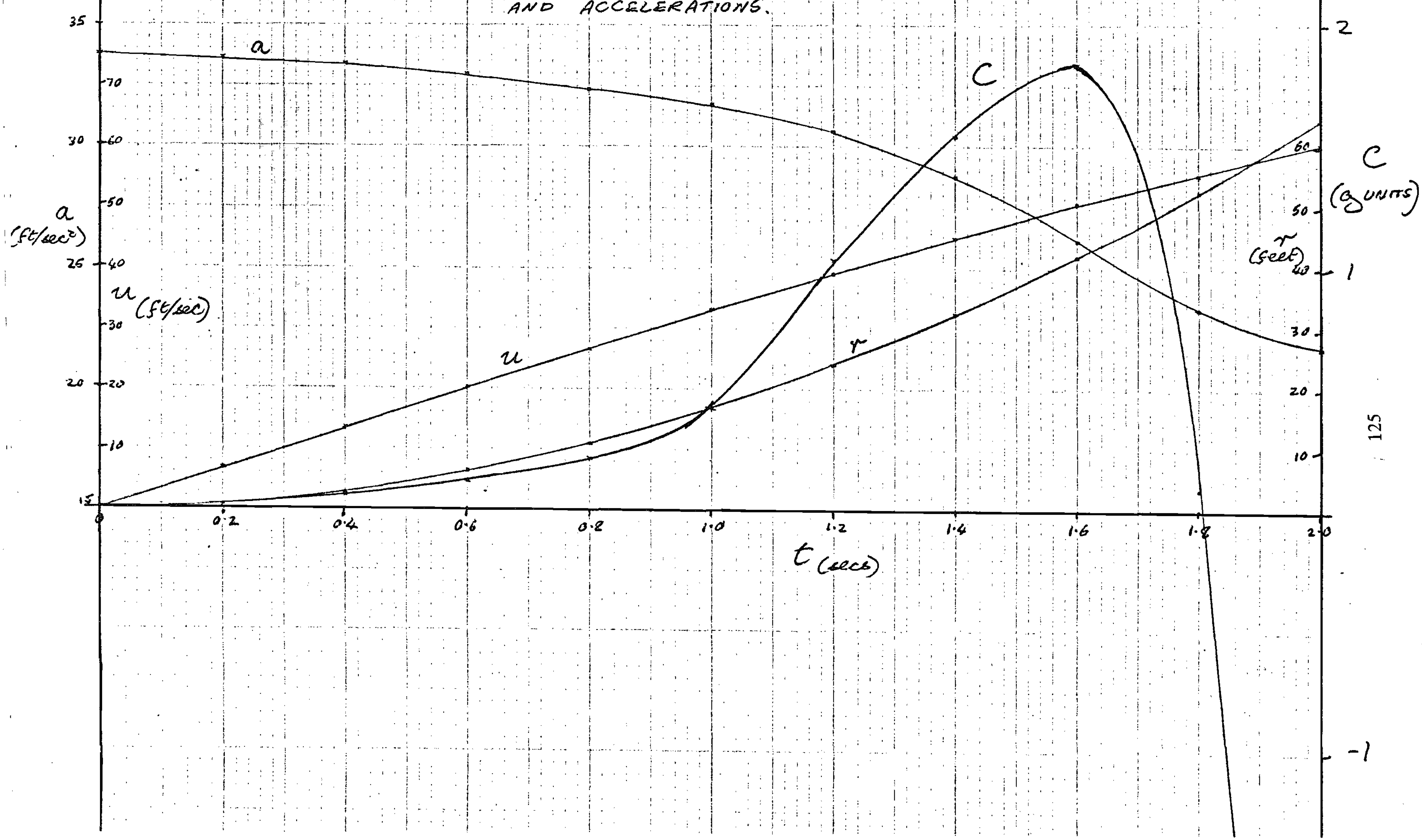


FIG 37

POWERED SPRINGBOARD

AIRCRAFT DISPLACEMENT, VELOCITY AND ACCELERATIONS.



CHAPTER 4

Application of Proposed Launching Methods to Seagoing Ships

4.1 The operation of any aircraft from any ship is an immensely complicated affair when compared with the same business conducted on land. The complications are all interlinked round the two common factors of ship motion and lack of space. The smaller the ship the more critical these factors become until ultimately they are prohibitive.

4.1.1 Ship motion mainly affects take-off landing and manouvering the aircraft on deck. It also affects certain maintenance operations such as those involving jacking the aircraft or the handling of heavy objects as, for instance, an engine change.

4.1.2 Shortage of space affects everything. Take-off and landing obviously, also hangarage. Less obviously but equally certainly it complicates the handling and maintenance tasks. An aeroplane is a bulky and awkwardly shaped object. It is also heavy, relatively flimsy and expensive. It cannot be allowed to get out of control on a rolling, pitching deck yet it must be manouvered within inches of its neighbours or hard structure.

Defects occur unpredictably and so it frequently happens that maintenance on an aeroplane must be interrupted while it is moved to make way for a serviceable aircraft stowed further in the hangar. This aircraft shuffling is a major delaying factor and any ship design must seek to minimise it. Workshop facilities will be required. It must be possible to refuel or defuel the aircraft whatever their location. They must be armed and re-armed. They will also require liquid oxygen which must be stored and possibly manufactured on board.

4.1.3 The above problems are formidable enough but so far only the machine has been considered. Men fly and maintain it and men are great consumers of space. They must be housed, fed, clothed and bathed. Though they are small in size and light in weight their accommodation takes-up a great deal of space and much of it must be air conditioned so that they can perform efficiently. The men therefore must be few in number and this means that each must encompass a wide range of knowledge and skills.

4.1.4 The fact that emerges is that each addition of an aircraft to a ship makes a relatively small demand on it for weight and a large demand for space. Space therefore is the overriding consideration since ship motion is largely dependent on size and space requirements dictate the size. In all but extreme sea states space is likely to become a critical factor before ship motion.

4.1.5 We require to operate V/STOL aircraft from ships of Frigate size and so the application of a Harrier to a typical Frigate is the first subject to be studied. It is also the most crucial since in this size of ship the problems of ship motion and space are approaching the limit in severity.

4.2 Effect of Ship Motion

Ship motion is a complex subject and a detailed analysis is beyond the scope of this paper. As an alternative, arbitary expressions and values are prescribed which, if met, should provide a satisfactory performance from a Frigate.

The required conditions are that the aircraft should be able to take off and land under the following ship motions:

Roll $\pm\ 5^{o}$

Pitch $\pm\ 2^{o}$

Yaw $\pm\ 1\frac{1}{2}^{o}$

Roll period is 10 seconds and the rolling axis is 13 feet below flight deck level.

The pitching period is 5.5 seconds and the pitching and yawing axes are 167 feet forward of the take-off and landing point.

Rolling velocities and accelerations are 1.5 times those produced by Simple Harmonic Motion of the same amplitude.

The SHM formulae:

$$V = 2\pi f\sqrt{A^2 - x^2}$$

and

$$a = -(2\pi f)^2 x$$

where A = Amplitude = maximum displacement.

f = Frequency = $\frac{1}{\text{Period}}$

x = Displacement

v = Velocity

a = Acceleration

were used to calculate the results which follow.

4.2.1 Effects of Roll

The Semi-Ballistic launching technique gives the aircraft a vertical component of velocity during the launching process. If the ship is rolling at the same time the aircraft will suffer a coriolis component accleration which will result in a lateral force upon it. This force will be at its maximum at the end of the launching process when the launch velocity is achieved and will depend also on roll rate and acceleration according to the formula

$$C = 2\omega V_0 + \alpha r$$

where,

C = coriolis component acceleration.

V_0 = vertical component of launch velocity.

r = distance of aircraft C.G. from ship's roll axis.

ω = roll rate (1.5 x S.H.M)

α = roll acceleration (1.5 x S.H.M)

From the foregoing the following tables can be constructed:

Roll Angle	ω (Radians/sec)	α (Radians/sec^2)
5°	0	0.0525
4°	0.0495	0.042
3°	0.072	0.0315
2°	0.075	0.021
1°	0.081	0.0105
0°	0.0825	0

If a Ski-Jump launch is used the maximum launch velocity achieved is 83 ft/sec (from Fig 23). The vertical component of this is 83 Sin 30° = 41.5 ft/sec = V_o and r = 41 ft. The coriolis component acceleration then varies as follows:

Roll angle	C (g units)
5°	0.068
4°	0.18
3°	0.226
2°	0.22
1°	0.222
0°	0.21

For the Ballista the maximum launch velocity is

$$64 \text{ ft/sec and } V_0 = 64 \text{ Sin } 40^\circ = 41.14 \text{ ft/sec.}$$

Hence:

$$r = 41 \text{ ft as before.}$$

Roll Angle	C (g units)
5°	0.0668
4°	0.179
3°	0.224
2°	0.218
1°	0.22
0°	0.21

In the comparable case of the inclined catapult, launch velocity is 78 ft/sec and $V_0 = 78 \text{ Sin } 30^\circ = 39$ ft/sec and $r = 49$ feet.

Hence:

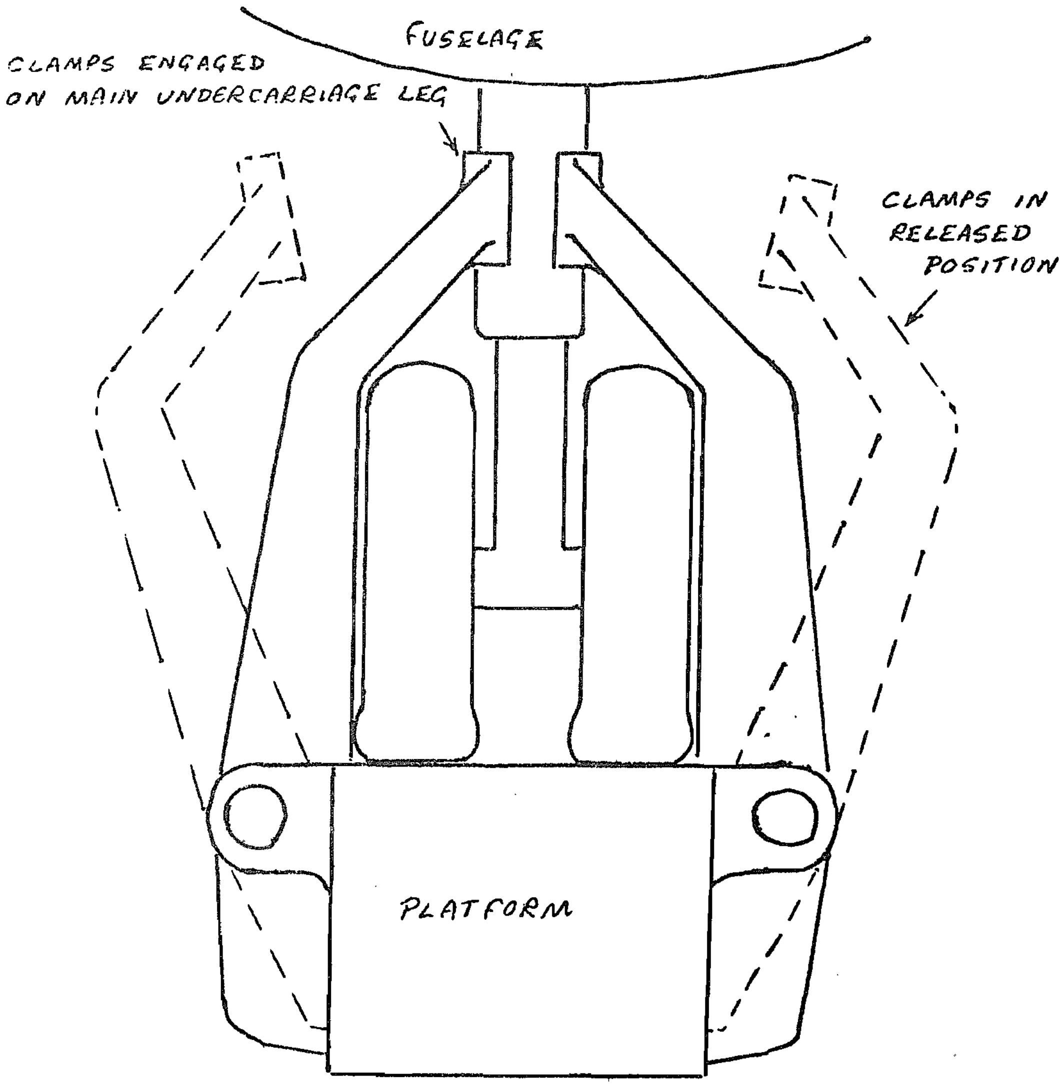
Roll Angle	C (g units)
5°	0.08
4°	0.184
3°	0.222
2°	0.214
1°	0.212
0°	0.2

In all cases the maximum lateral force is at about the same value of 0.22 g. If the aircraft weight is 24000 Lbs then the force is 0.22 x 24000 = 5280 Lbs.

A force of this magnitude causes no real problems to the structures of the various launching devices nor does it throw an excessive load on the aircraft undercarriage.
It does mean, though, that the aircraft will have to be restrained from skidding sideways or yawing during the launching process. In the case of the Ski-Jump the aircraft main and nose wheels must therefore run in channels in the deck. The ramp itself could consist only of channels for the main and nose wheels plus narrow strips for the outrigger legs.
The outrigger legs will restrain the aircraft in the rolling plane except in the case of the Ballista. The outrigger legs could be used in the Ballista but this would necessitate a T-shaped platform - a perfectly feasible solution. An alternative would be the use of a clamping device as shown in Fig 38 to be mechanically released at the end of the launching stroke. This would give a more compact structure. Either solution imposes a torsion load on the platform but it is a relatively small load and no real problem.
It seems evident that rolling motion, within the given limits presents no insuperable difficulties during the launching process.

FIG 38

BALLISTA - LATERAL AND ROLLING RESTRAINT FOR HARRIER



4.2.2 Effects of Pitching

Pitching motion affects the launching operation by imposing vertical velocities on the aircraft and launching machinery. These velocities will be added to or subtracted from the launch velocity of the aircraft.

If pitching is assumed to be Simple Harmonic Motion then maximum pitching velocity occurs when $x = 0$. At this point,

$$V = 2\pi f\sqrt{A^2 - x^2}$$

$$= 2\pi \times \frac{1}{5.5}\sqrt{(167 \text{ Sin } 2^{\circ})^2 - 0}$$

$$= 6.66 \text{ ft/sec.}$$

If this velocity were subtracted from the launch velocity the aircraft would fall 90 to 130 feet below launch height, depending on launch weight, and assuming a launch height of 40 to 50 feet the flight would terminate disastrously after a few seconds in a large splash.

Fortunately the situation is not as severe as this. The above calculation assumed that the aircraft was launched from near the stern of the ship. This need not necessarily be so since all the launching devices considered may be situated closer to the pitching axis of the ship with consequent reduction in pitching velocity. In addition, although in fact ship pitching is not an absolutely regular motion it is readily predictable by the launching officer. In the days when piston engined aircraft made free rolling take-offs from aircraft carriers it was regular practise to, so time the start of the take-off run that the aircraft left the flight deck as it pitched up, gaining vertical velocity thereby. This can be done again so that the vertical motion is used to assist the launch.

4.2.3 Effects of Yaw

Yawing velocities and accelerations are small and are unlikely to give rise to significant loads during the launching process. They may result in the aircraft being launched with a small amount of sideslip. This should not present any difficulties unless there is already a cross-wind close to the aircraft limitations.
(but see 1.14.7)

4.2.4 Effects of Heave

Heave is the bodily motion of the whole ship in a vertical direction. It is not easily predictable and must therefore be allowed for by increasing launch velocity or decreasing the aircraft weight. Velocities due to heave are generally less than pitching velocities developed at the ends of the ship so that adjustments to launch velocity or weight will be small.

4.3 Frigate Installation

4.3.1 Fig 39 gives an indication of the space problem and shows very clearly that a Frigate is the lower limit of ship size for operating a small VTOL aircraft. The illustration is not intended to represent any particular class of ship but the hull dimensions are those of a Leander class Frigate.

The hangar space shown is sufficient to accommodate the Harrier plus workshop facilities. The launching device is a Catapult Assisted Ski-Jump angled 10^{o} to port to clear the superstructure. If a Ballista were used it would require 25 feet less of the ship's length - a worthwhile saving in a ship of this size as is shown in Fig 40. Again the Ballista is angled to port so that the initial flight path of the aircraft is clear of the ship's superstructure.

4.3.2 It is immediately apparent that in the case of the Ballista installation the hangar and flight deck are of approximately equal area and that the space required could be halved if the two were to be combined by providing a folding or retractable shelter over the landing area. Technically this is quite feasible but there are powerful practical arguments against it. One is the increase in complexity, another and perhaps more important argument is that the hangar is also a workshop and the conversion of a workshop with its benches, tools and test equipment into a clear open space is a far from instantaneous process. Nevertheless these objections are not insuperable and the idea is most attractive if it makes possible the carriage of another aeroplane or helicopter.

4.3.3 Due to ship motion and the confined space the manouvering of aircraft on the flight deck and in the hangar cannot normally be accomplished by taxying or manhandling. The use of tractors is ruled out by space limitations. Aircraft will therefore have to be moved by means of a rail system in the deck and winched from place to place. Space limitations have a favourable effect here in that the possible locations of the aircraft are very limited and a very simple rail system will therefore suffice.

4.3.4 Figs 39 and 40 indicate that a Harrier can be accommodated in and operated from a Frigate. The aircraft would be a potent addition to the ships fighting capability. A minimum of 9 men would be required to fly and maintain the aircraft and the provision of all the space required would not be easy. A compromise involving a degradation of some other aspect probably habitability - would almost certainly be necessary. Clearly the very maximum that could be squeezed into a Frigate hull as we know it is two Harriers or one Harrier plus one helicopter. The exercise will be well worthwhile but if we want to do better we must turn to an entirely different concept.

4.3.5 We require a small, cheap, expendable ship to fight long range recconnaisance aircraft and submarines. It must itself have a medium range strike and recconnaisance capability. As well as passive detection equipment, Airborne Early Warning capability is essential but since the ship is expendable, the inability to provide AEW cover on a round-the-clock basis is acceptable. The Harrier is unlikely ever to have an AEW capability and this would almost certainly have to be provided by helicopters which may or, more likely, may not have a concurrent Anti-submarine capability. The number of aircraft required is debatable, the only certainty being that the ideal will not be attained. As a starting point, four aircraft would provide a formidable armoury for a small ship and could comprise, say, two Harriers and two helicopters or three Harriers and one helicopter. It is impossible to put these in a hull of 3000 tons or less displacement. There is insufficient deck area. The provision of more deck area means building vertically, with consequent unwelcome increase in top-weight and the necessity for lifts - a very undesirable complication. The deck area must be provided on one level and this leads naturally to the consideration of a twin hulled ship.

4.4 The Catamaran Frigate

4.4.1 It should be said at once that the catamaran layout is justified only by its ability to provide a large deck area on hulls of small displacement. In most other respects it is less satisfactory than a single hull. In general it will require more power for a given speed, although interaction between the hulls may give favourable effects over a narrow speed band. Its structure is likely to be heavier due to the strains on the bridge deck connecting the two hulls. Its motion, in the rolling plane, will be more severe, although probably of smaller amplitude than the single hull. There are incidental benefits in Damage Control and manoeuvreability and it seems probable that a towed Sonar could be lowered from the bridge deck which would be more effective than the hull mounted Sonar used in a single hull. However, its advantage is the dramatic increase in deck area available as is illustrated in Fig 41.

4.4.2 This represents a Catamaran Frigate of about 2000 tons displacement, having a length of 300 feet and an overall beam of 120 feet. Each hull has a length/beam ratio of 15:1. It is fitted with a Catapult Assisted Ski-Jump centrally located so that the aircraft takes-off between the twin masts and funnels and over the forward superstructure. The Ski-Jump could have been located to one side but in this position the rolling motion of the wide platform provided by the twin hulls would impose significant vertical velocities on the aircraft during launching. In addition the central position effectively divides the hangar space into two, providing two hangar entrances and easing the problem of aircraft shuffling mentioned earlier. The hangar and workshop space is ample for the four aircraft.

4.4.3 The landing area between the hulls, aft, consists of a grating which allows the Harrier's jet efflux to pass into the space between the hulls thus eliminating ground effect with consequent benefit during landing and vertical take-off. It also makes it possible for personnel to be on the flight deck during landing as there will be little or no horizontal jet blast. A further consequence is that weapons such as anti-submarine mortars could be mounted in the hulls aft with minimal interference from aircraft operations.

The deck grating forward could be used only in calm weather or in harbour but it provides a useful additional take-off and landing area.

4.4.4 The additional deck area provided by the catamaran configuration also eases the manpower accommodation problem. Indeed it would appear that with the catamaran the minimum size is no longer dictated by the space required for aircraft and men but by sea-keeping requirements.

4.4.5 The catamaran is an ancient craft and catamaran ships are not a new idea. The concept of a catamaran warship of Frigate size is, however, sufficiently unusual to be not

Fig 39

HARRIER IN FRIGATE

SCALE 1" = 50'

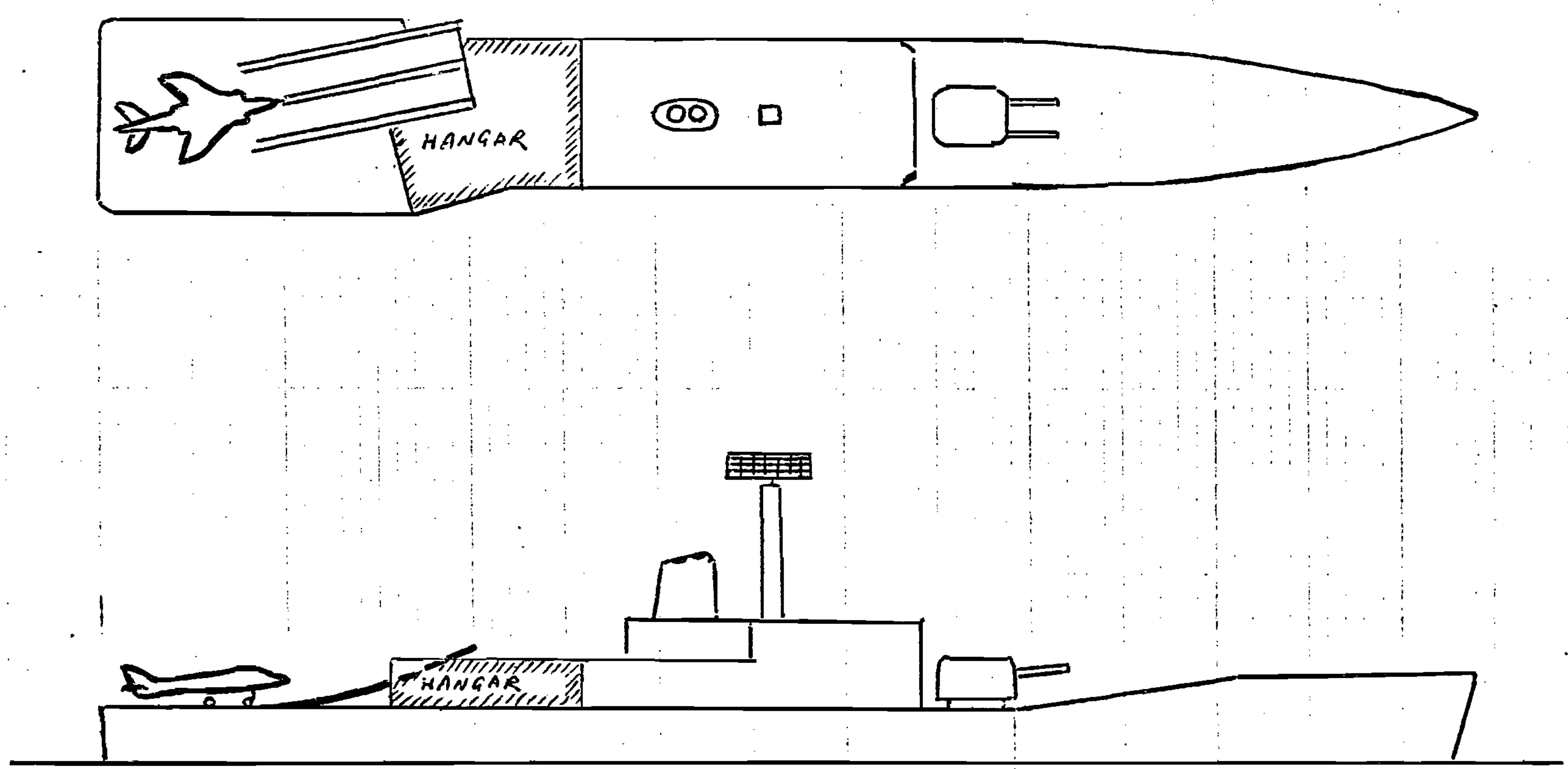


FIG 40

HARRIER IN FRIGATE

(LAUNCHING BY BALLISTA)

SCALE 1" = 50'

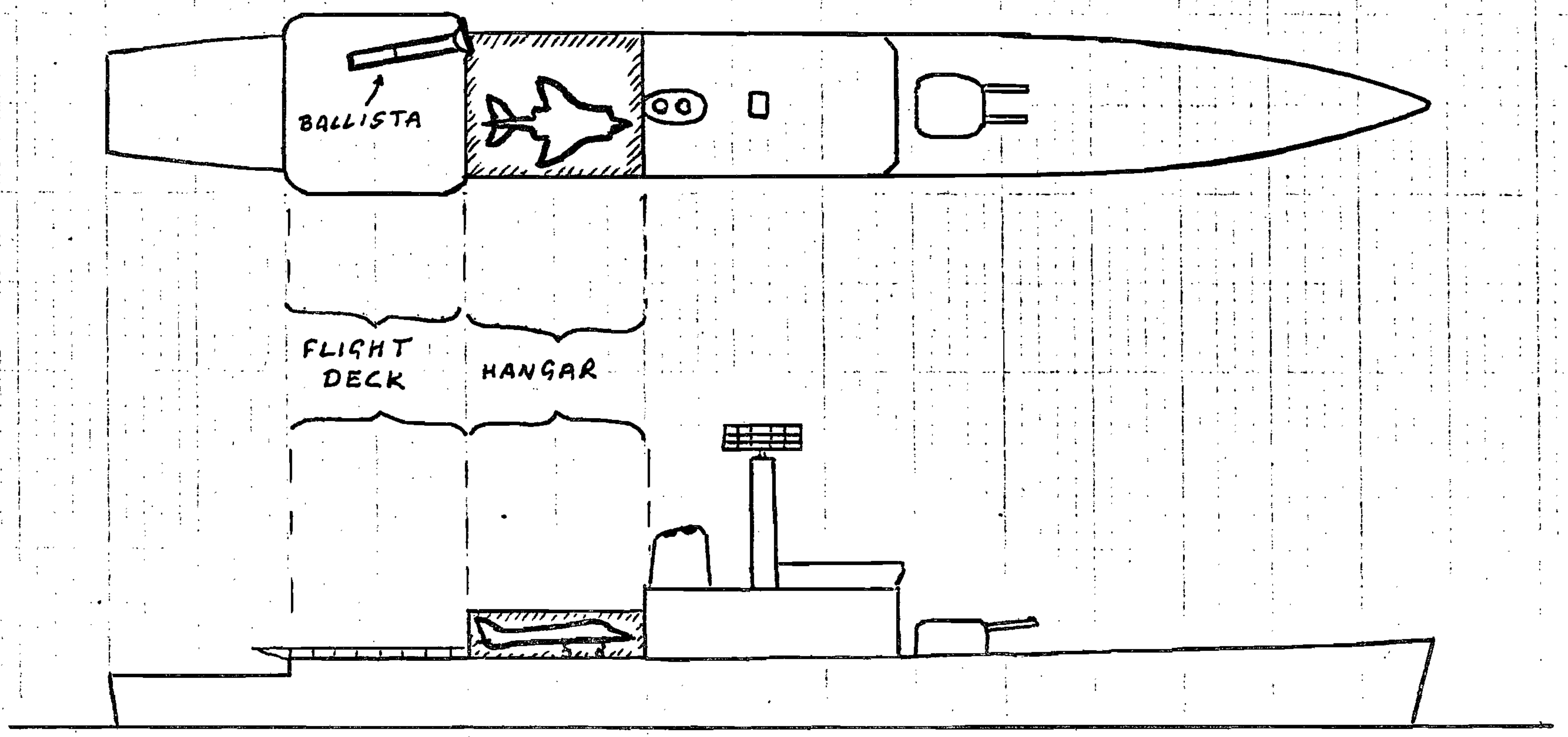
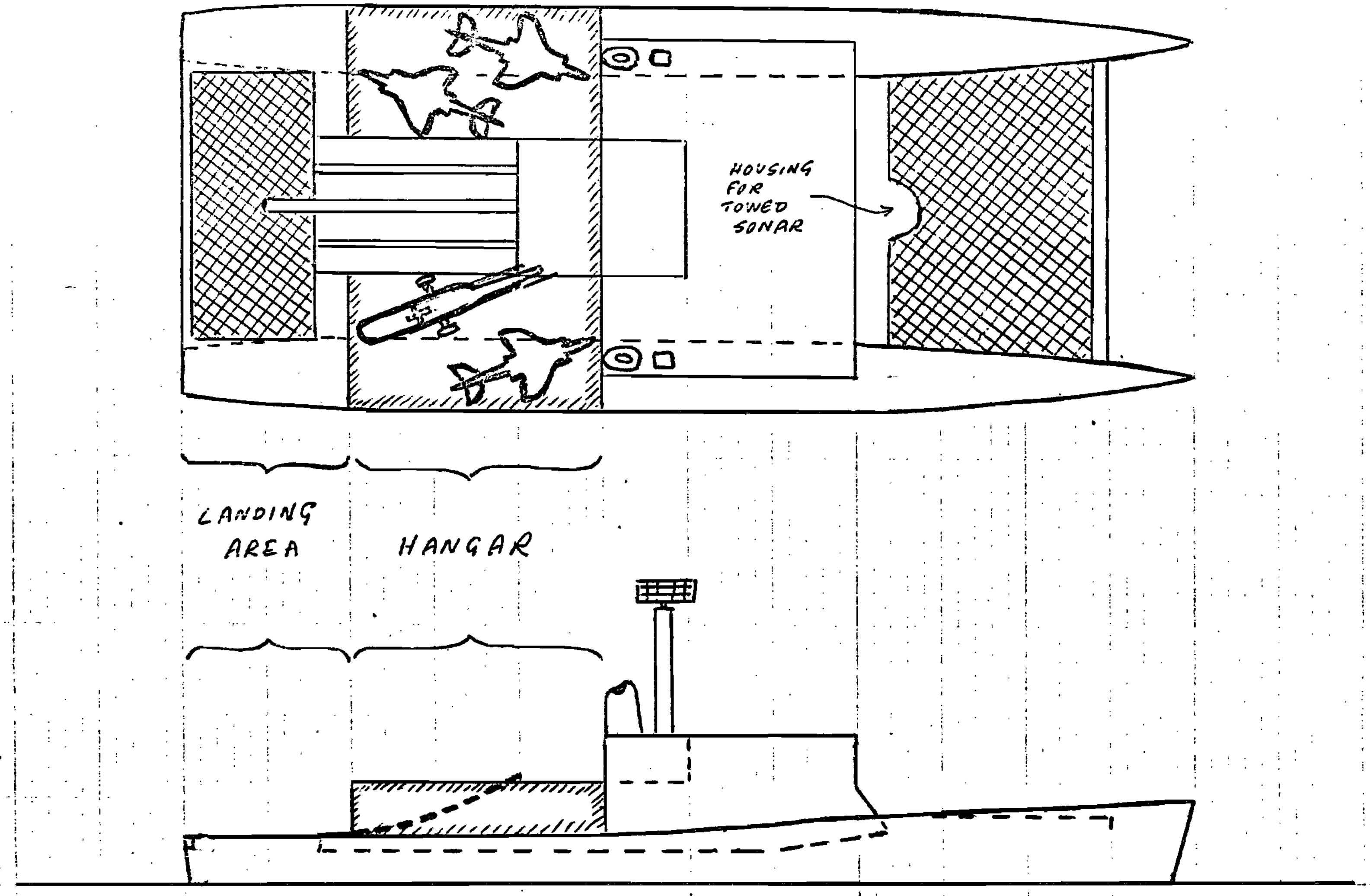


FIG 41 THE CATAMARAN FRIGATE

SCALE 1" = 50'



readily acceptable. Its deficiencies cannot be wished away nor on the other hand should they be exaggerated or used to excuse inaction. Like the VTOL aircraft, it has unique qualities which make it peculiarly suitable for specialised tasks such as this. It certainly merits further, more detailed study.

4.4.6 The objective of this study has been to find a method of launching Harriers at effective fighting weights from ships of Frigate size. If this has been successful then it follows that larger ships can operate more aircraft more efficiently by the use of the suggested launching devices than would otherwise be possible. We will therefore move up the scale and consider a projected ship, the Through Deck Cruiser.

4.5 Cruiser Application

4.5.1 The Through Deck Cruiser (hereafter designated TDC) is, at about 20,000 tons displacement, a respectable aircraft carrier and capable of accommodating an effective force of aircraft. Unfortunately this capability is severely curtailed by the space required for a Harrier to perform a rolling take-off. For this purpose an area of at least 450 feet by 40 feet - the greater part of the flight deck - must be left clear. While aircraft are taking off no other flying operations are possible.

Indeed very little other activity of any kind is possible. Aircraft cannot be repositioned on deck or taken to the hangar and refuelling and re-arming will only be possible in very limited areas. The use of a Ski-Jump would not only considerably increase the possible take-off weight of the aircraft but would transform the situation on the flight deck as is shown in Fig 42.

4.5.2 The ship is fitted with two Ski-Jump ramps angled about 10° outboard. There is ample space for unassisted take-off at maximum weight so no catapult assistance is necessary. The ramps occupy very little of the existing deck space and, being simple structures, can easily be fitted retrospectively. Assuming pre-fabrication, a month in a dockyard would be more than sufficient for the task. The through-deck capability is not compromised so that rolling take-offs could still be performed if required. However, the use of the Ski-Jumps allows Harriers to take-off while Helicopter operations, Harrier landings, refuelling and re-arming occur simultaneously while the command has a fairly wide range of acceptable ship's headings to choose from. Battle damage to the flight deck would have to be very extensive indeed to prevent aircraft operations. On paper at least, a very much more effective TDC is feasible for almost negligible cost.

Fig 42

THROUGH-DECK CRUISER

SCALE 1 CM = 25 FEET

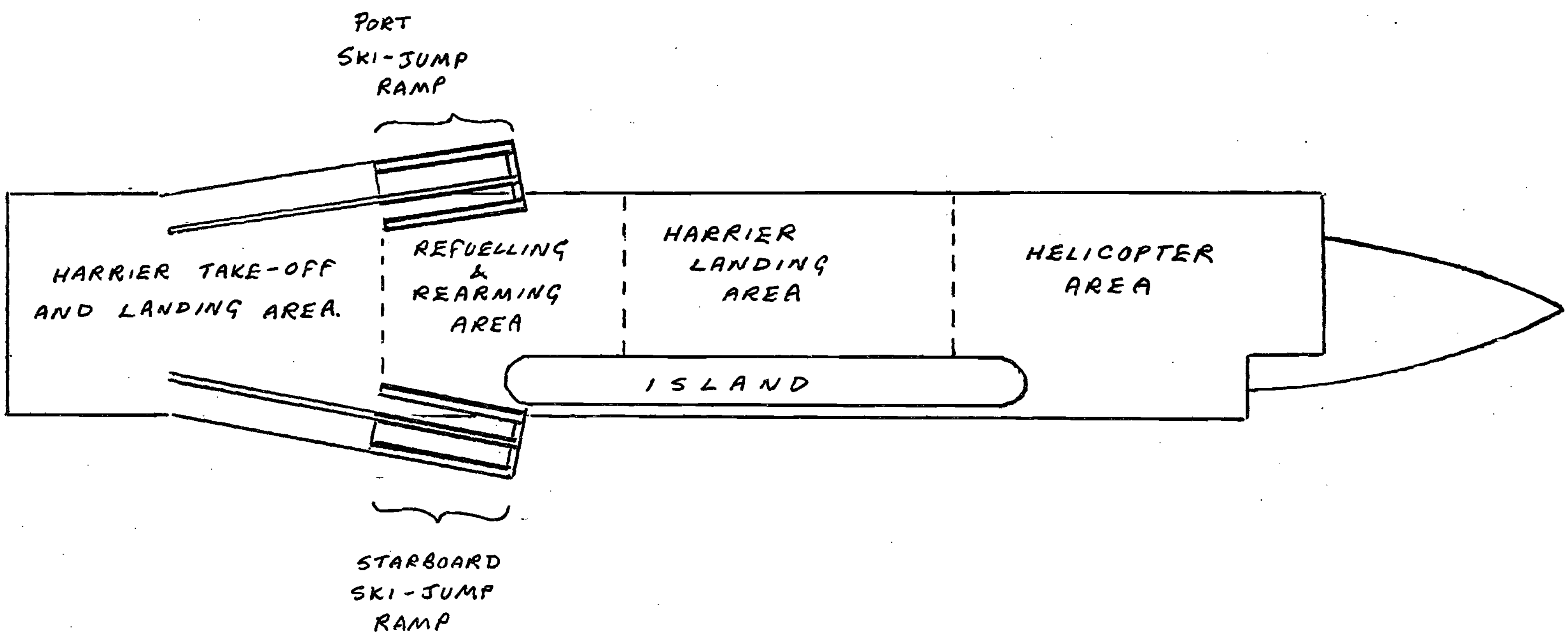
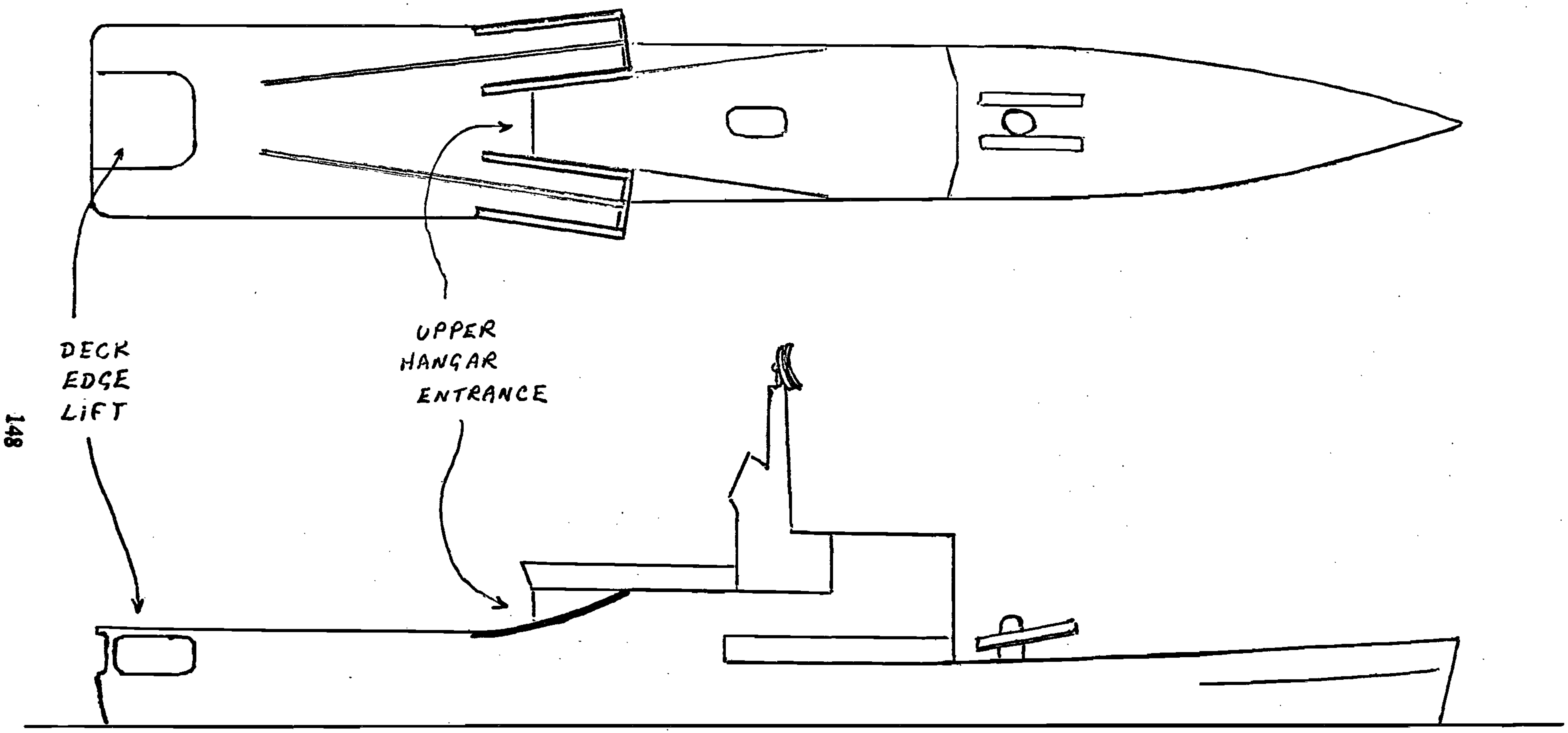


FIG 43

CRUISER

SCALE 1 CM = 25 FEET



4.5.3 Coming down in scale from the TDC, the use of the Ski-Jump makes it possible to design an effective aircraft carrying ship of more traditional cruiser shape and dimensions if the layout originated in the Russian ships Moscow and Leningrad is adopted. Fig 43 illustrates one possibility. The sketch shows a ship of about 10,000 tons capable of carrying a mixed bag of say, 8 aircraft. Two Ski-Jumps are fitted since they encroach very little on deck space and, being an integral part of the superstructure their contribution to topweight is small. Hangar accommodation is on two levels with a deck edge lift giving access to the lower hangar. More than half the ship is available for the more traditional cruiser functions. Without the launching devices any Harriers embarked would be limited to sorties of brief duration and on such a small deck this would greatly interfere with any concurrent helicopter operations.

4.6 Application in Merchant Ships

4.6.1 The problem is rather different in character to the case of a warship. Launching devices will only be fitted to these ships on outbreak of war or under the threat of imminent war. Then they will have to be fitted quickly and in large numbers. The Ski-Jump is the only device cheap and simple enough to meet the requirement.

4.6.2 Space is generally less of a problem than in a warship though difficulty will arise in those ships whose cargoes have to be loaded and unloaded by cranes through deck hatches. In this case removeable sections will have to be provided in the flight deck and Ski-Jump. For tankers and bulk carriers it would be relatively easy to provide temporary flight decks and one or more Ski-Jumps. The larger ships have the space to accommodate a squadron or more Harriers together with AEW and anti-submarine helicopters. Hangarage for the aircraft would not be necessary except, in rudimentary form, for one or two aircraft undergoing repair. The aircraft and their crews would be embarked for quite short periods with heavy repair and maintenance in depth being undertaken ashore between voyages. Most aircraft therefore would stand in the open with the usual protective covers when not flying.

4.6.3 Crew accommodation in modern merchant vessels is, by naval standards, palatial. In wartime the doubling or even trebling in numbers of the crew should be possible without reducing standards of habitability to unacceptable levels in view of the relatively brief periods of time under consideration.

4.6.4 A more fundamental problem is that of providing power supplies and communication and direction facilities for the embarked aircraft. The problem is one that has already been met and overcome in the operation of Harriers from dispersed sites ashore. The solution is to provide these facilities in self contained portable units. For this application the standard container currently in use for cargo seems most appropriate and is large enough to accommodate any of the generators, prime movers etc likely to be required.

4.6.5 Finally, fuel and ordnance will be required for the Harriers and helicopters. In a tanker the fuel supply can fairly easily be arranged by setting aside a tank for aviation fuel although some minor systems modifications may be necessary to eliminate the risk of contamination from cargo or bunker fuel. In other ships portable tanks and pumps will be necessary and could be placed on deck or in a hold according to the dictates of space and top-weight. As far as ordnance is concerned much of it is robust and insensitive and could be stored in the open in wartime. Missiles however will require protection for their delicate guidance equipment which will probably necessitate a covered stowage. Fuzes, detonators and pyrotechnics must be stowed in magazine conditions but fortunately the weight and volume of these is small so that the use of containers offers a feasible solution.

4.6.6 The conversion of the larger merchant ships to operate Harriers and helicopters clearly presents no problems which have not been met and overcome in the past. The main difficulty is the sheer size of the task; the number of ships involved. Under the spur of necessity it could be accomplished quite swiftly, as was the fitting of magnetic and acoustic mine countermeasures and guns etc during the second World War. Even so, it would not be an overnight process by any means. Likewise, the postulated threat, blockade, could not arise overnight. There must be an excuse for such overt action, a natural or contrived increase in international tension is an essential prerequisite and the movement and positioning of naval forces to apply the blockade would not go unobserved. The warning period is likely to be weeks or months rather than days, the preparations to meet the threat would form the visible measure of a nation determined to defend itself and intrinsically, a deterrent factor.

4.6.7 The only items which must be prepared long in advance are the most complex and expensive ones. The aircraft and their support equipment must at all times be ready. The naval crews to fly and maintain them must be trained and practised in their profession. The conclusion which follows from this is fundamental and has far-reaching implications:

The size and composition of the Fleet Air Arm must be related not only to that of the Royal Navy as a whole but also to the size of the Merchant Fleet which must, at all costs, be defended successfully.

CHAPTER 5

The Case for Operating V/STOL Fixed-Wing Aircraft from Small Ships

5.1 The Background

The Royal Navy is a defensive force though this does not mean that it is never used in offensive operations. The United Kingdom has, in the past, been provoked into initiating armed conflict and the possibility that in the future it may again be necessary to continue diplomacy by other means cannot be ignored. Nevertheless the prime functions of the Royal Navy are essentially defensive. They are firstly to deter an attack upon the UK and its allies and, if deterrence fails, to ensure national survival. It should be noted that national survival does not require the total defeat of an enemy - which may be quite impossible - but merely to persuade the enemy to cease his attack and resume normal diplomatic relations.

Since the Industrial Revolution the UK has become increasingly vulnerable to blockade and in the last 30 years alarmingly so. The same is true, in lesser degree, of Western Europe which is, in effect, already blockaded from the East. Seen in these terms the British Islands are geographically and politically the key to Western Europe and its main bastion.

It seems scarcely credible at the present time that the USSR would risk an invasion of Europe and the near certainty of nuclear war; indeed such an invasion is not necessary if domination of Europe is the objective. It would be quite sufficient to bend the European states to the communist will, to make of them self-governing vassals unoccupied by foreign armies but obedient to Moscow. If this cannot be achieved by political or subversive means some external leverage is necessary. A successful blockade could provide the leverage by bringing economic ruin with resultant chaos and anarchy. It has too the enormous advantage that no fundamental damage is done to the economic potential which thus quickly becomes available to the new masters.

Such a blockade could be mounted using conventional weapons although the employment of tactical nuclear weapons at sea cannot be ruled out as the risk of escalation is much less than if these weapons were used on land.

There can be no doubt that blockade figures largely in the Russian appreciation of the situation. The enormous increase in Russian naval strength and especially their massive investment in submarines - the blockade weapon par excellence - is proof enough. The lessons of two world wars and of the Cuban crisis have been well learned.

The means of blockade are available and the next question must be the probability of their use. Through all the imponderables it can be argued that the probability is small at present but of all the possibilities of major armed conflict blockade must rank as one of the most likely threats.

The probability remains small while American intervention is certain but there are signs that this certainty may not continue. The long term effects of the collapse of American morale following the indecisive war in South East Asia and the Presidential scandals are not easily predictable but it seems increasingly unlikely that the USA will be stirred to action by anything less than nuclear confrontation.

If the premise that blockade is increasingly a threat to the UK and to Europe is accepted, then it follows that our defences must be tailored accordingly. Blockade may also be employed against British interests elsewhere, the Icelandic "cod war" contains some of the elements of blockade and has already resulted in the loss of an important air base. The case for operating V/STOL fixed wing aircraft from small ships rests, to no small degree, on the implications of this background.

5.2 Some Tactical and Strategic Considerations

5.2.1 The traditional anti-blockade strategy is, of course, the convoy. The concept has been a foundation of naval doctrine for over a century though it was not at first applied during the first World War for reasons which need not be examined here. The convoy system was re-imposed by a reluctant Admiralty in 1917 when shipping losses due to submarines had created a desperate situation. It was an immediate success. The reasoning behind the original Admiralty objections to the system were not necessarily entirely at fault. It seems probable though that the limitations of the submarines particularly in speed and firepower when faced with a concentration of ships powerfully escorted, were not fully appreciated.

In the second World War convoys were used from the start but they faced new threats. The first of these, attack from the air, caused severe casualties especially when, as was often the case, the convoy's air defence was inadequate or non-existent. It was the second threat, however, which brought the convoys closest to defeat for in the open oceans the submarines had developed new and effective tactics. A sighting was the signal for the submarines to gather for a "wolf pack" attack. The problem of speed and firepower was solved by attacking on the surface, at night, using both guns and torpedoes. The situation was saved by long range aircraft using radar and carrier-borne aircraft similarly equipped which forced the submarines to remain submerged.

If a convoy encountered superior forces the tactics were usually to scatter the merchant ships while the escort did its best to occupy the enemy. On the few occasions when this tactic was used the results were disastrous because the scattering was too late.

The passage of nearly 30 years in which no major naval actions have occurred has left a situation surprisingly similar in some respects. Technical advances have however swung the balance strongly in favour of the submarine. The likely enemy has submarines in larger numbers than have been faced before but with a speed and endurance equalling or surpassing that of the surface ships and capable of diving to unprecedented depths. He also has surface ships equipped with surface to surface missiles with ranges in excess of 100 miles. The fears of the Admiralty, pre-1917 have been realised; the enemy has numbers, speed and fire-power - and near perfect concealment.

His target, too, has changed in his favour. In 1944 a 20,000 ton tanker was a big ship and a rarity, today 100,000 tons and more is commonplace. Other merchant ships frequently exceed 30-40,000 tons yet their vulnerability to the torpedo or missile is not significantly less. The effect of their loss, however, remains proportional to their tonnage or even more. A convoy of such ships is a vital target justifying the maximum efforts to ensure its destruction and conceivably tempting the enemy to resort to the use of tactical nuclear weapons to achieve a speedy decision.

The assembly of such a convoy and its escorts could not be concealed, its detection and tracking at sea either by satellite or long range aircraft is not difficult. Its characteristics make positive identification unnecessary. Forces sufficient to overwhelm and destroy it are available and have time to concentrate.

5.2.2. The greatest asset of a nuclear submarine is concealment, its greatest weakness its relative blindness. We must emploit the latter to the full with the object of inducing the submarine to reveal itself in some way, for once discovered it is one of the most vulnerable of all machines of war.

One conclusion seems inescapable, the convoy must not merely scatter, it must never assemble; for the present, at least, the convoy system must be abandoned. Ships must sail separately and by random routes to compound the enemy's search problem and to force him, in limited war, to make a positive identification before attack.

The submarine has a very limited search capability, to find and identify the scattered targets is a formidable task even with the numbers which the Russians can deploy. The reconnaissance and identification must be done by satellite and long range aircraft which must pass their information by radio. To receive the signals the submarines must come close to the surface thus greatly increasing the risk of detection. The weakest link is the long range aircraft which is very vulnerable to attack by fighter aircraft or ship to air missiles and reveals its location by its radio transmissions. The destruction of these aircraft is therefore a primary task which makes the use of ship borne aircraft essential in both fighter and airborne early warning roles.

The widely scattered merchant ships cannot possibly be individually escorted yet to make them sail defenceless in the face of the enemy is intolerable and foolish. They must have better eyes than the enemy and a measure of air and anti-submarine defence.

It is here that the deficiencies of the large aircraft carrier (including the so-called through deck cruiser) become manifest. Whatever its merits in terms of cost effectiveness the methods of accountancy cannot put it in more than one place at any given time; it could not conceivably protect more than a small fraction of the scattered merchant ships. Furthermore, it possesses inherently the worst features of the convoy, it is a key, a command centre, a concentration of air power and therefore a vital target. Like the convoy it is difficult to conceal and, when operating, its characteristics make identification unnecessary. Its loss would be disastrous and for this reason much of its immense fire-power must be dissipated in its own defence. Its use is therefore only justifiable, in this context, in defence of a convoy.

Although the lessons of the Bismark have largely been forgotten or ignored an aircraft carrier would not be an easy ship to sink. The American space exploits have shown that even the most complex and delicate equipment can be repaired by makeshift methods but even so it is most likely that a carrier subjected to massive attack would quickly be rendered ineffective as a fighting unit. If support for this view is needed, the accidents, in recent years in the USS Forrestal and Enterprise surely provide it as does the earlier accident in HMS EAGLE during the Suez crisis.

If the merchant vessels are to be unescorted they must contain their own defence rather like the MAC ships of World War Two though they were a last chance weapon, to be used once in desperation. The fixed wing V/STOL aircraft and helicopters can provide long range reconnaissance, early warning of attack, air defence and a measure of anti-submarine capacity on a continuing basis. The larger and more important the ship, the more defence it is able to carry.

The conclusion arising from this conception is worth repeating. It is that the size, composition and deployment of the Fleet Air Arm is a function not only of the size of the Royal Navy but also of the merchant fleet to which it gives protection.

5.2.3 We have so far considered the Navy only on a passive role and indeed with the convoy system this would be the role of the bulk of the surface fleet.

Freed from convoy escort by the V/STOL aircraft the surface ships are available for positive offensive operations. For the first time we are facing a greatly superior naval enemy and our offensive strategy must be modified accordingly. No clash of opposing fleets can be contemplated, a concentration of the enemy must be regarded as a target, not to be defeated, but to be damaged as much as possible at minimum cost to our own forces.

What is envisaged is a guerilla war at sea. The analogy is not complete; classic guerilla warfare theory requires operations to be conducted from a background of a friendly populace prepared to offer concealment and at least passive support. The sea, like the jungle and the desert is neutral, its support goes to friend and foe alike and it offers no concealment except to the submarine or from the weather. Success therefore can come only from superior vision and success must be measured not in the enemy's defeat but in the thwarting of his purpose sufficiently to make him abandon his efforts.

Our ships must therefore be designed to see first, to strike first and avoid the enemy's retaliation. In the nature of things the latter will not always be possible so our ships must be able to suffer damage and, in the last resort, to be expendable. It is here that we face a classic dilemma. The only ships we can afford in quantity are frigates and even they grow more expensive every year. To see first and strike first we must have aircraft on the spot - not at some distant shore base - yet we have already decided that a large aircraft carrier does not suit our purpose. The dilemma can be resolved by the use of V/STOL aircraft if their performance can be made comparable to conventional aircraft and if we can operate them from ships of frigate size. To be expendable the ship must be cheap, in relative terms, which in turn means that its weapons and equipment must reflect the state of the art rather than the vanguard of technology. Though giving less than the ultimate in performance a valuable effect of this philosophy is likely to be an increase in reliability and ability to survive battle damage. Reliability is too often overlooked or paid lip service in pursuit of the best. Reliability breeds confidence which enhances morale. Confidence in the most exotic equipment evaporates swiftly if it is delicate and uncertain in action.

5.2.4 Having briefly dealt with the main threat; the more usual occupations of the Navy against lesser foes must be examined. A lesser power may attempt to restrict or deny our use of straits, channels or sea areas adjacent to his coasts. This is, in effect, a limited blockade. If the enemy's naval and air strength is limited a case for the convoy and the large aircraft carrier can be made out. However, many of these countries have been equipped with fast patrol boats effectively armed with surface to surface missiles. These pose a real threat against which there is no complete defence. Even the smallest powers therefore have the ability to put a large carrier out of action with disproportionate effects on national prestige. The dispersal of our aircraft in small ships renders such national humiliation much less likely and increases the chances of effective retaliation. The ships would have more freedom of action in shallow waters and be more manoeuverable in narrow channels. They would not require to steam for long periods at high speeds on undesirable courses in order to launch and recover their aircraft, nor would there be delay in operations and possible loss of aircraft due to a fouled deck. Our seaborne airpower could be concentrated or dispersed at will and as the situation dictated. Above all, it could not all be lost as the result of one lucky hit.

It is in actions in support of ground forces that the large aircraft carrier really comes into its own. In this type of operation which has been a main activity of aircraft carriers since the war they are usually unopposed at sea and are able to concentrate on offensive operations. In this they have been consistently highly successful and the cost effective arguments in favour of large aircraft carriers have full force. There can be little doubt that this success and lack of counter attack has bred a degree of complacency and a lack of awareness of the vulnerability which has crept into these ships.

It might appear then that in this sphere of operations which has been the normal work of aircraft carriers in the past 20 years and could well continue to be for the next 20 years, the large aircraft carrier reigns supreme. The arguments for the small ship alternative lack the compelling force of actual experience. Yet we cannot base our defence against the main threat, blockade, on a weapons system developed against the artificial background of no opposition when we know that in the ultimate test it will be opposed, will suffer damage and is likely to lose most or all of its fighting value as a result of that damage.

However the ultimate argument is financial. The UK is unwilling, if not unable, to afford more than about three large aircraft carriers or five of the so-called through deck cruiser type. Each of these represents a large proportion of our defence effort with a value far outstripping its cost. As a result we cannot afford to lose one, we become reluctant even to risk one and consequently become timid and hesitant in the use of them. Even an accident, a grounding on entering harbour for instance assumes grotesque significance. A weapons system, having these characteristics is not an asset, it is a liability.

The evolution of weapons closely parallels the evolutionary process in nature but with the timescale compressed by many orders of magnitude. In less than a century we have seen the genesis, development and passing into extinction of the big battleship. The process is well advanced in the case of the aircraft carrier and of the large manned bomber. There are signs that the submarine itself is at the peak of its career. In every case the extinction has been hastened by the advent of a weapon not necessarily more sophisticated but better adapted to fight and survive. As in nature these new weapons have two principal common characteristics; small size and a much smaller demand on resources.

The large aircraft carrier, like the battleship before it, has become a dinosaur. It is time for it to go.

It's passing cannot be without pain, particularly for the Royal Navy. It is an imposing vehicle from which to show the flag, an instantly recognised symbol of power. The command of such a ship may be the best event of a man's career and to command a fleet of them, the summit of his ambition. A solitary frigate is not an impressive vehicle for an Admiral's flag.

Nor can sentiment be dismissed, it is subtle but essential ingredient of morale without which all is useless. Sentiment also has its negative aspects; when it flies in the face of reality it becomes stultifying tradition. This must not happen, the change to a small ship navy can be so contrived that it is accompanied by the cheerful confidence which stems from faith in the weapon and acknowledged competence in its use.

5.2.5 The case for operating V/STOL fixed wing aircraft from small ships may be summarised in the following terms:

(a) It provides the basis of a new naval strategy which, by rendering the enemy's forces less effective and increasing his probable losses forms a credible deterrent.

(b) It allows seaborne air power to be deployed widely in numbers of inexpensive ships or concentrated as required.

(c) It increases tactical flexibility.

(d) It greatly increases the enemy's search and identification problem.

(e) Makes more warships available for offensive operations.

(f) Makes possible an effective self-defence for merchant ships.

5.2.6 This then, is the background to the foregoing work and its justification. The V/STOL aeroplane as it exists today can operate from small ships but only at the cost of an unacceptable degradation in range and payload. The objective of this study has been to evolve a means of operating V/STOL aircraft from confined spaces whilst retaining a performance capability comparable with that of similar conventional aircraft.

Appendix 1

Manual Integration Method for Determination of Trajectory

This method was used in conjunction with equations (1) and (2). The result required was the vertical distance from the launch point symbolised by $\mathfrak{z}$.

Given the initial conditions and constants the variables were calculated at time intervals of 1 second.

Example:

$W = 22{,}000$ Lb, $\alpha = 30^{\circ}$, $\theta j = 55^{\circ}$, $V_L = 83.33$ ft/sec,

$U_o = 72.2$ ft/sec $V_o = 41.65$ ft/sec, $K_L = 0.1343$,

$K_D = 0.0269$, $T = 19{,}200$ Lbs

At $t = 0$ the above apply and:

$$F = T \text{ Cos } \theta_T = 19{,}200 \text{ Cos } (55^{\circ}+30^{\circ}) = 1{,}673 \text{ Lbs}$$

$$L_T = T \text{ Sin } \theta_T = 19{,}200 \text{ Sin } (55^{\circ}+30^{\circ}) = 19{,}126 \text{ Lbs}$$

$\dot{u}$ is given by equation (2)

$$= \frac{32.2}{22{,}000}\left[1673-\sqrt{72.2^2+41.65^2}\,(0.1343\text{x}41.65+0.0269\text{x}72.2)\right]$$

$$= 1.53 \text{ ft/sec}^2$$

$\dot{v}$ is given by equation (3)

$$= \frac{32.2}{22{,}000}\left[19126-22000+\sqrt{72.2^2+41.65^2}\,(0.1343\text{x}72.2-0.0269\text{x}41.65)\right]$$

$$= -3.02 \text{ ft/sec}^2$$

To find conditions at $t = 1$ sec,

$\dot{u}$ is added to the previous value of u giving a new value for u of $(72.2+1.53) = 73.73$ ft/sec.

Similarly $\dot{v}$ is added to the previous value of v giving a value of $(41.65-3.02) = 38.63$ ft/sec.

α Can now be calculated since:

$$\tan \alpha = \frac{v}{u} = \frac{38.63}{73.73} \text{ whence}$$

$$\alpha = 27.65^\circ \text{ at } t = 1 \text{ sec}$$

$$\theta_T = (55^\circ + 27.65^\circ) = 82.65^\circ$$

$$F = 19{,}200 \cos 82.65^\circ = 2{,}456 \text{ lbs}$$

$$L_T = 19{,}200 \sin 82.65^\circ = 19{,}042 \text{ lbs}$$

The process is then repeated in successive steps of 1 second. The value of v decreases and then becomes negative and the successive calculations are continued until v again becomes positive, indicating the lowest point of the trajectory.

z is found by plotting v against t and finding the area under the curve by the trapezium rule or similar methods to give a first approximation. Since the method assumes that each value of $\dot{v}$ and $\dot{u}$ remains constant for 1 second there is an accumulation of errors in v and u. In u they are not large but in v they are significant.

A second approximation may be made by averaging the difference between successive values of $\dot{v}$ in the same sense. For example in the table of results below; $\dot{v}$ is increasing between t = 0 and t = 8 seconds so differences are averaged over this period. Thereafter the change in $\dot{v}$ is in the opposite sense and differences are averaged over the period t = 8 to 20 seconds.

Since, change in distance = $\frac{1}{2}$ acceleration x time^2

the correction factor to be applied to the first approximation of z is:

$\frac{1}{2}$ Mean difference of $\dot{v} \times 1^2 \times$ elapsed time

From t = 1 to t = 8, $\frac{1}{2}$ x mean difference = -0.104

and from t = 8 to t = 20,

$\frac{1}{2}$ x mean difference = +0.504

So, at t = 5 secs, the correction factor to be applied to $\mathfrak{z}$ is -0.104 x 5 = -0.520 and $\mathfrak{z}$ = (168.5-0.520) = 167.98 feet.

At t = 14, the correction factor is 0.504 x 6 (since we are now counting from t = 8) = 3.024

$\therefore \mathfrak{z}$ = 220 + 3.024 = 223.024 feet

The following table shows the results for this example:

t	θ_T $(\theta_j+\alpha)$	F	L_T	α	u	v	$\dot{u}$	$\dot{v}$	$\mathfrak{z}$ (1st approx)	$\mathfrak{z}$ (2nd approx)
0	85.0°	1673	19126	30°	72.2	41.65	1.53	-3.02	0	0
1	82.65°	2456	19042	27.65°	73.73	38.63	2.72	-3.12	40.14	40.04
2	79.9°	3367	18902	24.9°	76.45	35.51	4.08	-3.27	77.2	76.99
3	76.8°	4384	18693	21.8°	80.53	32.24	5.59	-3.47	111.08	110.76
4	73.47°	5463	18406	18.47°	86.12	28.77	7.17	-3.72	141.6	141.18
5	70.03°	6557	18045	15.03°	93.29	25.05	8.76	-4.02	168.5	167.98
6	66.64°	7613	17626	11.64°	102.05	21.03	10.29	-4.31	191.5	190.87
7	63.46°	8578	17176	8.46°	112.34	16.72	11.67	-4.55	210.4	209.67
8	60.6°	9425	16727	5.6°	124.00	12.17	12.89	-4.68	224.8	223.96
9	58.13°	10137	16305	3.13°	137.00	7.49	13.9	-4.64	234.7	235.2
10	56.08°	10714	15932	1.08°	151.00	2.85	14.7	-4.39	239.8	240.8
11	54.47°	11157	15625	-0.53°	165.7	-1.54	15.19	-3.93	240.5	242.00
12	53.27°	11482	15388	-1.73°	180.8	-5.47	15.32	-3.24	237.0	239.00
13	52.46°	11698	15224	-2.54°	196.12	-8.71	15.27	-2.35	230.0	232.5
14	52°	11820	15130	-3°	211.4	-11.06	15.08	-1.26	220.0	223.02
15	51.9°	11847	15109	-3.1°	226.5	-12.32	14.77	+0.013	208.3	211.8
16	52.1°	11794	15150	-2.9°	241.3	-12.31	14.38	+1.43	196.0	200.0
17	52.57°	11669	15247	-2.43°	255.7	-10.88	13.96	+2.98	184.6	189.1
18	53.32°	11469	15398	-1.68°	269.7	-7.9	13.5	+4.64	175.2	180.2
19	54.34°	11193	15599	-0.66°	283.2	-3.26	13.04	+6.4	169.6	175.1
20	55.6°			+0.6°	296.2	+3.14			169.5	175.5

Results obtained by this method are an approximation at best but the degree of accuracy is considered to be acceptable in the light of results obtained by other methods.

The method may, of course, be used to find trajectories with constant ΘT in which case F and L_T also become constants.

Appendix 2

Determination of Trajectory - Simplified Method

In most cases the horizontal distance covered by the aircraft during semi-ballistic flight is of no interest. The result required is the change in vertical distance from the launch point with time. For present purposes it is assumed that point P is required to be at launch height, ie $z = 0$. At this point $\frac{dz}{dt} = v$ will also be zero and of course z must be positive throughout the trajectory up to this point.

Given these conditions, if the approximate required values of Uo and Vo are unknown their determination is a tedious but straightforward process probably best done graphically as follows:

(i) Estimate values for Uo and Vo.

(ii) Using equation 3 calculate successive values of v with t and plot them. The result is a curve of the form shown in Fig A2.1. Since z must be zero at point P the areas above and below the line Vo = 0 must be equal.

(iii) Adjust the line until this is so. (Shown by shaded areas and dotted line in Fig A2.1).

(iv) The new zero line determines the correct value of Vo for the selected value of Uo.

(v) For final adjustment apply the new value of Vo to equation 4.

(If approximate values of Uo and Vo are known, steps (i)

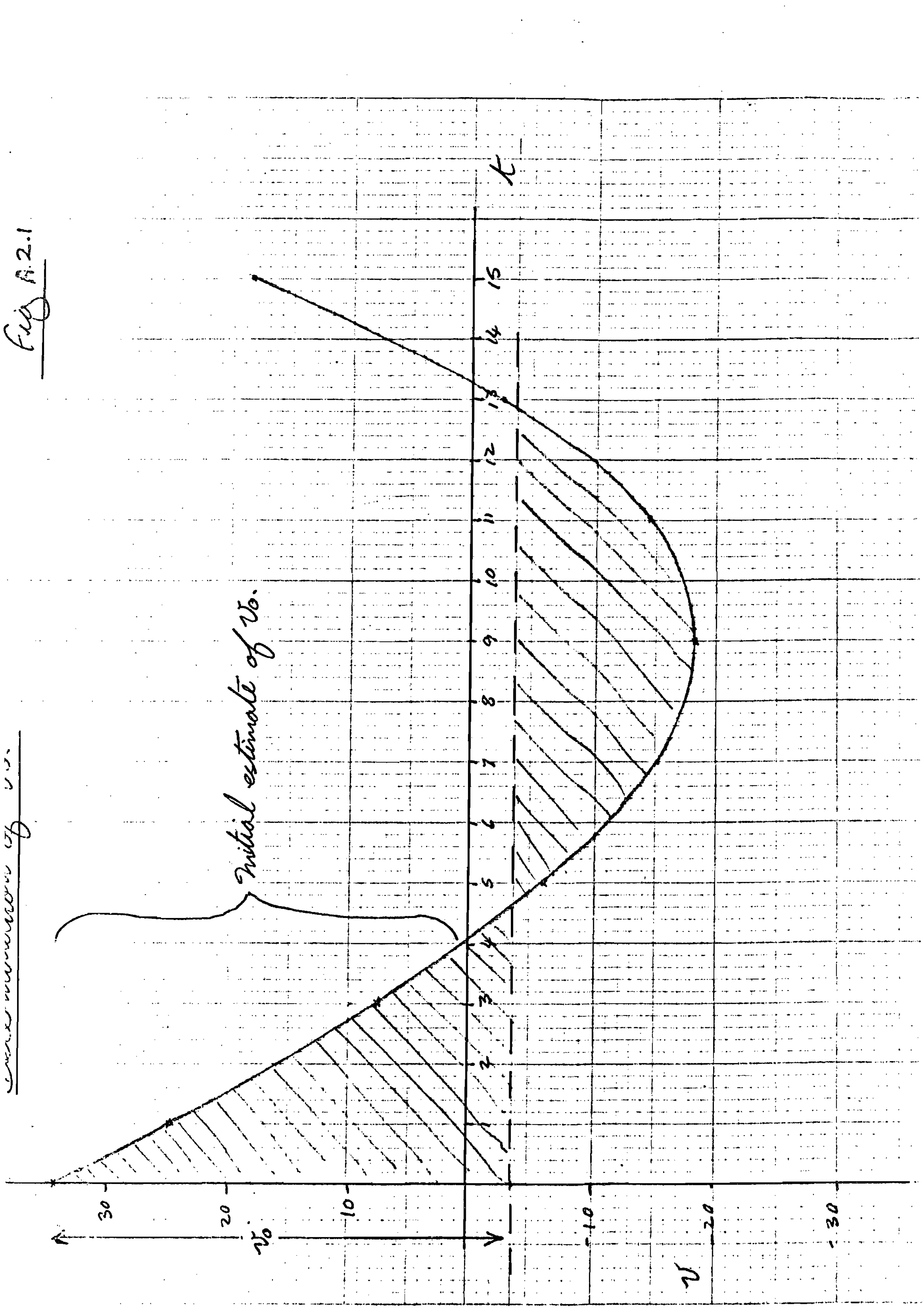


Fig A.2.1
initial estimate of v_0.
v_0
x
v
30
20
10
-10
-20
-30
1
2
3
4
5
6
7
8
9
10
11
12
13
14
15

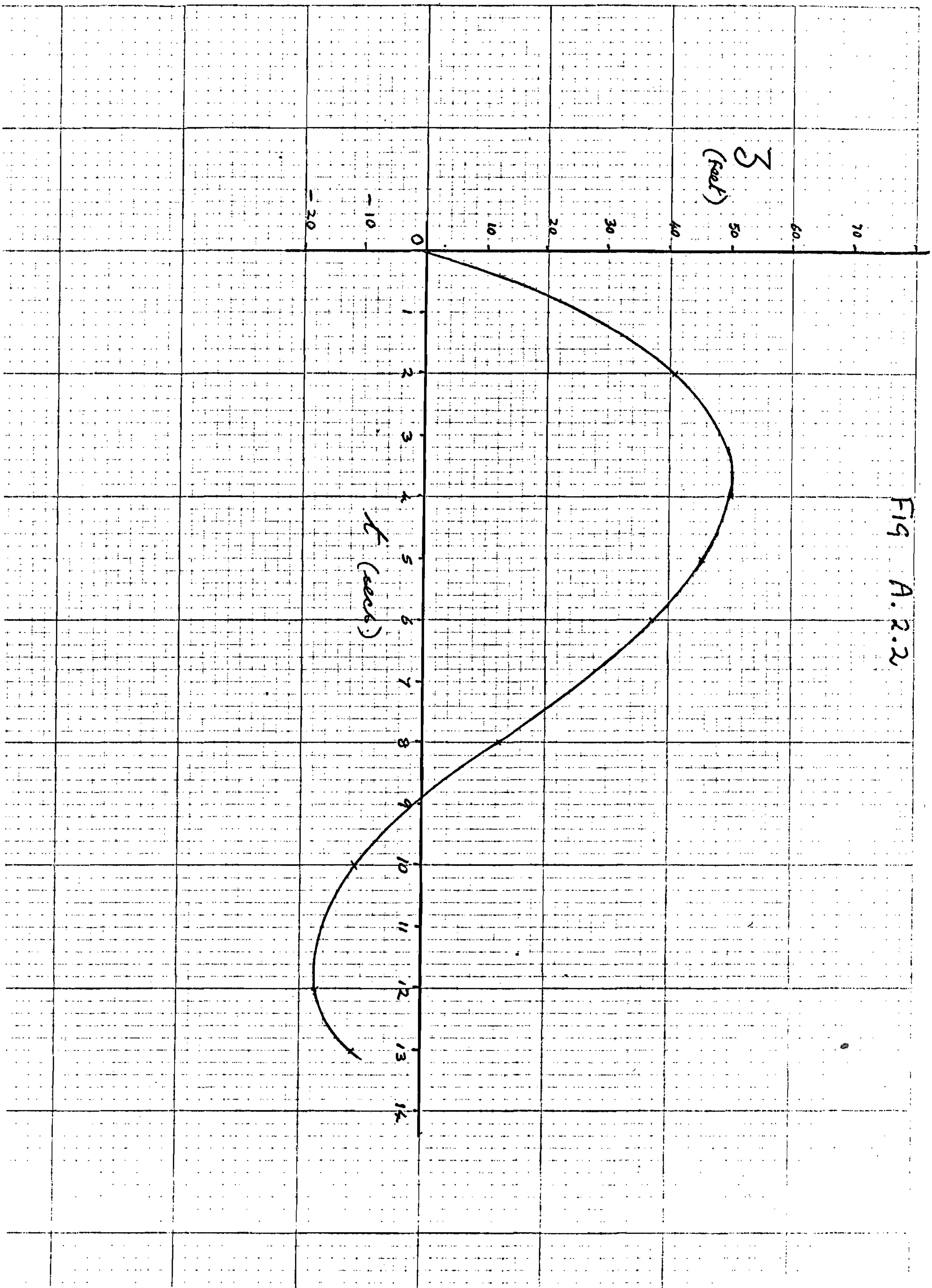


FIG. A.2.2

to (iv) may be omitted).

This results in a curve of the form shown in Fig A2.2.

In this instance point P occurs at a value $z = -17.6$ feet at $t = 12$ seconds. To correct this Vo should be increased by $\frac{17.6}{12} = 1.47$ ft/sec.

(vi) As Vo and Uo are now known, V_L and α may easily be calculated.

Appendix 3

E.A.I Computer Program and Scaling

The program is shown in Fig A3.1.

The equations used are:

$$0 = -m\dot{v} + T\cos\theta_T, \cos\gamma - (W - T\sin\theta_T)\sin\gamma - K_D V^2 \quad - \quad (5)$$

$$0 = -mv\dot{\gamma} - (W - T\sin\theta_T)\cos\gamma - T\cos\theta_T\sin\gamma + K_L V^2 \quad - \quad (6)$$

$$0 = -\dot{z} + v\sin\gamma \quad \text{———} \quad (7)$$

Scalings

(m.u = machine unit = 10 volts)

$\underline{\gamma}$ $\quad \pi$ radians $\equiv$ 0.9 m.u.

γ radians $\equiv \frac{0.9}{\pi}\gamma$ m.u. $= 0.286\,\gamma$

$\underline{V}$ $\quad$ Maximum value of v not exceeding 400 ft/sec.

$nV = 1$ m.u.

$n = \frac{1}{v} = \frac{1}{400}$ m.u./ft/sec

$\underline{V\dot{\gamma}}$ $\quad$ Maximum value $= 12$ ft/sec^2

$$\therefore \frac{v\dot{\gamma}}{12} \Big/ \frac{v}{400} = \frac{33.3\dot{\gamma}}{10} = 3.33\dot{\gamma}$$

$\underline{z}$ $\quad$ 1 m.u. = 200 ft.

Input of V_L was via potentiometer 2/5 and launch angle via potentiometer 2/6.

Potentiometer values:

$$2/0 = \frac{400\,K_D}{m} \times 20$$

$$0/0 = \frac{W - T\sin\theta_T}{400\,m} \times 20$$

$$0/1 = \frac{T \cos \theta_T}{400\ m} \times 20$$

$$0/2 = \frac{W - T\sin \theta_T}{12\ m \times 10}$$

$$2/1 = \frac{K_L}{m} \times \frac{4 \times 10^4}{3}$$

$$0/3 = \frac{T \cos \theta_T}{12\ m \times 10}$$

$$2/2 = \frac{0.9}{\pi \times 3.33} \times \frac{20}{10} = 0.172 \text{ constant}$$

$$1/0 = 0.1 \text{ constant}$$

Some typical results by pen recorder are shown in Figs A3.2 and A3.3. In all cases the launch angle is 30° and θ_T is 60° constant. Height scales, in feet, are approximate.

FIG A.3.1.

FLIGHT PATH EQUATIONS

PROGRAM FOR E.A.I 580 ANALOGUE COMPUTER

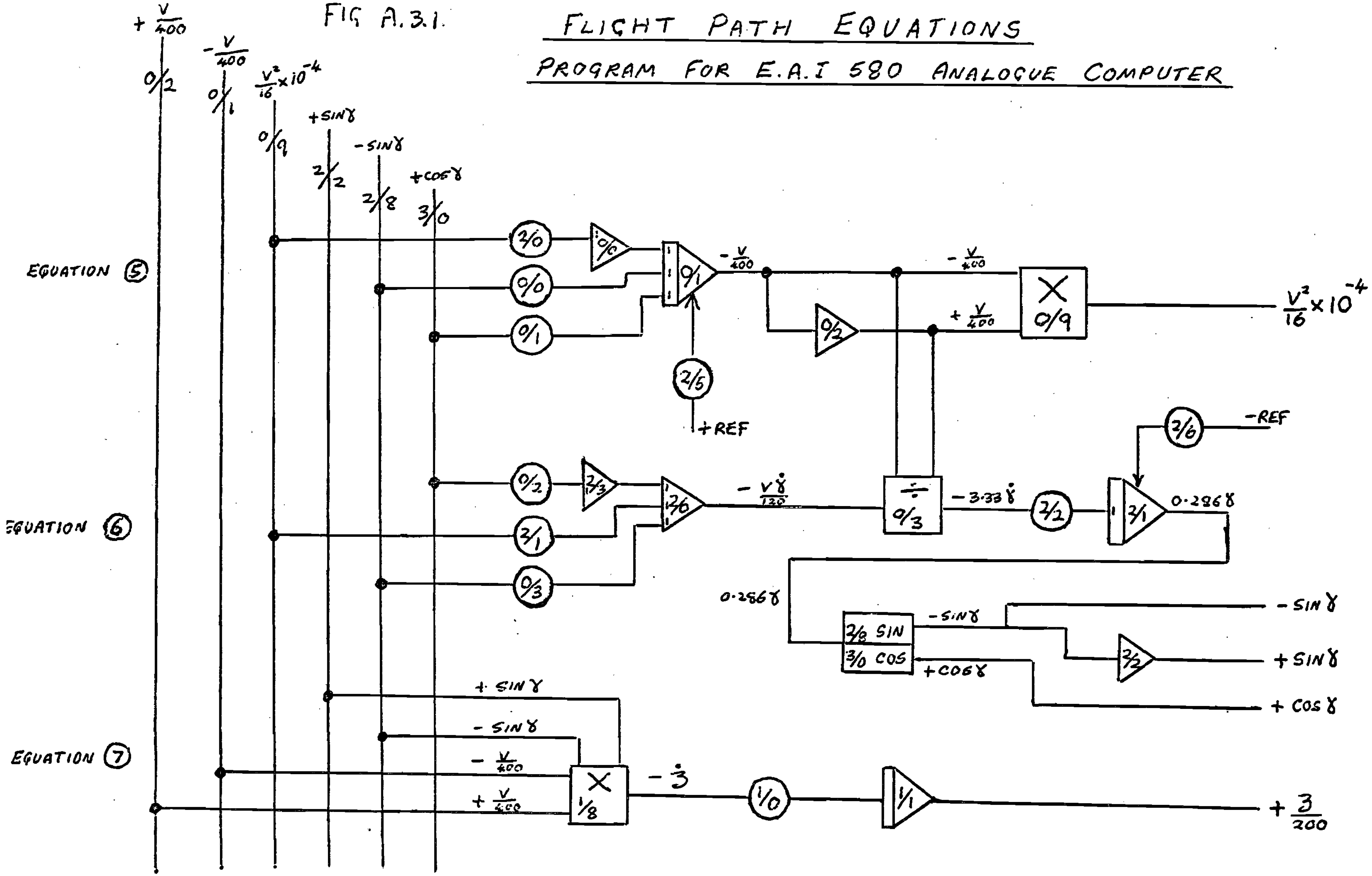
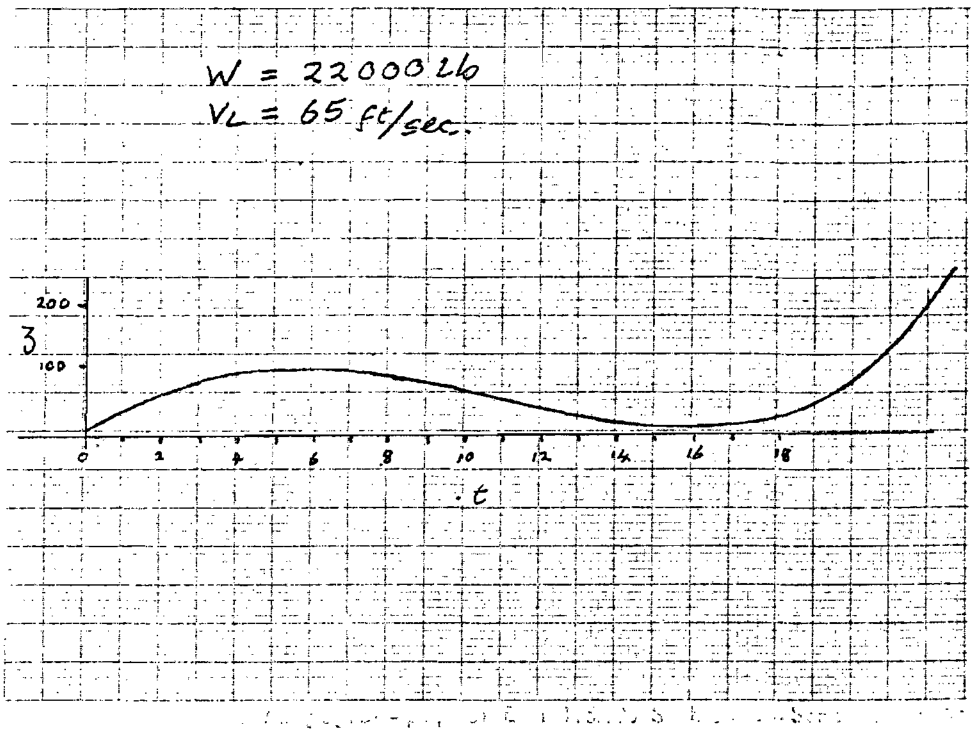


FIG A3.2


W = 22000 lb
VL = 65 ft/sec
200
100
3
0
2
4
6
8
10
12
14
16
18
t

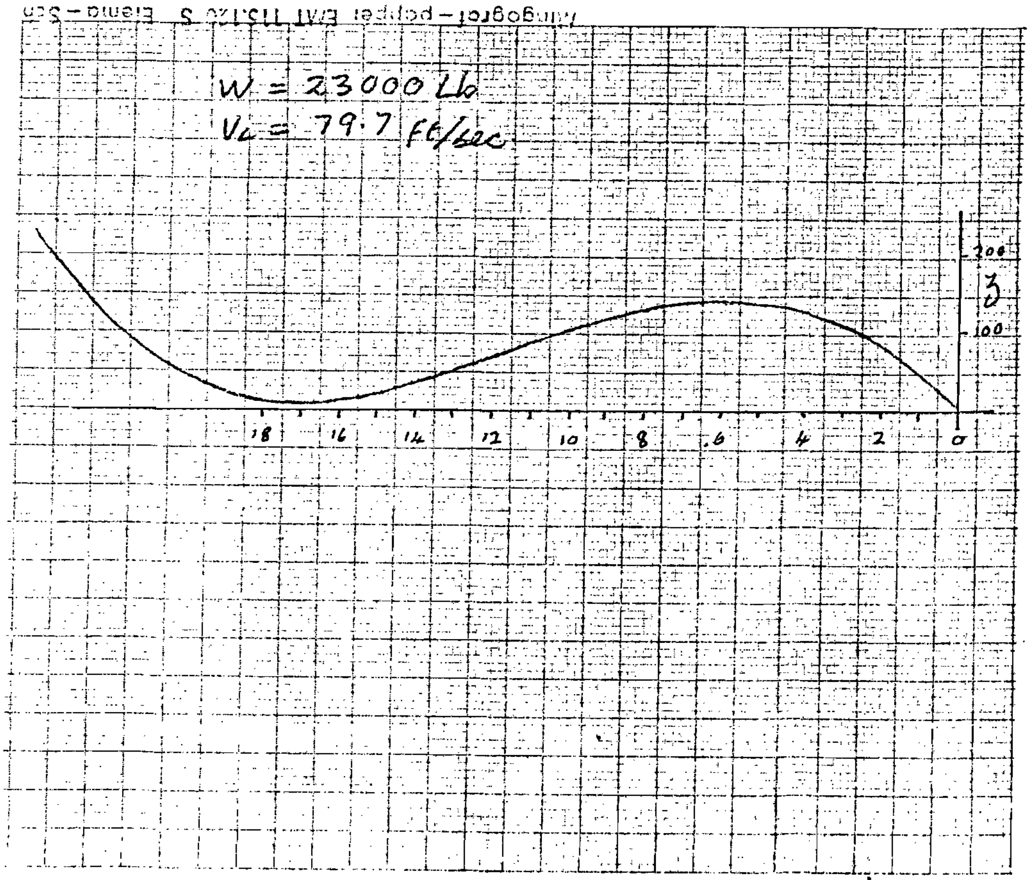
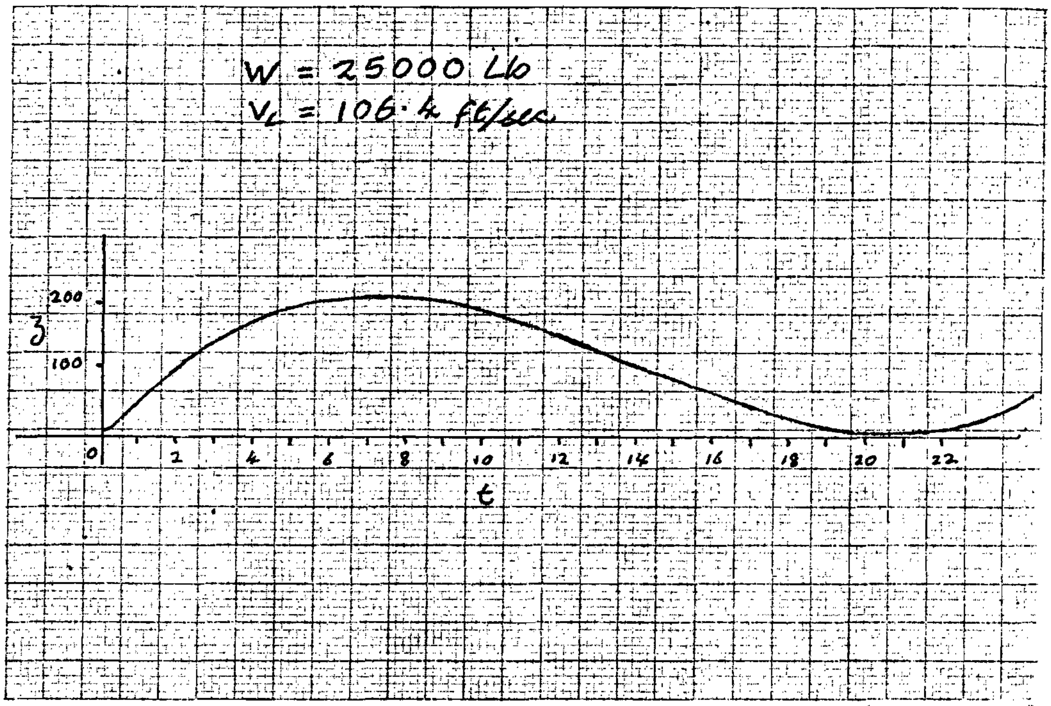
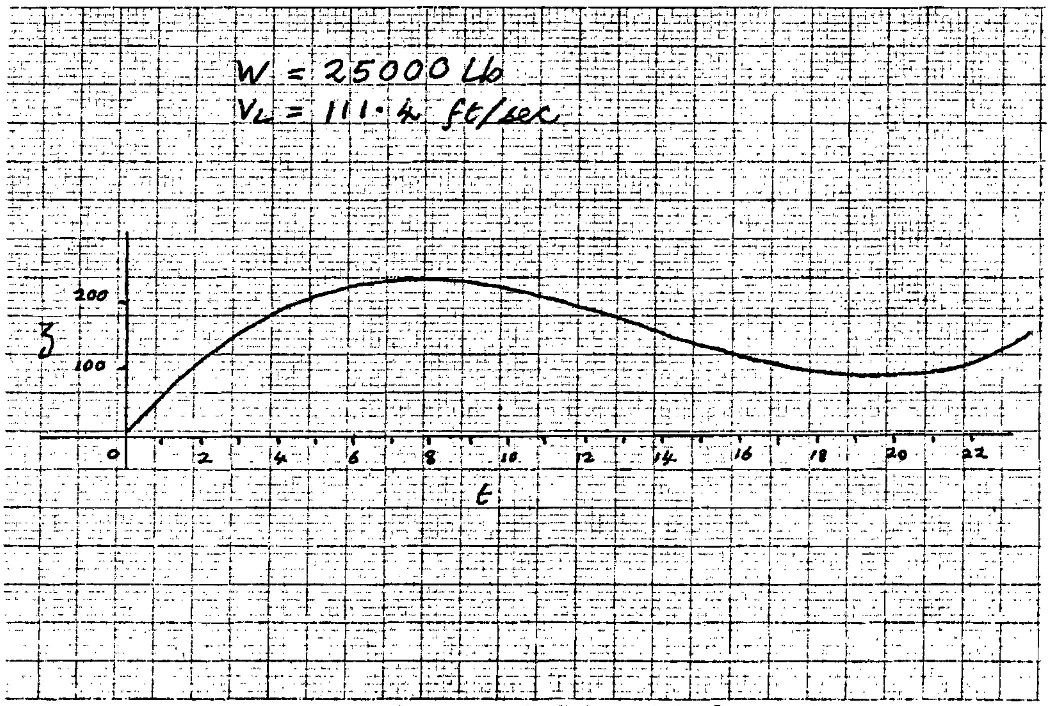

W = 23000 lb
VL = 79.7 ft/sec
200
100
3
18
16
14
12
10
8
6
4
2
0

FIG A3.3


W = 25000 Lb
V_L = 106·4 ft/sec
200
100
3
0
2
4
6
8
10
12
14
16
18
20
22
t


W = 25000 Lb
V_L = 111·4 ft/sec
200
100
3
0
2
4
6
8
10
12
14
16
18
20
22
t

Appendix 4

Effect of Wind over Deck (w.o.d.)

For a typical example consider the case of the hypothetical aircraft under the conditions:

$$W = 24{,}000 \text{ Lbs}, \ \theta_T = 60^o$$

At a launch angle of 15^o, required value of V_L is 120 ft/sec.

$$Uo = 120 \text{ Cos } 15^o = 115.9 \text{ ft/sec.}$$

$$Vo = 120 \text{ Sin } 15^o = 31.05 \text{ ft/sec.}$$

If w.o.d. = 15 knots = 25 ft/sec then the value of Uo to be imparted to the aircraft is:

$$(115.9 - 25) = 90.9 \text{ ft/sec.}$$

Vo is unaffected = 31.05

$$\therefore V_L \text{ at 15 knots w.o.d.} = \sqrt{90.9^2 + 31.05^2} = 96.05 \text{ ft/sec}$$

$$\text{Tan}\,\alpha = \frac{31.05}{90.9} \text{ and } \alpha = 18.8^o$$

Similarly, at $\alpha = 20^o$, $V_L = 108$ ft/sec,

$$Uo = 101.5 \text{ ft/sec}, \ Vo = 36.9 \text{ ft/sec}$$

At 15 knots w.o.d. $Uo = (101.5 - 25) = 76.5$ ft/sec

$$\text{and } V_L = \sqrt{76.5^2 + 36.9^2} = 84.9 \text{ ft/sec}$$

$$\alpha = 25.75^o$$

By this means a table of V_L and α for various values of w.o.d. can be constructed for each weight as in the following example:

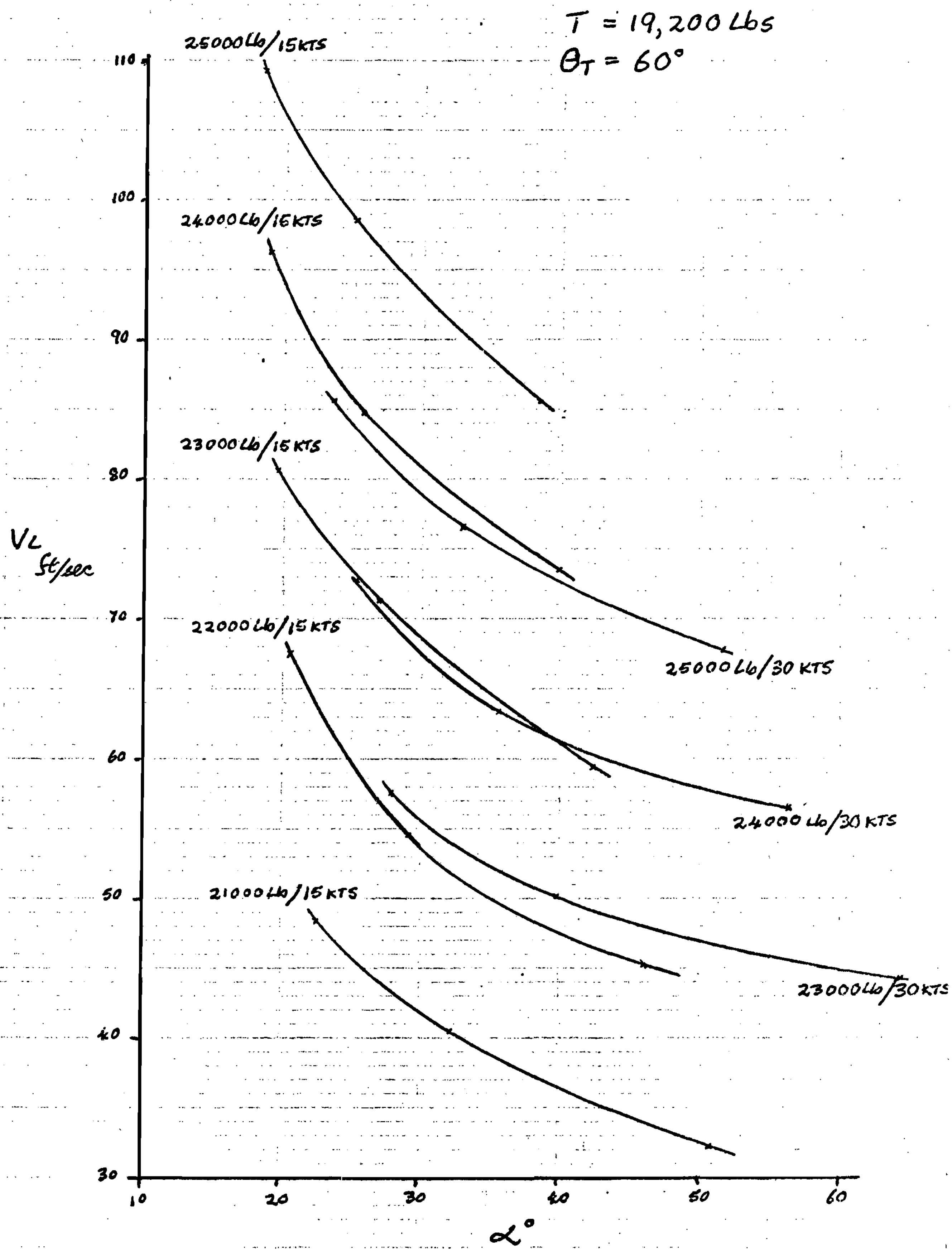
W = 24,000 Lbs

Uo	Vo	V_L	α	
115.9	31.05	120	15°	
101.5	36.9	108	20°	w.o.d. = 0
81.4	47	94	30°	
90.9	31.05	96.05	18.8°	
76.5	36.9	84.9	25.75°	w.o.d. = 15 knots
56.4	47	73.4	39.8°	
65.9	31.05	72.80	25.22°	
51.5	36.9	63.35	35.62°	w.o.d. = 30 knots
31.4	47	56.5	56.25°	

From these tables a series of curves can be constructed as in Fig A4.1.

FIG A.4.1

EFFECT OF WIND OVER DECK



Appendix 5

Effect of Delaying Thrust Deflection

Launch conditions: W = 22,000 Lbs, T = 19,200 Lbs, $\alpha = 30^\circ$.

$\theta_j = 0^\circ$. $V_L = 65$ ft/sec.

It is assumed that $\theta_J = 0^\circ$ until t = 1 second at which time $\theta_T = 60^\circ$ is selected. Nozzles reach this position at t = 1.5 seconds.

The calculation is done in two stages, from t = 0 to t = 1.5 by manual integration of equations (1) and (2). From t = 1.5 onwards, by use of equation (4).

t	$\theta_j + \alpha$	F	L_T	α	u	v	$\dot{u}$	$\dot{v}$	z 1st Approx	z 2nd Approx
0	30°	16,627	9,600	30°	56.29	32.5	23.78	-17.43	0	0
1	*24°	17,540	7,809	*24°	80.07	15.07	25.17	-19.49	23.78	22.75
1.5	60°	9,600	16,627	*18°	92.65	5.32	13.62	- 6.17	33.98	36.67

	z
1.5	36.67
3.5	35.61
5.5	14.12
7.5	- 22.63
9.5	- 68.42
11.5	-116.03
13.5	-157.34
15.5	-183.08
16.5	-186.95
17.5	-183.02
18.5	-169.86

* It is assumed that the pilot maintains an angle of attack of 13°. This is therefore added to the calculated value of α.

Appendix 6

Stability and Control Calculations

All calculations are for a weight of 23000 Lbs which is considered to be a working maximum for the immediate future.

The following data were used:

Moments of Inertia:

Roll	0.405×10^{6} Lb ft^{2}
Pitch	0.987×10^{6} Lb ft^{2}
Yaw	1.264×10^{6} Lb ft^{2}

Maximum Reaction Control Moments:

Roll	$\pm$ 9,200 Lb ft
Pitch	+ 16,000 Lb ft
	- 23,500 Lb ft
Yaw	$\pm$ 13,000 Lb ft

Air Mass Flow at hovering power = 400 Lb/sec.

Lift Curve Slope, $\Delta C_L = 3.15$ per radian.

Wing Area = 201 square feet

(a) Angular Accelerations due to Reaction Controls

Yaw Control Moment = $\pm$13,000 Lb ft

$\therefore$ Max angular acceleration

$$= \frac{13{,}000 \text{ Lb ft} \times 32.2 \text{ ft} \times 57.29 \text{ degrees}}{1{,}264{,}000 \text{ Lb ft}^2 \text{ sec}^2 \quad \text{radian}}$$

$$= \underline{18.95 \text{ degrees/sec}^2}$$

Similarly, in pitch:

Max angular accelerations: Nose up 29.8 degrees/sec^2

Nose down 44 degrees/sec^2

(b) Intake Momentum Drag

The drag force is assumed to act through a point in the centre of the front face of the engine, 7.5 feet forward of the aircraft CG.

Momentum drag $= \frac{mv}{g}$

where m = mass flow and v = forward speed.

If the aircraft has an initial yaw angle the momentum drag produces a yawing moment equal to:

Momentum drag x 7.5 Tan (initial yaw angle).

The following table can thus be constructed.

Speed	Drag	Yawing moment (Lb ft) at Initial Yaw Angle			
(knots)	Lbs	5°	10°	15°	20°
30	623	409	816	1230	1715
50	1034	688	1355	2040	2700
70	1440	944	1886	2840	3770
90	1860	1220	2420	3670	4870

We are particularly interested in the 50 knot case since this is the launch speed of the Harrier at a weight of 23,000 Lbs. It is also likely to represent a worst case because the fin becomes effective at higher speeds.

The angular acceleration due to intake momentum drag at 50 knots can now be calculated in a similar manner to that due to Reaction Controls and is as follows:

Initial yaw angle	5^o	10^o	15^o	20^o
Angular Acceleration (deg/sec^2)	1.02	1.97	2.93	3.94

(c) Effects of Initial Yaw

If the aircraft is launched at an initial yaw angle of 10^o and the yaw is allowed to develop for two seconds before an opposing control movement is made, then:

Angular acceleration due to momentum drag = $1.97^o/\text{sec}^2$

$\therefore$Aircraft yaws additionally $\dfrac{1.97 \text{ deg} \times 2 \text{ secs} \times 2 \text{ secs}}{\text{sec}^2 \times 2}$

$= 3.94^o$

$\therefore$Final yaw angle (1st approximation) $= 10 + 3.94$

$= 13.94^o$

But angular acceleration due to momentum drag at yaw angle of 13.94^o is $2.72^o/\text{sec}^2$.

$\therefore$Average angular acceleration over 2 seconds

$$= \frac{1.97 + 2.72}{2} = 2.345^o/\text{sec}^2$$

$\therefore$Additional yaw angle $= \dfrac{2.345 \times 2 \times 2}{2} = 4.690^o$

$\therefore$Final yaw angle $= 10 + 4.69 = 14.69^o$

(2nd approximation)

Angular acceleration at $14.69^o = 2.86^o/\text{sec}^2$

$\therefore$Average angular acceleration over 2 seconds

$$= \frac{1.97 + 2.86}{2} = 2.41^o/\text{sec}^2$$

$\therefore$Additional yaw angle $= \dfrac{2.41 \times 2 \times 2}{2} = 4.82^o$

(3rd approximation)

$\therefore$Yaw angle after 2 seconds $= 14.82^o$

Similar calculations were made for initial yaw angles of 5^{o}, 15^{o} and 20^{o}.

This simple method can only yield approximate results but they suffice for present purposes as they err on the pessimistic side. The acceleration of the aircraft during the two seconds will, in fact, have brought the aircraft to a speed at which the fin is providing significant damping and restoring forces.

(d) Restoration of Un-yawed flight

If an average of 0.6 of the maximum reaction control power is used to oppose yaw then the time required to restore the aircraft to un-yawed flight can be calculated as follows:

Assuming initial yaw angle of 10^{o}

Yaw angle after 2 seconds $= 14.82^{o}$

Maximum yaw rate $= \dfrac{2.41 \text{ deg} \times 2 \text{ secs}}{\text{sec}^2}$

$= 4.82^{o}/\text{sec}$

Average angular acceleration produced by jet reaction control in yaw $= 0.6 \times 18.95$

$= 11.4^{o}/\text{sec}$

$\therefore$ Time to reduce yaw rate to zero $= \dfrac{4.82}{11.4}$

$= 0.422$ secs

Time to reduce yaw angle to zero

$$= \sqrt{\frac{2 \times 14.82}{11.4}} = 1.61 \text{ secs}$$

$\therefore$ Time to achieve zero yaw $= 0.422 + 1.61$

$= \underline{2.03 \text{ secs}}$

Similar calculations were made for the other initial yaw angles.

Again the results are pessimistic as the effect of the rudder is ignored.

(e) <u>Effect of Divergencies in Pitch</u>

Consider the case in which, 5 seconds after launch the aircraft is at optimum attitude but pitching nose-down at 6^{o}/sec. The pitch rate then requires to be reduced to zero and the pilot uses 0.6 of maximum reaction control power in pitch to achieve this:

Nose-up pitching acceleration from reaction controls

$$= 0.6 \times 29.8 = 17.94^{o}/\text{sec}^2$$

$\therefore$ Time to reduce pitch rate to zero

$$= \frac{6}{17.94} = 0.334 \text{ secs}$$

$\therefore$ Maximum divergence from optimum attitude during this time

$$= 6 \times 0.334 - \frac{17.94 \times 0.334^2}{2}$$

$$= 1.006^{o}$$

For simplicity, average divergence is assumed to be half this value, ie 0.503^{o}.

Time required to return to optimum attitude

$$= \frac{1.006}{17.94} = 0.056 \text{ secs}$$

$\therefore$ Total time diverged from optimum

$$= 0.334 + 0.056 = 0.39 \text{ secs}$$

Change in C_L for divergence to 0.503^{o}

$$= \frac{3.15}{57.29} \times 0.503$$

$$= -0.0277$$

Lift constant, $K_L = \frac{\rho S C_L}{2}$ where S = 201

$\therefore$ Change in K_L due to divergence from optimum attitude

$$= -.0066$$

Vertical acceleration of aircraft,

$$\dot{v} = \frac{g}{W}(L_T - W + K_L v^2)$$

Where $L_T = 19{,}200 \sin 60^{o} = 16{,}627$ Lbs

$v = Uo + at$

Where Uo = horizontal component of launch speed

a = horizontal acceleration

$Uo = 81 \cos 30^{o} = 70.148$ ft/sec

$a = \frac{g}{W}(19{,}200 \cos 60^{o}) = 13.44$ ft/sec^2

$\therefore v = 70.148 + 13.44 \times 5 = 137.35$ ft/sec

$\therefore$ Change in vertical acceleration due to divergence from optimum attitude,

$$\dot{v} = \frac{32.2}{23{,}000}(16627 - 23000 - 0.006 \times 137.35^2)$$

$$= -0.175 \text{ ft/sec}^2$$

This change applies for 0.39 secs resulting in a change of vertical velocity of

$$0.39 \times -0.175 = -0.068 \text{ ft/sec}$$

Distance fallen during 0.39 secs $= \frac{0.175 \times 0.39^2}{2} = 0.0133$ ft

But the change in vertical velocity applies for the remainder of the semi-ballistic flight which lasts a total of 17 seconds.

$\therefore$ Total difference in height at the end of the semi-ballistic trajectory is

$$0.133 + (17 - 5) \times 0.068 = \underline{0.83 \text{ feet}}$$

A similar calculation was used for the case described at 1.14.11.

Appendix 7

Calculation of Harrier Take-Off Distance - Unassisted Ski-Jump

Refer to 2.8.3

W = 22,000 Lb. T = 19200 Lb. w.o.d = 0.

V_L = 72.6 ft/sec. Launch angle = 30°, $\therefore$ average value of acceleration on ramp occurs when $\alpha = 15^\circ$.

From (8),

$$a_2 = \frac{32.2}{22000}\left(19200 - 22000 \sin 15^\circ\right) = 19.77 \text{ ft/sec}^2$$
(1st approximation).

From (9),

$$V_r = \sqrt{72.6^2 - 2 \times 19.77 \times 100 \times 0.5236}$$
$$= 56.6 \text{ ft/sec (1st approximation).}$$

From (10),

$$a_2 = \frac{32.2}{22000}\left[19200 - 22000 \sin 15^\circ - 0.025 \times 22000\left(\frac{64.6^2}{32.2 \times 100} + \cos 15^\circ\right)\right]$$

$$= 17.95 \text{ ft/sec}^2 \text{ (2nd approximation)}$$

$$\therefore V_r = \sqrt{72.6^2 - 2 \times 17.95 \times 100 \times 0.5236}$$
$$= 58.23 \text{ ft/sec.}$$

From (11),

$$= \frac{58.23^2 \times 22000}{2 \times 32.2 \times (19200 - 0.025 \times 22000)}$$

$$= 62.1 \text{ ft}$$

Pitching motion applied to aircraft

$$= \frac{72.6}{100} = 0.726 \text{ radians/sec.}$$

$$I = 30{,}900 \text{ Lb sec}^2/\text{ft when } = 22000 \text{ Lb.}$$

$$I_2 = \frac{30{,}900 + 22000 \times 4.25^2}{32.2} = 43200 \text{ Lb sec}^2/\text{ft.}$$

$\therefore$ Resultant pitching motion on leaving curved ramp is, from (12),

$$= \frac{72.6 - 4.25 \times 22000 \times 11.4 \left(\frac{72{,}6^2}{32.2 \times 100} + \text{Cos } 30^{\circ}\right)}{43{,}200 \times 72.6}$$

$$= -\ 0.1246 \text{ radians/sec} = -\ 7.14^{\circ}/\text{sec.}$$

Since pitch rate limit is 6°/sec a straight section 11.4 feet long must be added to the curved ramp.

$\therefore$ Total horizontal space required for take-off

$$= S + (\text{horizontal length of ramp}) + (11.4 \text{ Cos } 30^{\circ})$$

$$= 62.1 + 50 + 9.872 = 121.97 \text{ feet.}$$

Say 122 feet.

Appendix 8

Calculation of Harrier Take-Off Distance - Catapult Assisted Ski-Jump

Refer to 2.9

$W = 25{,}000$ lbs, $T = 19{,}200$ lbs, w.o.d. $= 30$ knots,

$V_L = 81.5$ ft/sec. Launch angle $= 30^\circ$.

$R = 100$ ft.

Maximum permissible load on undercarriage when

$W = 25{,}000$ Lbs is 2.76g

$\therefore n = (2.76 - 1) = 1.76$

$$\therefore v = \sqrt{nRg} = \sqrt{1.76 \times 100 \times 32.2} = 75.28 \text{ ft/sec.}$$

$$L = \frac{2\pi R \times 30^\circ}{360^\circ} = \frac{2\pi \times 100 \times 30^\circ}{360^\circ} = 52.38 \text{ ft}$$

$$\therefore a = \frac{v^2}{2L} = \frac{75.28^2}{2 \times 52.38} = 54.09 \text{ ft/sec}^2 = 1.68g$$

$$\therefore d = \frac{V_L{}^2}{2a} = \frac{81.5^2}{2 \times 54.09} = 61.4 \text{ ft}$$

A straight section of 11.4 ft minimum is required to stabilise the aircraft. $(d - L) = 9.02$ ft but an additional 3 feet must be added for deceleration of the moving parts of the catapult. This provides the length required.

$\therefore$ Total horizontal space required = horizontal space occupied by curved ramp (50 ft) + $(d - L)\text{Cos } 30^\circ + 3 \text{ Cos } 30^\circ$.

$+ \quad = 50 + 9.02 \text{ Cos } 30^\circ + 3\text{Cos } 30^\circ = 60.4$ feet

Say, 61 feet

If R $=$ 75 feet but conditions are otherwise as before:

$$v = \sqrt{1.76 \times 75 \times 32.2} = 65.2 \text{ ft/sec}$$

$$L = \frac{2\pi \times 75 \times 30^\circ}{360^\circ} = 39.28 \text{ ft}$$

$$\therefore a = \frac{65.2^2}{2 \times 39.28} = 54.1 \text{ ft/sec} = 1.68g$$

$$\therefore d = \frac{81.5^2}{2 \times 54.1} = 61.39 \text{ ft}$$

$$\text{and } (d - L) = 22.1 \text{ ft}$$

Horizontal space occupied by curved ramp of 75 ft radius $=$ 37.8 feet.

$\therefore$ Total horizontal space required

$$= 37.8 + 22.1 \cos 30^\circ + 3 \cos 30^\circ$$

$$= 59.54 \text{ feet}$$

Say, 60 feet

Appendix 9

Ballista - Calculation of Stresses and Weight

(Refer to 2.10.11)

Maximum stresses are assumed to occur simultaneously.

Aircraft CG is 5 feet vertically above the platform and 11 feet horizontally from nose tow attachment point.

Platform

Consists of two tapered side members of constant thickness which take all loads. These side members are joined at the top by a non-structural plate 0.25 inch thick. Lower edges are stiffened by strips 2 inches x 0.25 inches (see Fig A.9.1.A)

Stresses on platform:

Maximum bending stress occurs at the end of launch.

Force applied $= \sqrt{Fv^2 + F_H^2} = 3.613g$

$= 3.613 \times 24{,}000$ Lb $= 86{,}709$ lbs

Since the platform is at 30° to the horizontal, direction of force relative to platform $= \mathrm{Tan}^{-1} \dfrac{-1.104}{3.44} + 30^\circ$

$= 17.7^\circ + 30^\circ = 47.7^\circ$

$\therefore$ Component of force normal to platform $= 86{,}709$ Sin 47.7

$= 64{,}133$ Lbs

$\therefore$ Loading of platform is as Fig A.9.1.B

$\therefore$ Maximum bending stress at pivot point $= 64{,}133 \times 4 \times 11$

where Z = section modulus $= \dfrac{bd^2}{6}$

where b = thickness of member, d = depth of member

If maximum permissible bending stress = 10 tons/in^2

Then, $Z = \dfrac{64133 \times 4 \times 11 \times 12 \times 12}{10 \times 15 \times 12 \times 2240}$

$$b = \frac{6Z}{d^2} = \frac{6 \times 64133 \times 4 \times 11 \times 12 \times 12}{10 \times 15 \times 12 \times 24 \times 24 \times 2240}$$

$$= 1.05 \text{ ins}$$

and since there are two side members, thickness of each

$$= \frac{1.05}{2} = \underline{0.525} \text{ ins}$$

Stress at a point 1 foot from front end of platform

$$= \frac{64133 \times 4 \times 1 \times 12 \times 12}{z \times 15 \times 12} = 10 \text{ tons/in}^2$$

$$\therefore Z = \frac{64133 \times 4 \times 1 \times 12 \times 12}{10 \times 15 \times 12 \times 2240} = 9.162$$

$$d = \sqrt{\frac{6Z}{b}} = \sqrt{\frac{6 \times 9.162}{1.05}} = 7.236 \text{ ins}$$

$\therefore$ If beam is a straight taper, depth at front of platform

$$= 7.236 - \left(\frac{24-7.236}{10}\right)$$

$$= \underline{5.56 \text{ inches}}$$

Stress at point 1 foot from rear end of platform

$$= \frac{64133 \times 11 \times 1 \times 12 \times 12}{z \times 15 \times 12} = 10 \text{ tons/in}^2$$

$$\therefore Z = \frac{64133 \times 11 \times 1 \times 12 \times 12}{10 \times 15 \times 12 \times 2240} = 25.195$$

$$d = \sqrt{\frac{6 \times 25.195}{1.05}} = 11.998 \text{ ins}$$

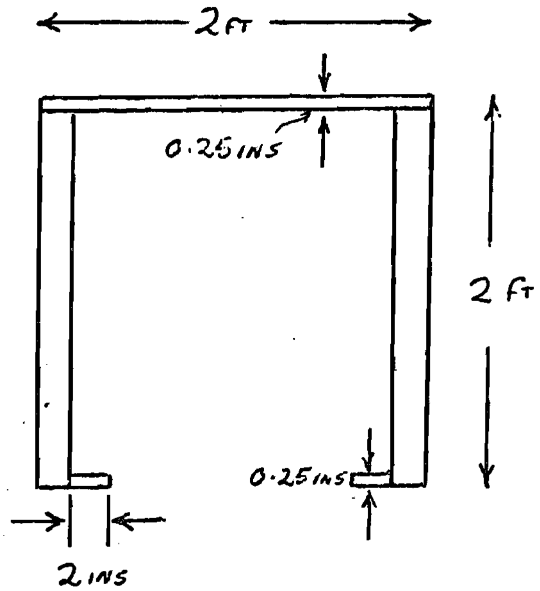
So depth at rear end of platform

$$= 11.998 - \left(\frac{24-11.998}{3}\right)$$

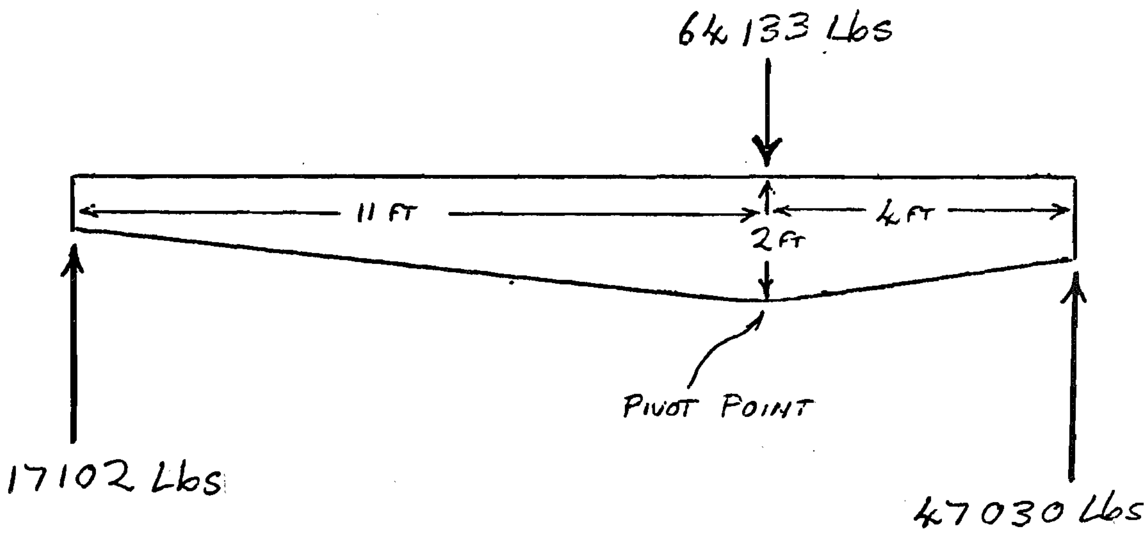
$$= \underline{7.998 \text{ inches}}$$

FIG A.9.1.

A. PLATFORM CROSS-SECTION



B. PLATFORM LOADING



Volume of steel in side plates

$$= \left[11 \times 12\left(\frac{24+5.56}{2}\right) \times 1.05 \right] + \left[4 \times 12\left(\frac{24+7.998}{2}\right) \times 1.05 \right]$$

$$= 2048.5 + 806.35 = 2855 \text{ ins}^3$$

Volume of steel in top plate $= 15 \times 2 \times 12 \times 12 \times 0.25 = 1080$ ins^3

Volume of steel in stiffening strips $= 15 \times 1 \times 12 \times 2 \times 0.25 = 180$ ins^3

$\therefore$ Total volume of steel in platform $= 2855 + 1080 + 180 = 4115$ ins^3

Steel weighs 0.2816 lb/in^3, so total weight of steel in platform

$$= 4115 \times 0.2816 = 1159 \text{ Lbs}$$

Assume bearings and housings weigh 30% of platform weight

$$= 348 \text{ Lbs}$$

$\therefore$ Total weight of platform $= 1159+348 = 1507$ Lbs

$= \underline{0.67 \text{ Tons}}$

Linkage

Maximum load on linkage arises from the pitching moment of the aircraft and platform about the top pivot which puts the linkage in compression. This occurs at the end of the launching stroke at $\theta = 50^\circ$.

At this point the force applied by the aircraft is 86709 Lbs at 47° to the platform. The line of action of this force through the aircraft CG passes 8.7 feet from the top pivot.

So maximum moment applied by aircraft

$$= 86709 \times 8.7 \times 12 = 9052419 \text{ Lb ins}$$

Centrifugal loading on platform $= \dfrac{\omega^2 r}{g}$

$$= \frac{1.787^2 \times 31}{32.2} = 3.07g$$

Centre of area of the portion of side plates forward of pivot point is

$$\frac{5.56}{24} \times 11 \times 12 = 30.58 \text{ inches from pivot.}$$

Centre of area of the portion of side plates aft of pivot point

is $\frac{7.998}{24} \times 4 \times 12 = 16$ inches from pivot

∴Resultant moment of side plates

$= (2048.5 \times 30.58 - 806.35 \times 16) \times 0.2816 \times 3.07$

$= 43002$ Lb ins

Resultant moment of top plate

$= (11 \times 12 \times 2 \times 12 \times 0.25 - 4 \times 12 \times 2 \times 12 \times 0.25) \times 0.2816 \times 3.07$

$= 534.7$ Lb ins

Resultant moment of stiffeners

$= (11 \times 12 \times 2 \times 2 \times 0.25 - 4 \times 12 \times 2 \times 2 \times 0.25) \times 0.2816 \times 3.07$

$= 72.62$ Lb ins

Moment of bearing at top of linkage which weighs 174 Lbs

$= -174 \times 4 \times 12 = -8352$ Lb ins

∴Resultant total moment of aircraft and platform

$= 9052419 + 43001 + 435.7 + 72.62$

$= 9095929$ Lb ins

This is reacted through links at a moment arm of 4 feet

∴Force applied to end of linkage

$= \frac{9095929}{4 \times 12} = 189499$ Lbs

There are two links. Each is assumed to be a tubular strut of uniform cross section and 31.5 feet in length.

Radius of gyration of cross section is estimated at 4 inches

∴$l/r = \frac{31.5 \times 12}{4} = 94.5$

Maximum allowable compressive stress at this value of l/r

$= 5.45$ Tons/in^2

Load on each strut $= \frac{189499}{2} = 94750$ Lbs

$\therefore$Cross section area of each strut $= \frac{94750}{5.45 \times 2240}$

$= 7.76 \text{ in}^2 = 0.7854\ (D^2 - d^2)$

Where D = outside diameter and d = inside diameter

Assume D = 8.3 ins

$\therefore$d = 7.68 ins and wall thickness = 0.31 ins

$\therefore$Volume of steel in each link = 31.5 x 12 x 7.76

$\therefore$Weight of each link = 31.5 x 12 x 7.76 x 0.2816

= 826 Lbs

Allow 25% for bearings and end fittings = 206.5 Lbs

$\therefore$Weight of each link = 826+206.5 = 1032.5

$\therefore$Total weight of links = 2x1032.5 = 2065

= <u>0.92 Tons</u>

<u>Arm</u>

The arm consists of two tapered side members of constant thickness which take all loads. They are joined at the top by a non-structural top plate 0.25 ins thick and 2 feet wide. The lower edges are stiffened by plates 3 ins x 0.5 ins in section (see Fig A.9.2.A)

(a) Forces on arm: (First approximation)

The maximum bending moment occurs where the jack joins the arm 19 feet from the lower pivot point.

(i) Moment applied by aircraft.

This is maximum at $\theta = 50^{\circ}$. The moment arm of the reaction force at $\theta = 50^{\circ}$ is also 19 feet.

$\therefore$Moment applied by aircraft = 19x86709 Lb ft.

$\therefore$ Reaction force normal to arm required to balance this at jack attachment radius

$$= \frac{19 \times 86709}{19} = \underline{86709 \text{ Lbs}}$$

(ii) Moment applied by platform.

This also is maximum at $\theta = 50^{\circ}$

Reaction force applied by platform $= r\sqrt{\alpha^2 + \omega^4}$

$$= 31\sqrt{1.832^2 + 1.787^4} = 3.544g$$

$$= 3.544 \times 1507 = 5341 \text{ Lbs}$$

and angle of reaction is $\text{Tan}^{-1} \dfrac{1.832}{1.787^2}$

$$= 30^{\circ} \text{ to arm}$$

This line of reaction passes 11 feet from the lower pivot.

$\therefore$ Moment applied by platform = 5341x11 Lb ft

Force required to balance this at 29 feet radius.

$$= \frac{5341 \times 11}{19} = \underline{3092 \text{ Lbs}}$$

(iii) Inertia of links

If m = mass of each link

L = length of link

D = external diameter

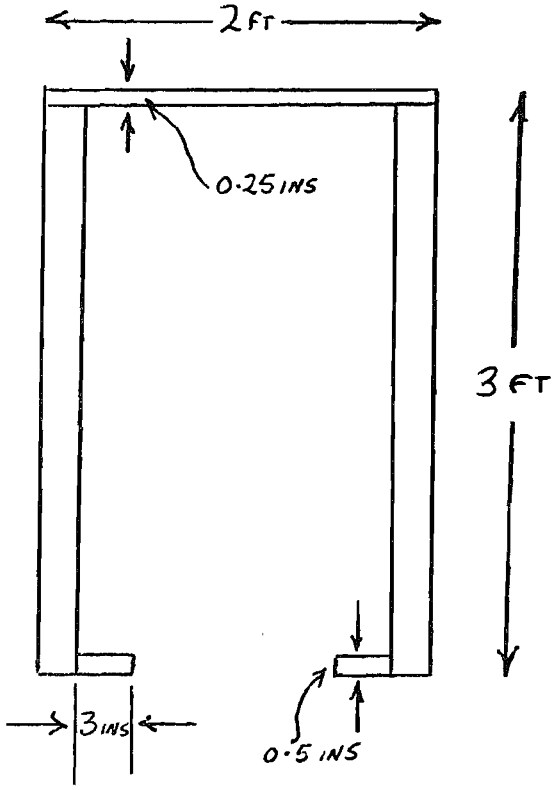
d = internal diameter

Then moment of inertia of each link about an axis through its CG.

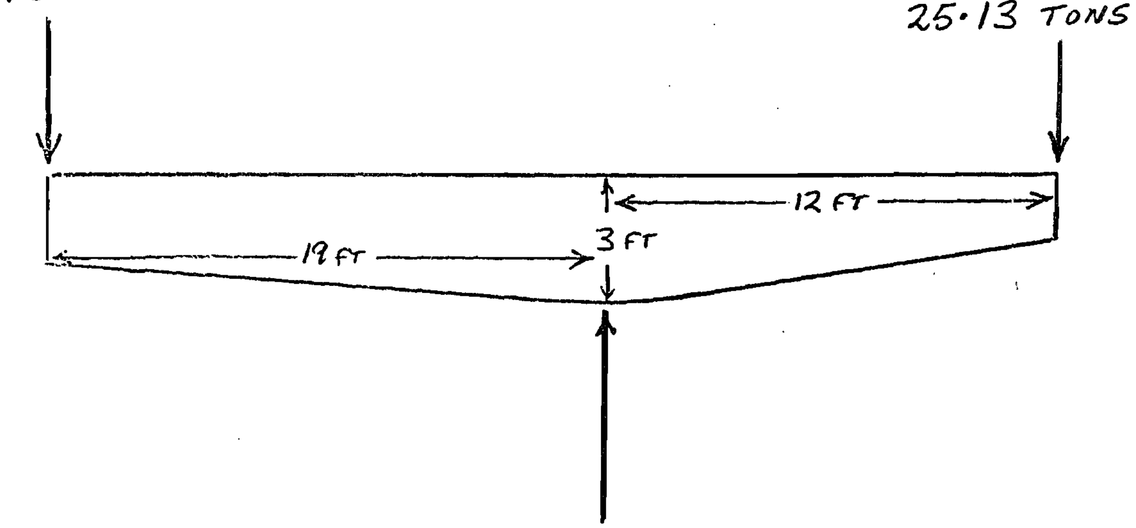
$$= m\left[\frac{L^2}{12} + \frac{(D^2+d^2)}{16}\right]$$

FIG A.9.2.

A. ARM CROSS-SECTION



B. ARM LOADING



$$= m\left[\frac{31.5^2}{12} + \left\{\frac{\left(\frac{8.3}{12}\right)^2 + \left(\frac{7.68}{12}\right)^2}{16}\right\}\right]$$

$$= 82.743m$$

$\therefore$ Moment of inertia of each link about its bottom pivot, distant 15.75 feet from CG.

$$= 82.743m + 15.75^2 m = 331m$$

$$= \frac{331 \times 1032.5}{32.2}$$

$\therefore$ Inertia force $= \frac{331 \times 1032.5}{32.2} \times 1.832$ radians/sec^2

$= 19433$ Lbs for each link

$\therefore$ Total inertia force $= 2 \times 19433$ Lbs

Force required to balance this at 19 feet radius

$$= \frac{2 \times 19433}{19} = 2046 \text{ Lbs}$$

(iv) Total force required normal to arm at 19 feet radius

$$= 86709+3092+2046$$

$= 91847$ lbs $=$ <u>41 Tons</u>

(b) The arm may now be considered as a beam loaded as in Fig A.9.2.B

Stress at jack attachment point

$$= \frac{41 \times 19 \times 12 \times 12 \times 12}{Z \times 31 \times 12}$$

Maximum permissible bending stress = 10 Tons/in^2

$$\therefore Z = \frac{41 \times 19 \times 12 \times 12 \times 12}{10 \times 31 \times 12} = \frac{bd^2}{6}$$

If the maximum depth of the side members at the jack attachment point is 36 inches, then:

$$b = \frac{6 \times 41 \times 19 \times 12 \times 12 \times 12}{10 \times 31 \times 36 \times 36 \times 12} = 1.675 \text{ ins}$$

$\therefore$ Thickness of each side plate $= \frac{1.675}{2}$

$= \underline{0.838 \text{ ins}}$

Bending stress at a point 1 foot from top of arm

$$= \frac{41 \times 19 \times 1 \times 12 \times 12}{z \times 31 \times 12}$$

$$\therefore z = \frac{41 \times 19 \times 1 \times 12 \times 12}{10 \times 31 \times 12} = 30.155$$

$$\therefore d = \sqrt{\frac{6z}{6}}$$

$$= \sqrt{\frac{30.155 \times 6}{1.675}} = 10.39 \text{ ins}$$

$\therefore$ Depth at top of arm (assuming straight taper)

$$= 10.39 - \frac{(36-10.39)}{2} = \underline{8.256 \text{ ins}}$$

Bending stress at a point 1 foot from lower pivot

$$= \frac{41 \times 12 \times 1 \times 12 \times 12}{z \times 31 \times 12}$$

$$\therefore z = \frac{41 \times 12 \times 1 \times 12}{10 \times 31} = 19.045$$

$$\therefore d = \sqrt{\frac{6 \times 19.045}{1.675}} = 8.26 \text{ ins}$$

$\therefore$ Depth at lower pivot

$$= 8.26 - \frac{(36-8.26)}{19} = 6.8 \text{ ins}$$

But the jack applies an axial load on the arm below the jack attachment point.

The angle of the jack to the arm is nearly constant at 20°.

If the force applied by the jack is F then the force normal to the arm is F Sin 20° = 41 Tons

$\therefore F = 119.876$ Tons

$\therefore$ Axial load on arm below jack attachment point

$$= F \cos 20^{\circ} = 112.647 \text{ Tons}$$

If tensile stress must not exceed 9 Tons/in^2, additional cross section area needed

$$= \frac{112.647}{9} = 12.516 \text{ ins}^2$$

$\therefore$ Each side plate must be increased in depth below jack attachment point by $\frac{12.516}{1.675}$

$$= 7.472 \text{ ins}$$

So, side plate dimensions become:

At bottom pivot $6.8+7.472 = 14.272$ ins

A jack attachment $36+7.472 = 43.472$ ins

At top pivot $= 8.256$ ins

With constant thickness of 0.838 ins

$\therefore$ Weight of each side plate

$$= \left[\left(\frac{14.272+43.472}{2}\right) \times 0.838 \times 19 \times 12 + \left(\frac{36+8.256}{2}\right) \times 0.838 \times 12 \times 12\right] \times 0.2816$$

$$= 2305 \text{ Lbs}$$

$\therefore$ Total weight of side plates $= 2 \times 2305$

$= 4610$ Lbs

If length of top plate is 28 feet

Then weight of top plate

$$= 28 \times 12 \times 2 \times 12 \times 0.25 \times 0.2816 = 568 \text{ Lbs}$$

Weight of stiffening strips

$= 2 \times 31 \times 12 \times 2 \times 0.5 \times 0.2816 = 314$ Lbs

Add 500 Lbs for jack attachment bearing

$\therefore$ Total weight of arm $= 4610+568+314+500$

$= 5992$ Lbs $= \underline{2.675 \text{ Tons}}$

to first approximation

Note: Axial loading due to centrifugal force of arm and platform is small and has been ignored.

(c) Second approximation for weight of arm:

The CG of the arm is about 17 feet from the lower pivot point. For calculation of inertia the arm is treated as a solid uniform rod rotating about the lower pivot.

If length of arm = l, then moment of inertia about CG $= \frac{ml^2}{12} = \frac{31^2 m}{12}$

$= 80.083m$

Moment of inertia about lower pivot

$= 80.03m + 17^2 m$

$= 369m = I$

Torque required to produce $= 1.832$ radians/sec^2 in arm

$= I\alpha = \frac{369 \times 5992 \times 1.832}{32.2}$

Force required to produce this torque at 19 foot radius

$= \frac{369 \times 5992 \times 1.832}{32.2 \times 19}$

$= 6621$ lbs $= 2.956$ Tons

So, new total force normal to arm $= 41+2.956$

$= 43.956$ say $\underline{44 \text{ Tons}}$

and, $\therefore$ New value of $F = \frac{44}{\sin 20^\circ} = 128.65$ Tons

Then stress at jack attachment point

$$= \frac{44 \times 19 \times 12 \times 12 \times 12}{z \times 31 \times 12}$$

$$\therefore z = \frac{44 \times 19 \times 12 \times 12}{10 \times 31} = 388.335$$

$\therefore$ Thickness of each side plate

$$= \frac{6 \times 388.335}{2 \times 36 \times 36} = 0.899 \text{ ins}$$

Depth of side plates remains as before.

So, new weight of each side plate

$$= \left[\left(\frac{14.272 + 43.472}{2}\right) \times 0.899 \times 19 \times 12 + \left(\frac{36 + 8.256}{2}\right) \times 0.899 \times 12 \times 12 \right] \times 0.284$$

$= 2473$ Lbs

$\therefore$ Total weight of side plates $= 2 \times 2473$

$= 4946$ Lbs

Weight of top plate and stiffening strips remains the same and final weight of arm becomes

$4946 + 568 + 314 + 500 = 6328$ Lbs

$=$ <u>2.825 Tons</u>

<u>Size and Weight of Jack</u>

The maximum force required from the jack = 128.65 Tons.

If the operating pressure is 3000 Lb/in^2 then the area of the

$$\text{ram} = \frac{128.65 \times 2240}{3000}$$

$$= 96.059 \text{ ins}^2 = \pi r^2$$

$$\therefore \text{Ram diameter} = 2\sqrt{\frac{96.059}{\pi}} = 11.057 \text{ ins}$$

Total length of ram = 12 feet

Unsupported length = 9 feet

$\therefore l/r = 19.53$

At this value of l/r the maximum permissible compressive stress is 9.3 Tons/in^2

So, cross section area of ram $= \frac{128.65}{9.3} = 13.833$ in^2

If the ram is tubular in section and the outside diameter, D = 11.057 ins, then inside diameter

$$d = \sqrt{\frac{0.7854 \times 11.067^2 - 13.833}{0.7854}}$$

$$= 10.2296 \text{ ins}$$

and wall thickness = 0.414 ins

$\therefore$Weight of ram = 12 x 12 x 13.833 x 0.2816

= 560.93 Lbs = 0.25 Tons

Maximum permissible hoop stress in jack cylinder = 9 Tons/in^2

Using Lame's formula, $R = r\sqrt{\frac{S+P}{S-P}}$

Where R = outer radius of cylinder

S = maximum permissible hoop stress

P = internal pressure

r = internal radius of cylinder

In this case S = 9x2240 = 20160 Lb/in^2

and $r = \frac{11.057}{2} = 5.529$ ins

$$\therefore R = 5.529\sqrt{\frac{20160+3000}{20160-3000}} = 6.424 \text{ ins}$$

ie wall thickness = 0.895 ins

Cross section area $= 0.7854\ (D^2-d^2)$

$= 0.7854\ (12.848^2-11.57^2)$

$= 33.62\ \text{ins}^2$

$\therefore$Weight of jack cylinder $= 12 \times 12 \times 33.62 \times 0.2816$

$= 1364$ Lbs

Plus 200 lbs for bearing and housing

$= 1564$ Lbs $=$ <u>0.698 Tons</u>

Volume of jack cylinder $=$ 8 foot stroke $\times 12 \times \pi \times 5.529$

If the hydraulic fluid has specific gravity of 1 then the weight of fluid in the cylinder is

$$= \frac{8 \times 12 \times \pi \times 5.529^2 \times 62.5}{12 \times 12 \times 12} = 333.6 \text{ Lbs}$$

$= \underline{0.149 \text{ Tons}}$

<u>Weight of links connecting jack and arm pivot</u>

Angle between jack and links $= 72^{\circ}$

Component of jack force along links

$= 128.65\cos 72^{\circ} = 39.755$ Tons

Maximum permissible stress $= 9\ \text{Tons/in}^2$

$\therefore$Cross section area $= \frac{39.755}{9} = 4.417\ \text{ins}^2$

Length of links $= 8.5$ ft

$\therefore$Weight of links $= 8.5 \times 12 \times 4.417 \times 0.2816$

$= 126.87$ Lbs $=$ <u>0.057 Tons</u>

Weight of Moving Parts of Ballista

	Lbs	Tons
Arm	6328	2.825
Platform	1507	0.67
Linkage	2065	0.92
Jack Ram	560.93	0.25
Jack Cylinder	1564	0.698
Hydraulic Fluid	333.6	0.149
Bottom Links	126.87	0.057
Total Weight =	12485 Lbs	5.57 Tons

Appendix 10

Ballista - Deceleration of Moving Parts

With the exception of the main arm and the linkage, components are assumed to be compact masses at the end of weightless arms, ie the platform is a compact mass of 1507 Lbs on the end of a 30 foot arm. The jack assembly and bottom links are a mass of 2583 Lbs on a 7 foot arm.

Then, moments of inertia about lower pivot are:

Arm	$= \frac{6328 \times 369}{32.2}$	$= 72,516$
Links	$= \frac{2065 \times 331}{32.2}$	$= 21,227$
Platform	$= \frac{1507 \times 31^2}{32.2}$	$= 44,975$
Jack Assembly and bottom link	$= \frac{2585 \times 7^2}{32.2}$	$= 3,934$

∴ Total moment of inertia of moving parts = 142,652 Lbs ft sec^2

Until the jack reaches the end of its stroke the jack assembly and bottom link can be considered as not moving.

∴ Total angular momentum just before the end of the jack stroke

$= (72516+21227+44975) \times 1.787$ radians/sec

$= 247889$ Lbs ft sec

∴ Angular velocity of all moving parts just after the end of the jack stroke $= \frac{247889}{142652} = \underline{\underline{1.738 \text{ radians/sec}}}$

Retardation forces produced by compression of Bolster

It is assumed that the bolster is made of an airtight impermeable fabric of cylindrical section and a constant 2 feet in diameter. The length is 17 feet and the bolster is compressed between two flat surfaces. When compressed the cross section changes from circular to a rectangle plus two semi circles and the vertical section is modified by the imposition of a flat face at the side. These changes are shown in Fig A.10.1

Symbols used are as follows:

D = original diameter of bolster

x = percentage reduction in diameter when compressed

b = width of flattened surface

l = length of flattened face

h = length of remaining tubular section ignoring hemispherical end.

The circumference remains constant at 6.28 feet

At a given point on the flattened portion of the bolster the total circumference of the semi-circular portions of the cross section is

$$\pi\,(D-xD)$$

$$\therefore b = \frac{6.28-\pi\,(D-xD)}{2}$$

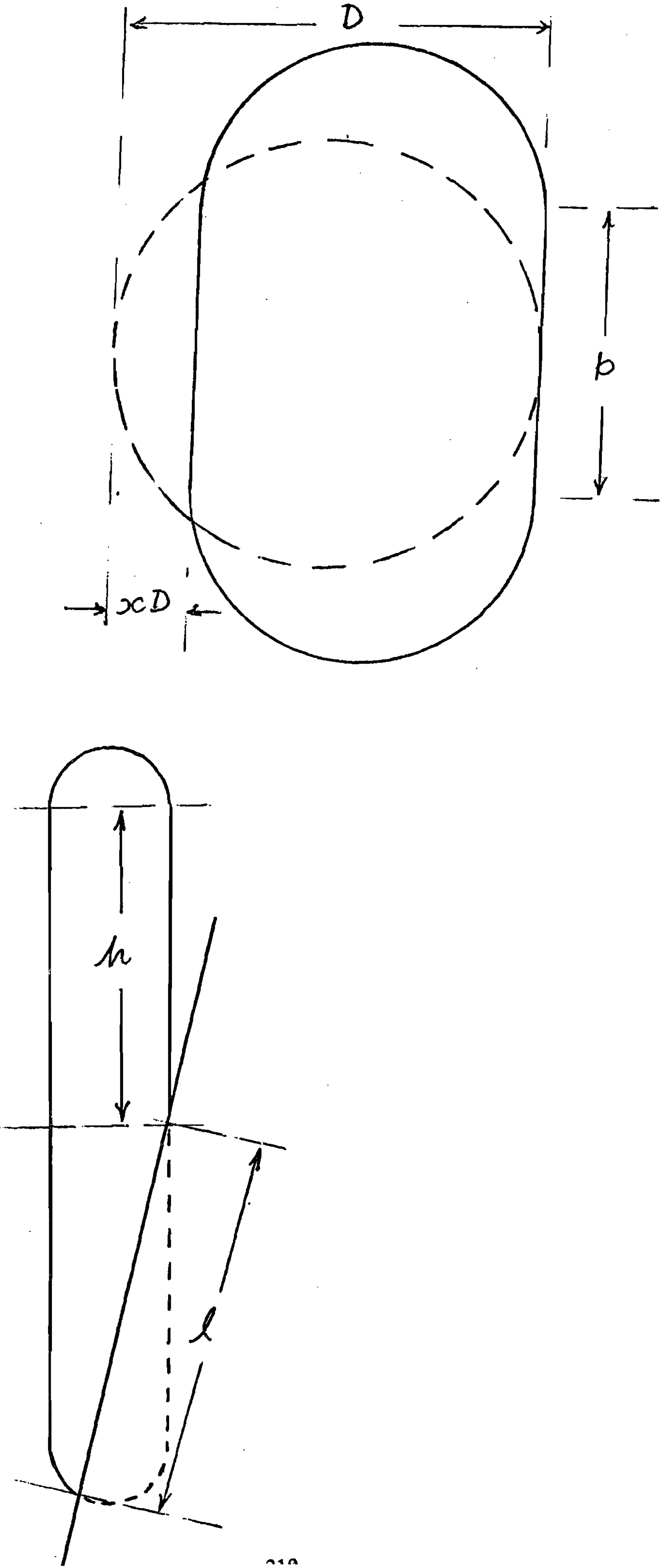
$\therefore$ Area of compressed cross section

$$= b(D-xD)+\frac{\pi(D-xD)^2}{4}$$

$$= (D-xD)\;\frac{6.28-\pi(D-xD)}{2} + \frac{\pi(D-xD)^2}{4}$$

This formula yields the following results:

FIG A10.1 COMPRESSION OF BOLSTER



x(%)	Area (ft^2)
0	3.14
20	3.01
40	2.64
60	2.01
80	1.13

The initial volume of the bolster $\approx 3.14x(17-1) + \frac{4\pi}{3}$

$$\approx 54.43 \text{ ft}^3$$

Volume when compressed $\approx \frac{2\pi}{3} + 3.24h + \left[(17-1)-h\right]$ x area at 1/2

If it is assumed that compression of the air in the bolster follows the law $PV^{1.4}$ = Constant then the following table can be constructed where:

θ = angle of ballista arm

V = volume

M = moment arm of retarding force

P = pressure

A = effective area (bxl)

θ	V (ft^3)	M (ft)	P	b (ft)	l (ft)	A (ft^2)	Total Moment (MxAxP)
50^o	54.43	0	1	0	0	0	0
60^o	52.08	2.6	1.063	0.628	2	1.256	3.47
70^o	51.19	3.6	1.089	0.93	4	3.72	14.58
80^o	48.2	6	1.185	1.2	9	10.8	76.79
85^o	39.7	10	1.55	1.57	16.5	25.9	401.5

It should be noted that the pressure is expressed as multiples of the original pressure. The total moment is a function of this and thus the figure quoted for total moment is a factor, not an absolute value.

The average total moment factor from $\theta = 50^\circ$ to $\theta = 85^\circ$ is 99.27

Deceleration of moving parts due to gravity

Moments at $\theta = 50^\circ$:

Weight of Part		Moment Arm	Moment
Platform	1507 Lbs	19 feet	28633
Arm	6328	14	88592
Links	2065	15	30975
Jack Assembly	2459	9	22131
Bottom Links	127	3.5	445

$\therefore$ Total Moment at $\theta = 50^\circ$ = 170776 Lb ft

" " " " $= 60^\circ$ = 132840

" " " " $= 70^\circ$ = 90868

" " " " $= 80^\circ$ = 46135

$\therefore$ Mean moment due to gravity = 110155 Lb ft

The forces bringing the arm to rest are provided by gravity, by compression of the bolster of the rebound buffer. The gravitational force is fixed and the remaining work is shared between the bolster and the rebound buffer in proportions which are a matter of choice.

If we assume that the arm is brought to rest at $\Theta = 85^o$ then the angular distance available for deceleration is $85^o - 50^o = 35^o = 0.6109$ radians.

If we choose a constant retardation force from the rebound buffer of 10 Tons operating at a moment arm of 5 feet, then:

$$\text{Average deceleration required} = \frac{1.738^2}{2 \times 0.6109}$$

$$= 2.472 \text{ radians/sec}^2$$

∴Total decelerating moment to be provided

$= 2.472\ I$ where I is the moment of inertia of the moving parts

$= 2.472 \times 142652$

$= 352636$ Lb ft

Moment provided by gravity $= 110155$ Lb ft

Moment provided by rebound buffer $= 10 \times 2240 \times 5 = 112000$ Lb ft

∴The bolster average total moment factor must be multiplied by

$$\frac{352636-(110155+112000)}{99.27}$$

$= 1314.4$

∴Initial bolster pressure $= \dfrac{1314.4}{144} = \underline{\underline{9.128 \text{ Lb/in}^2}}$

and maximum bolster pressure $= 9.128 \times 1.55$

$= \underline{\underline{14.148 \text{ Lb/in}^2}}$

Loads on Ballista due to deceleration of moving parts

It is necessary to check the loads due to deceleration to ensure that they are less than launching loads. For this purpose the loads imposed on the arm by the platform are a valid comparative yardstick.

Maximum decelerating moment of bolster

$= 10 \times 25.9 \times 14.148 \times 144$

$= 527664$ Lb ft at $\Theta = 85^o$

Decelerating moment of rebound buffer = 112000 Lb ft

∴ Total decelerating moment = 639664 Lb ft

∴ Maximum deceleration = $\frac{639644}{142652}$ = -4.48 radians/sec^2

∴ Force applied by platform = $\frac{-4.48 \times 31}{32.2}$ = $4.313g$

∴ Bending moment applied to arm by platform

= 4.313x1507 Lbs x 12 ft

= 77996 Lb ft

This compares with a total bending moment during the launching phase of:

5341x11 = 58751 (applied by platform)

86709x19 = 1647471 (applied by aircraft)

= 1706222 Lb ft total

This indicates that deceleration loads are well within the strength limits of the structure.

Appendix 11

Flexible Launcher - Calculation of Bag Weight and Thrust

Since the Flexible Launcher is not immediately applicable to aircraft launching, calculations are limited to sufficient accuracy to indicate approximate sizes, pressures and weight.

The launcher is required to impart to an aircraft weighing 24000 Lbs a velocity of 69 ft/sec at an angle of 30° to the horizontal. The acceleration during the launching stroke is 4g.

Length of launch stroke $= \dfrac{69 \times 69}{2 \times 4 \times 32.2} = 18.5$ feet

Engine thrust along launch stroke $= 19200$ Lb x Cos 30°

$= 16628$ Lb

Thrust required for 4g acceleration $= 4 \times 24000$ Lb

$\therefore$ Thrust required from launcher $= 4\text{x}24000-16628 = \underline{79372 \text{ Lbs}}$

If the driving bag is 5 feet diameter then its cross-section area is

$\pi \times 2.5^2 = 19.643 \text{ ft}^2 = 2828 \text{ ins}^2$

$\therefore$ Bag pressure necessary to generate the required thrust $= \dfrac{79372}{2828}$

$= 28 \text{ Lb/in}^2$

At full stroke the bag is subjected to a bending moment tending to buckle it. This bending moment is the unsupported weight of the aircraft x horizontal distance from the aircraft CG to the end of the bag support at the point where it passes through the deck.

Unsupported weight of aircraft = 24000 Lbs - vertical component of engine thrust

$= 24000-19200 \text{ Sin } 60^{o}$

$= 7372$ Lbs

$\therefore$ Moment causing buckling = 7372 x 19.5

= 143754 Lb ft

The moments opposing buckling are:

(i) Bag pressure x area x $\frac{1}{2}$ diameter

(ii) Thrust of bag x distance of line of action of thrust from aircraft CG. (Assuming thrustline passes under the CG.)

If the thrustline passes 9 inches below the CG then the moment opposing buckling is 79372x0.75 = 59529 Lb ft

$\therefore$ Remaining buckling moment = 243754-59529

= 84225 Lb ft

$\therefore$ Bag pressure required to oppose remaining buckling moment

$$= \frac{84225}{19.643 \times 2.5 \times 144}$$

$$= 11.91 \text{ Lb/in}^2$$

If buckling commences when the longitudinal stress at any point in the bag falls to zero then this pressure must be added to that required for thrust.

$\therefore$ Total bag pressure = 28+11.91, say <u>40 Lb/in^2</u>

The bag is a composite structure with woven nylon fibres bearing hoop stress and steel wires bearing longitudinal stresses.

Maximum permissible stresses are assumed to be:

Nylon 10000 Lb/in^2

Steel 24000 Lb/in^2

Hoop stress $= \frac{PA}{2t}$ for unit length

where,

P = internal pressure

A = projected area (diameter in this case)

t = thickness of material

For calculation both nylon and wire are treated as solid, not woven, materials.

$$\therefore t = \frac{PA}{2x10000} = \frac{40x60}{2x10000} = 0.12 \text{ inches}$$

Total moving length of bag = 18.5+10+5 = 33.5 feet

Weight of nylon = length of bag x circumference x thickness x specific weight of nylon

$= 33.5 \times 12 \times \pi \times 5 \times 12 \times 0.12 \times 0.03$

$= 273$ Lbs

Weight of proofing material is assumed to be 200 Lbs.

Weight of thrust transmitting structure is assumed to be 300 Lbs.

If the bag is brought to rest in 5 feet then deceleration

$$= \frac{69x69}{2x5x32.2} = 14.78g$$

The longitudinal stresses on the steel wire are:

(i) Stress due to unopposed bag pressure at end of launch.

(ii) Stress due to deceleration of bag and thrust transmitting structure.

Force applied by bag pressure

$= 40x\pi x2.5^2x144$

$= 113143$ Lbs

Force applied by deceleration of bag (1st approximation)

$= 14.78\ (273+200+300) = 11425$ Lbs

$\therefore$ Force causing longitudinal stress

$= 113143+11425 = 124568$ Lbs

$\therefore$ Cross section area of wire $= \dfrac{124568}{24000} = 5.19$ ins^2

$\therefore$ Weight of steel wire $= 5.19 \times 33.5 \times 12 \times 0.2816$

$= 588$ Lbs

Force applied by deceleration of bag (2nd approximation)

$= 14.78\ (588+273+200+300)$

$= 20116$ Lbs

$\therefore$ Force causing longitudinal stress

$= 113143+20116 = 133259$ Lbs

$\therefore$ Cross section area of wire $= \dfrac{133259}{24000}$

$= 5.55$ ins^2

$\therefore$ Weight of steel wire $= 5.55 \times 33.5 \times 12 \times 0.2816$

$= 628$ Lbs

$\therefore$ <u>Weight of bag</u> $= 628+273+200 =$ <u>1101 Lbs</u>

Weight of bag and thrust transmitting structure $= 1101+300 =$ <u>1401 Lbs</u>

Appendix 12

Some Suggestions for Further Research

1. The Use of Ground Effect to Assist Landing

When hovering close to the ground a jet V/STOL aircraft suffers a loss of lift, largely due to vortex-induced suction under the mainplanes and fuselage. This loss of lift can be eliminated or greatly reduced by operating the aircraft from a grating placed some distance above solid ground. This offers the prospect of exploiting ground effect and controlling it such that it can be used to centralise the aircraft over the intended landing spot, bringing it down firmly and positively when correctly positioned.

2. Anchoring the Aircraft after Landing

Immediately after landing on a small ship which may be subject to violent motion the aircraft must be firmly held pending movement to its hangar or another position. The device to achieve this could be allied with 1 above and should ideally possess the following characteristics:

a. Require no modification to the aircraft.

b. Be activated by touch-down of the aircraft and virtually instantaneous in operation.

c. Tolerant of normal landing position variations.

d. Capable of being quickly released by a single operation.

e. Impose no restriction on movement when released.

Preliminary investigation indicates that such a device is feasible and the author hopes to pursue this as opportunity occurs.

3. In Flight Refuelling from the Surface

Circumstances may be visualised in which it may be advantageous or necessary for the aircraft to take on fuel without landing, either over the sea or over land. Unlike helicopters, fixed wing jet V/STOL aircraft obtain no significant benefit in carrying capacity by hovering over a ship steaming into the natural wind. Any additional fuel will therefore have to be lifted from a stationary or relatively stationary position on the surface to the aircraft in forward flight at comparatively high speed.

The technical problems are likely to be formidable but their solution may be justified by operational benefits.

4. Use of the Flexible Launcher as a Missile Launcher

As suggested in 3.2.8 the flexible launcher may prove to be a more efficient means of launching the heavier guided weapons than booster rockets.

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