VI. Aerodynamics of the KAPPA Rocket

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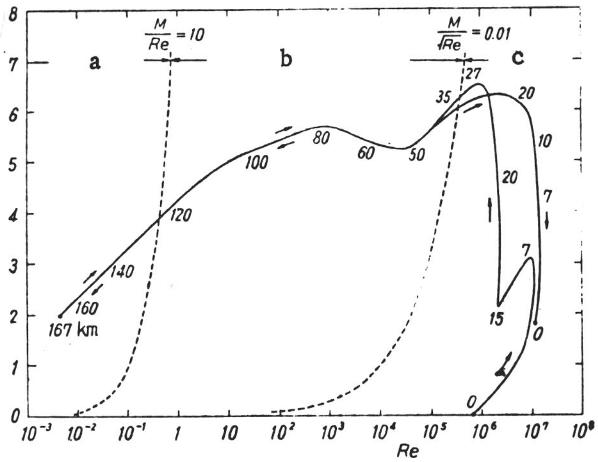
1. Introduction

A feature of the aerodynamics of sounding rockets lies in the fact that the Mach numbers in flight cover a wide area, and it is necessary to find the aerodynamic characteristics as a function of the Mach numbers within this wide range. Also, the range of Mach numbers increases sharply as the size of the rocket motor is increased. For example, the maximum Mach number of the 2 stage KAPPA-3 which flew in 1957 (that of the main or second stage rocket) was approximately 3, while the KAPPA-6 flown the next year reached 4.5 and the KAPPA-8 flown this year reached 6.5. Furthermore, the three stage 9 which is now in the planning stage is slated to reach Mach 9.5.

When investigating the aerodynamic characteristics of this kind of sounding rocket in a wind tunnel, from the consideration of air viscosity, it is desirable to bring the Reynolds number $VL\rho/\mu$ as close as possible to actual flight conditions. (V: velocity, L: characteristic dimensions of body, ρ: air density, μ: viscosity coefficient). The range of Reynolds numbers varies widely according to changes in ρ due to altitude, because the sounding rocket reaches into the upper atmosphere. Up to an altitude of 100 km, the atmospheric density drops one digit for each 16 km rise in altitude, and the Reynolds number reaches a 6 digit figure for a rocket reaching 100 km.

Figure 1 is a graph of Mach numbers M and Reynolds numbers Re taken during flight from calculated values of flight characteristics when the K-8 was fired at a 75° angle from the horizontal. In the calculations of Re, we used a diameter of 25 cm as the dimension of the main rocket. In these calculations the maximum altitude reached was 167 km, and climb and fall are shown by arrows. The numbers added to each place along the curves express the altitudes in Km. The two dotted curves indicate the scale of influence of the rarefaction of the air. The $M/\sqrt{Re} = 0.01$ curve indicates the condition occurring when the average free path of the molecules reaches 1% of the thickness of the boundary region. To the right of this hydrodynamics as a continuous medium are realized. To the left, the rarefaction effect first appears within the boundary region (slip flow) and then gradually extends to the outside. The M/Re = 10 curve expresses the fact that the average free path is 10 times the body dimension. To the left of this is the range of free molecular flow treated according to the theory of molecular motion as if the molecules independently struck the body.



a) free mole-flow
b) transition flow, slip flow
c) continuous flow

Figure 1. Mach and a Reynolds numbers during flight of K-8 (representative length 25 cm)

Although the sounding rocket flies into the area of this type of rarefied air we can say that this area is not of special importance in considering the climb capabilities of the rocket. Above 50 km, the atmospheric density is less, and the aerodynamic forces are negligible in comparison with the force of gravity. However, if we measure atmospheric pressure by making pressure measuring holes in the surface of the fuselage or measure the ion density by exposing a probe from the rocket, it becomes necessary to find the condition of the surrounding air stream accurately. We may say that the dynamics of rarefied gases is an important factor in this type of sounding technology.

In order to investigate aerodynamic problems related to sounding and the aerodynamic characteristics of the sounding rockets described above, an extremely large wind tunnel or similar facility is necessary. However, since this requires great expense at present we have to obtain the necessary design data by using more limited facilities. Below I will give an outline of the wind tunnel tests made in developing the KAPPA rocket, stability, and other problems.

2. Outline of Wind Tunnel Tests

(1) Low Velocity Wind Tunnel Tests. The flight stability of the rocket immediately after launching is the most important point to be considered in flying rockets safely. Because the velocity of the rocket

when it leaves the rails of the launching platform is about 40m/s, the aerodynamic characteristics of this condition are investigated in a low velocity wind tunnel. Every time that a new rocket is designed, we perform a 3 component force test. For this purpose we made use of the facilities of the Institute of Aeronautics. In 1955, using the 2 m wind tunnel at the Institute of Aeronautics, we conducted a 40m/s wind velocity test of the BABY rocket using a full scale model (1). Later we used 1/2 to 1/4th size scale models according to rocket dimensions and tested them at a wind velocity of 35m/s in a 3m wind tunnel (2-6).

In addition, by using a 60 cm x 60 cm small wind tunnel at the Institute, we measured the static pressure distribution on rockets and conducted preliminary tests of the characteristics of wing-body combinations (1).

(2) Supersonic Speed Wind Tunnel Tests. In preparation for rocket research, a 15 cm x 15 cm blow-off type of supersonic speed wind tunnel having a section for measurement was built at this Institute in 1954, and then a resistance wire strain meter balance was installed, and measurements were made of lift, resistance and pitching moment. The Mach number was 1.88, and the Reynolds number Re_D for the diameter was approximately 3×10^5. This wind tunnel was used for experimenting with the first KAPPA rockets, and as a result we may say that it was effective in designing the KAPPA 6. However, as the capability of the KAPPA rocket improves it becomes necessary to experiment with higher Mach numbers and in 1959, a 15 cm x 15 cm impact wind tunnel (Figure 2) was constructed (7-9). Although the time that the air stream could be maintained was only 40 millisec, by using a resistance wire strain meter type balance, it was possible to measure normal force (the force perpendicular to the center axis of the rocket) and the center of wind pressure. By substituting a wooden nozzle, the Mach number changes from 2.7 to 4.4 and Re_D from 2×10^5 to 1×10^6. In addition to systematic studies of the properties of cylindercone bodies in this wind tunnel, we conducted tests on bodies having wings or cone-shaped tail sections.

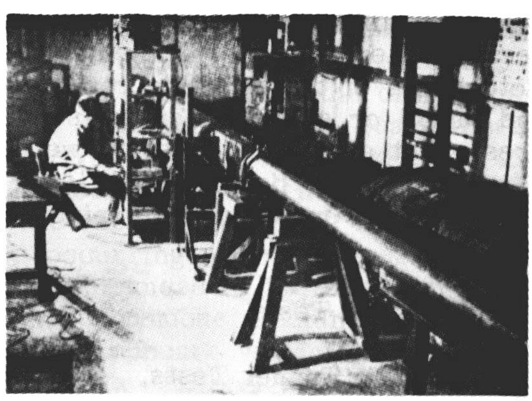


Figure 2. Wind Tunnel at the Institute.

(3) Low Density Wind Tunnel. A low density wind tunnel was constructed in 1959 for the purpose of studying the dynamics of rarefied gases when a rocket reaches the upper atmosphere (Figure 3). This wind tunnel is a closed cycle type which, after first exhausting all the air in the tunnel with a Keany [phonetic] pump having an exhaust wind velocity of 6,500 1/min circulates an airstream by the use of a Root mechanical booster (stage 1, two 5,000m^3/h, stage 2, one 2,500m^3/h). The diameter of the measuring section (the nozzle exit section) is 5 cm. While we are presently in the process of determining the wind tunnel characteristics, we are hopeful that we will be able to create an air stream of up to Mach 3 with a static pressure of 100 to 10μHg (corresponding to an altitude of 65 to 80 km). At present we are considering the use of the tunnel for various problems relating to observing the upper atmosphere -- testing different types of probes and measuring atmospheric pressures by rocket. Also, the measuring chamber of this wind tunnel has a diameter of 1.1 m and a length of 1.3 m. Thus it is used as a vacuum container, and is being considered for use in vacuum testing the payload instruments (up to 1μHg).

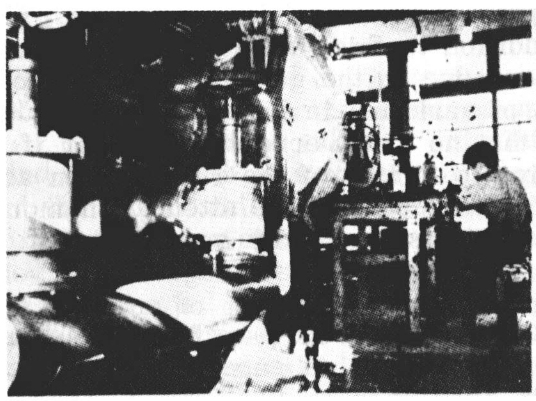


Figure 3. Low density wind tunnel at the Institute.

We have described in general outline the above equipment used in conducting aerodynamic tests. Since the majority of individual test results have already been published, I will not treat them here, but instead I would prefer to consider the results of 2 or 3 problems related to the design of the KAPPA rocket.

3. Flight Stability

Like many sounding rockets, the KAPPA rocket is unguided and is stabilized by tail fins. In Figure 4, if the rocket inclines only at angle α

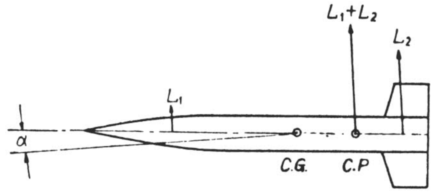


Figure 4. Forces acting on and inclined rocket.

in respect to its forward direction for some reason, the lifts operate on the rocket fuselage and the tail in the manner shown in the illustration. If the point of action of the forces combining these (center of pressure C.P.) is behind the rocket's center of gravity (C.G.), the rocket becomes stable, since the moment which tends to reduce this inclination is in effect. The rate of variation $dC_M/d\alpha$ of the angle of attack of the moment function C_M about the center of gravity expresses the degree of static stability. (C_M is the division of the movement by the product of the dynamic pressure, fuselage section area and overall length, with the nose-down moment considered as positive.)

Actually in addition to this, the attenuation moment which is due to aerodynamic forces and due to the gas jet emitting from the nozzle is proportional to the rate of variation in time of the attitude angle of the rocket. We have discovered that no great errors will occur if we only consider the static stability relative to motion of the rocket when accelerating due to burning of fuel, to the exclusion of the attenuation moment. (10)

As long as the tail is especially small, the lift on the tail is one digit greater than that acting on the fuselage at low velocities. Therefore, C.P. is located near the tail fin (80-90% of entire length from the tip). Moreover, since C.G. is around 60%, sufficient restoring moment is obtained and the rocket is stable. At supersonic speeds, however, since the wing lift decreases in proportion to $1/\sqrt{M^2 - 1}$, C.G. moves forward as M increases. Since this amount of shift between M = 0 to 6 reaches 20% of the overall length, it is considered that stability becomes insufficient due to the position of C.G., or C.P. moves further forward then C.G. and instability occurs. The normal force function is written C_N (the force perpendicular to the fuselage axis divided by the product of the dynamic pressure and the body section area) and $dC_N/d\alpha$ in the vicinity of angle of attack $\alpha = 0$, is simply expressed as $C_{N\alpha}$. α is measured in radians. The distances from the tip of the rocket to C.P., and C.G. are expressed as X_{cp} and X_{cg}, and if overall length is L, the product of $C_{N\alpha}$ and $(X_{cp} - X_{cg})/L$ expresses $C_{M\alpha}$. (Considering that $C_{M\alpha}$ expresses $dC_M/d\alpha$, when near $\alpha = 0$).

As has already been explained above, we are only able to measure these amounts directly in respect to several limited Mach numbers and calculations are necessary for those Mach numbers which come between those above and those extending to higher Mach numbers. In general the calculations we use are as follows. (5) (9)

(1) Subsonic Speeds. According to low velocity wind tunnel tests, $C_{N\alpha}$ for cone cylinder bodies is close to a theoretical value of 2 for a cone, and the effect of the cylinder section is small. Expanding these results over the entire subsonic region, the $C_{N\alpha}$ for the body is considered as 2, and we consider the C.P. to be located at a point 2/3 the distance from the tip to the cone section. The wing is similar to a rectangle, and using the results of lift surface theory, (11) the interference effects of the wingbody are calculated by the slender body theory. (12)

(2) Supersonic Speeds. At supersonic speeds, especially at high Mach numbers, due to the fact that the flow peels off the back surface of the rectangular section, and since the normal force of this section increases, we assume that relative to a cone-cylinder body, $C_{N\alpha} = 6$ and $x_{CP}/L = 0.35$. This numerical value is based on the results of testing the body in an impact wind tunnel, and we believe it a suitable value for Mach 4 and above. Strictly speaking, we should use a suitable value for each Mach number, but here we use the above value irrespective of the Mach numbers. Since the wing effect is great at low Mach numbers, even if there is some error in the force of the body, on the whole, we can say that it causes no great errors in $C_{N\alpha}$ of C.P. We can use the results of the linear theory (11) for the wing characteristics and Morikawa's theory (13) for interference to the wingbody.

Due to the existence of the tail fin on the main rocket in a two stage rocket, the tail effect of the booster is impaired and effectiveness is calculated to be 0.7. This numerical value is calculated from the results of low velocity wind tunnel tests of several two stage rockets, and for simplicity we extend the use of this value to supersonic speeds. The curves in Figure 5 indicate the calculated values of $C_{N\alpha}$ and x_{CP}/L for the K-8 main rocket. The points for the supersonic speed tests are the values determined in an impact wind tunnel and the point M = 0 is the value determined by the 3 m wind tunnel at the Institute of Aeronautics. There is a suitable agreement between calculations and experiments and it

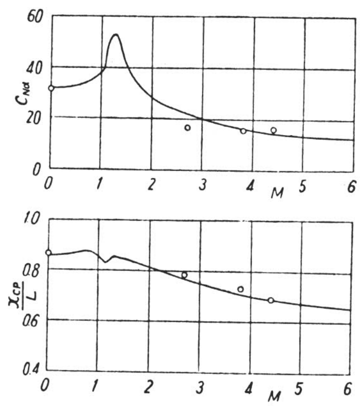


Figure 5. Characteristics of K-8 main rocket.

appears safe to estimate the characteristics in the vicinity of M = 6 to 7 by this calculation. I wish to make note of the fact that the degree of forward movement of C.P. at the supersonic speeds indicated here is the same as the results already reported for the V - 2(14) and CAJUN rockets. (15)

Figure 6 is a comparison of experiments and calculations for the 2 stage K-8 rocket. The experiment points are the same as above, but since we used a model booster tail with a chord length somewhat larger than K-8 in the 3 m wind tunnel test, we have placed directly below the black dot indicating the measured values, white circles indicating corrections made for the wing surface. Additional corrections in C.P. are small enough to be disregarded. The calculations for a two stage rocket at supersonic speeds do not match those for the test of a single body. Although the reasons are not clear, the dimensions of the model two stage rockets that we could use in our impact wind tunnel were approximately 1/60 the actual rocket, and we could not hope for very high accuracy. Consequently, we cannot but rely on the above calculations for the time being.

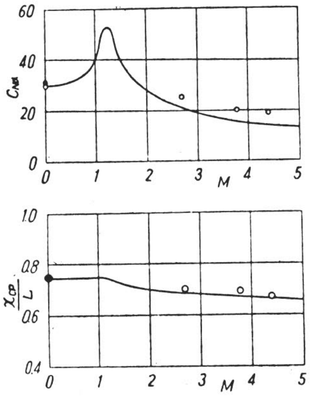


Figure 6. Characteristics of the K-8 (two stage rocket).

The next problem was how far apart the C.P. and C.G. should be in order to fly the rocket safely, and what should be a suitable magnitude for $C_M\alpha$.

We consider gusts of wind, thrust deviation and poor wing attachments as elements impairing the motion of the rocket, but their magnitudes are not known, and since the degree of allowable deviation of the rocket from the course is related to how well the rocket and the instruments inside can resist cross velocity, it is very difficult to determine the distance between C.P. and C.G. or the minimum limitation of $C_M\alpha$. In reference to this point, let us show the results of a flight test experiment using a small scale model of the K-8 in February of this year. This rocket was designed for the purpose of studying stability during the combustion of the booster, and the section corresponding to the main rocket had no propellant and was not separated. The overall length was 1.18 m, the total weight from 2.6 to 2.8 kg and the maximum velocity was approximately 560 m/s(M = 1.6) launched at a 10 to 15 degree angle to the horizontal. According to calculations for C.P., M = 0 or 75%, M = 1.6 or 70% and for $C_N\alpha$ when M = 0, 30%, when M = 1.6, 35%.

For the above we flew 2 rockets each for 3 locations of center of gravity and observed conditions during flight. The positions of center of gravity are as follows for before combustion and after combustion (after combustion in parentheses).

I: 63(57)%, II: 65(60)%, III: 68(64)%

According to this in I, the rocket flew straight with the exception of one deviation just before burn-out. Since we sometimes observed this shaking motion after combustion in the KAPPA rockets previously, we believe that when the internal pressure within the combustion chamber dropped, impact waves entered the nozzle. We believe that this is due to the fact that these waves caused the gas flow to slip into an asymmentral pattern and the direction of the exhaust deviated. If stability is sufficient, this shaking motion will occur only one time and have no later effect on the flight. The trajectory of II is slightly more irregular than I, but hardly noticeable. In contrast, the combustion in III was relatively stable at first, but a severe spiral motion was observable in the middle of the flight. We believe that this motion is probably due to the development of non-symmetry by impact waves at sonic speed. Figure 7 shows 16 mm movie shots of the flights of the above rockets.

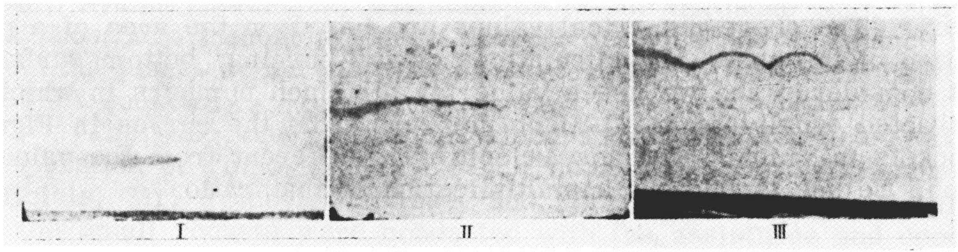


Figure 7. Flight tests of small rocket.

In view of this experiment, we think it necessary to make $C_M\alpha$ = 3+ and C.P. - C.G. distance 10% of overall length at the time of launching in order to cause a stable flight of the two-stage rocket. In the main rocket, since a high Mach number is reached and the C.P. moves forward, near burnout in this condition, it is difficult to maintain the above numerical values. However, the stability can, we believe, be maintained even when the

C.P. - C.G. distance is 7% of overall length, and $C_{M\alpha}$ is around 0.8.

4. Drag Coefficient

In calculating the climb capability and flight path of a rocket, it is necessary to know correctly the air resistance acting in the direction opposite to the forward movement of the rocket. The drag coefficient C_D is defined as the division of the air resistance by the product of the dynamic pressure, and the body section area since a rocket with sufficient stability can be considered to maintain constantly a direction during flight tangent to the trajectory, it is permissible to consider the angle of attack for drag as zero. With this in mind, we simply assign the symbol for the case when $\alpha = 0$.

Although we must know the values for C_D over a wide range of Mach numbers, it is difficult to determine drag coefficients correctly in the wind tunnels now in use, except at low velocities, as the rocket models are very small. Although we have determined the drag coefficients of several small scale models using a resisting wire strain meter scale in the supersonic speed wind tunnel at the Institute, because the models are small and there limitations to the scale, one difficulty was that we were unable to measure bottom drag.

Also, the numerical values known for C_D in conventional sounding rockets are haphazard and when we first designed the KAPPA rocket we did not know whether to look for a value for C_D for estimating the capability of the rocket. However, it later became possible to find the value of C_D from measuring acceleration and deceleration of the rocket during actual launching, since rocket curve used for calculating present rocket capabilities was constructed from reference to values in the flight testing of the K-150 and the DEACON (16) (this graph is by Hirosawa). The nose body section of the KAPPA rocket is thicker than the engine section (area is 7% larger). The above numerical values are based on the area of a particular section of the nose. In calculations during burning, bottom surface drag is not considered and we use a value for all Mach numbers in which 0.2 is subtracted equally from all Mach numbers from the curves in Figure 8. For rockets in which C_D at low velocities is different from the values on the graph, we use throughout a multiplication of that ratio.

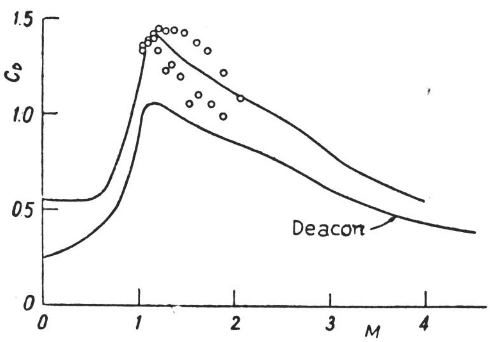


Figure 8. Drag Coefficient.

This is a very convenient method and the estimation of capability during actual flight is quite accurate.

Now to sum up the results of low velocity tests using the 3 m wind tunnel at the Institute of Aeronautics. The nose sections of the K-15, K-245, K-420, etc. are cones with a semi-vertical angle of 10°. The ratio of overall length to diameter is 16 to 22, the four tails are flat sheets, with a V-shaped leading edge (rounded) and the ratio of a pair of exposed tail surfaces to the area of a particular body section is 6 to 7. The low velocity drag coefficient (value based on the area of a particular section of the nose) obtained in the 3 m wind tunnel at the Institute of Aeronautics is 0.5 to 0.6, and the experimental value of only a wingless body is about 0.35.

In the two stage K-6 and K-8 rockets, the ratio of the diameters of the main and booster rockets is 0.6, and the ratios of the overall lengths and the diameters of the boosters are 22 and 23 respectively, but the drag coefficients based on the area of the booster section are respectively 0.61 and 0.58, numerical values which are hardly different from single rockets.

In addition to the items described above, other problems related to aerodynamics are aerodynamic heating and aerodynamic elasticity. At the present time these are being studied by Assistant Professor Mori from the standpoint of airframe construction. Since there is no wind chamber sufficient at present for studying these problems, theoretical calculations and comparative examinations are being made through flight tests. In other words, for aerodynamic heating tests, many measurements were made of the temperature of each section of the airframe during actual flight tests of the K-6, K-8 and others. Also body flutter is being studied by flying and testing small rockets.

Finally, I wish to express my sincere thanks to professors Ichiro Tani, Ryuma Kawamura, assistant professor Hirosuke Sato, and assistant Shosaburo Inouchi for their profitable advice and assistance in using the wind tunnel at the Institute of Aeronautics for the many tests of the KAPPA rockets.

The wind tunnel tests described above were conducted with the cooperation of Tomo Mitsuishi, and Tatsunari Nagai, both engineering officials, Mr. Tomoyuki Matsuo, and Mr. Michio Takei. The flight tests of the small rockets were conducted with the assistance and cooperation of technical assistant Iwao Yoshiyama, Akio Hirosawa, and all the personnel of the Itogawa Laboratory. To them many thanks are due.

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