Flight Tests for N51VS

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N51VS

N51VS is a Stewart 51 experimental amateur-built aircraft modeled after the North American P51D Mustang developed during world war II. The aircraft utilizes a 540 cubic inch V-8 engine. Dynamometer tests show the engine develops 495 hp at a maximum of 4750 rpm at sea level and standard conditions. The dynamometer data was taken with the engine in the same configuration as installed in the aircraft, except no with no load on the alternator or propeller governor and with standard exhaust headers installed. The in-flight power would be somewhat less, principally because to the substitution of short exhaust stacks for the headers results in reduced volumetric efficiency.

The aircraft itself has an empty weight of 2728 lbs and a center of gravity at a fuselage station of 82.85 inches, e.g. 82.85" aft of the tip of the spinner. This corresponds to 9.2% of mean aerodynamic cord. In most of the tests discussed here the gross weight was 3393 lbs and the CG was at 88.46", or 19.2% MAC. A few tests were flown under different weight and balance conditions, and where this applies the appropriate weight and balance data are noted in the text.

Airspeed calibration

The airspeed read off the airspeed indicator is indicated airspeed. It is a function only of the flight static and dynamic pressures. At seas level and standard atmospheric conditions this should be equal to the true airspeed, however it's not uncommon to have installation dependent errors. These are typically caused by the pitot probe being located in an region of perturbed flow or misalignment of the pitot axis tube with respect to the relative wind. When corrected for installation errors, the indicated airspeed becomes the calibrated airspeed. When corrected for non-standard atmospheric effects (pressure, temperature and relative humidity), the calibrated airspeed becomes the true airspeed. When further corrected for the wind speed, true airspeed becomes the groundspeed. Airspeed calibration is the process of characterizing installation error and determining the necessary correction.

In these tests airspeed calibration was done by comparing the GPS measured ground speed on the four cardinal headings (N,S,E,W) with the indicated airspeed. The indicated airspeed was then converted to true airspeed at the flight condition assuming no installation error. This was repeated for several airspeeds to build up a table of true airspeed (assuming no installation error) vs measured ground speed on four headings. Since there are only 3 unknowns in this system (the actual true airspeed (including installation error) and the two components of the wind velocity), we can recover the unknowns using any three of the measured groundspeeds. The last groundspeed makes the system over specified, and is used to check the data for self consistency. This method has the advantage that it is easier to fly a constant heading than a constant course or ground track.

The necessary equations are simple trigonometric relations based on wind triangles. In the following v_i represents the measured groundspeed on one of the cardinal headings and v_t is the true airspeed (accounting for installation error). w_i and w_{i+1} are the windspeed components in the i and $i+1$ directions respectively. The headings must be at right angles; for example, if i refers to north, then $i+1$ refers to east and $i+2$ refers to south.

$$(v_t - w_i)^2 + w_{i+1}^2 = v_i^2 \tag{1}$$

$$(v_t - w_{i+1})^2 + w_i^2 = v_{i+1}^2 \tag{2}$$

$$(v_t + w_i)^2 + wi + 1^2 = v_{i+2}^2 \tag{3}$$

Adding equations 1 and 3 gives

$$2v_t^2 + 2w_i^2 + 2w_{i+1}2 = v_i^2 + v_{i+2}^2 \equiv \alpha \tag{4}$$

Subtracting equations (2-3) and (1-3) respectively give

$$-2v_t(w_i + w_{i+1}) = v_{i+1}^2 - v_{i+2}^2 \equiv \beta \tag{5}$$

$$-4v_t w_i = v_2^2 - v_{i+1}^2 \equiv \gamma \tag{6}$$

$alpha$, β and γ can be calculated directly from the measured groundspeeds. Equations 5 and 6 give

$$w_i = -\frac{\gamma}{4v_t} \tag{7}$$

$$w_{i+1} = \frac{\gamma - 2\beta}{4v_t} \tag{8}$$

then equation 4 yields a quadratic equation in v_t^2

$$v_t^4 - \frac{\alpha}{2}v_t^2 + \frac{2\beta^2 - 2\beta\gamma + \gamma^2}{8} = 0 \tag{9}$$

At a given indicated airspeed, a good dataset will give the same value for true airspeed and wind components for each permutation of the groundspeeds. Assuming all the tests are conducted at the same altitude and in the same general area, the computed wind components for a given flight should be approximately constant. The calibrated airspeed is derived from the actual true airspeed determined as above by correcting this back to the flight conditions.

Flights conducted over a couple of days showed the airspeed indicator in N51VS to read uniformly 5.6% low over the range 100-160 K IAS. These tests were done on a hot day at an altitude of about 10000 MSL, well above the prevailing terrain. The results are displayed in Table 1 and Figure 1. All speeds are in nautical miles per hour (knots).

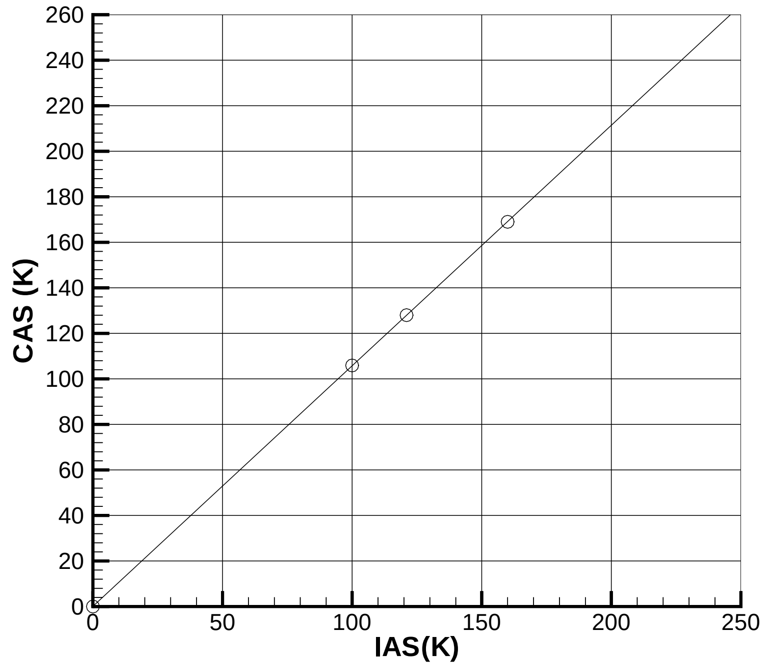

Figure 1. Calibrated airspeed vs indicated airspeed, both in nautical miles per hour (knots).

IAS	IAS corrected for PA and OAT	TAS	CAS
160	192.5	204.1	169
121	146	121	128
100	117.7	124.6	105.9

Table 1. The second column is the TAS calculated from the IAS, accounting for the effects of pressure altitude and outside air temperature, but without correction for installation error. The third column is the TAS as determined from the GPS groundspeed. Column 4, the CAS, is the TAS of column 3 corrected back to the flight conditions under which the IAS in column 1 was measured.

Climb Rates

Climb rates were determined by timing constant CAS climbs at 2000 ft intervals of pressure altitude. All these tests were done in a clean configuration (gear and flaps up) at the standard weight and balance condition. The tests were conducted starting at 500 ft over the Pacific ocean. It usually took at least 1000 ft climb to stabilize the pitch attitude at a given airspeed. Data collection began 2500 ft and ended at about 10500 ft. I only made measurements for a cruise climb condition (full throttle and 3800 engine rpm). The dynamometer data at 3800 rpm shows the engine develops 430 hp (about 86% power) under standard conditions, so the data do not represent a maximum performance climb. Figure 2 shows the climb rate as a function of density altitude for CAS in the range 89.8-137.3k. The initial (low altitude) segments conducted at 89.8k were somewhat uncomfortable with a high pitch attitude, very light forces and slow control response. This is expected since the clean, power-off stall speed is 83k CAS (the power-on stall speed is likely slightly lower). For this reason I did not try any lower speeds, although as we will see that would have been of value in determining V_y.

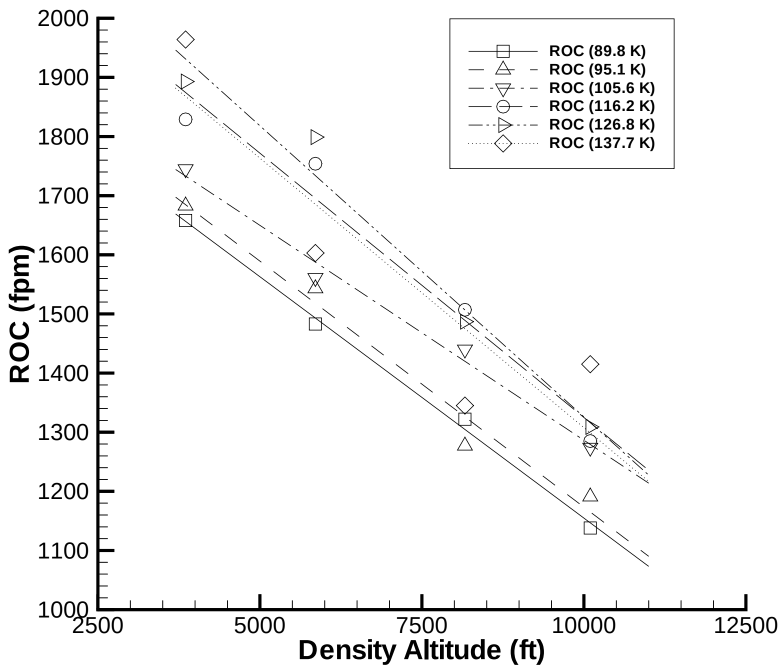

Figure 2. Rate of climb vs density altitude at constant calibrated airspeeds. Full throttle and 3800 engine RPM in a clean configuration.

Generally the climb rate for a given airspeed decreases linearly with density altitude. The lines in Figure 2 are linear least squares fits to the data. The data fit well except at the 137.3 k CAS, where there is considerable scatter.

Figure 3 shows the rate of climb (in ft/min) and angle of climb (expressed as ft altitude increase per nautical mile traveled) at constant density altitudes of 3850, 6000, 7850 and 9900 ft.

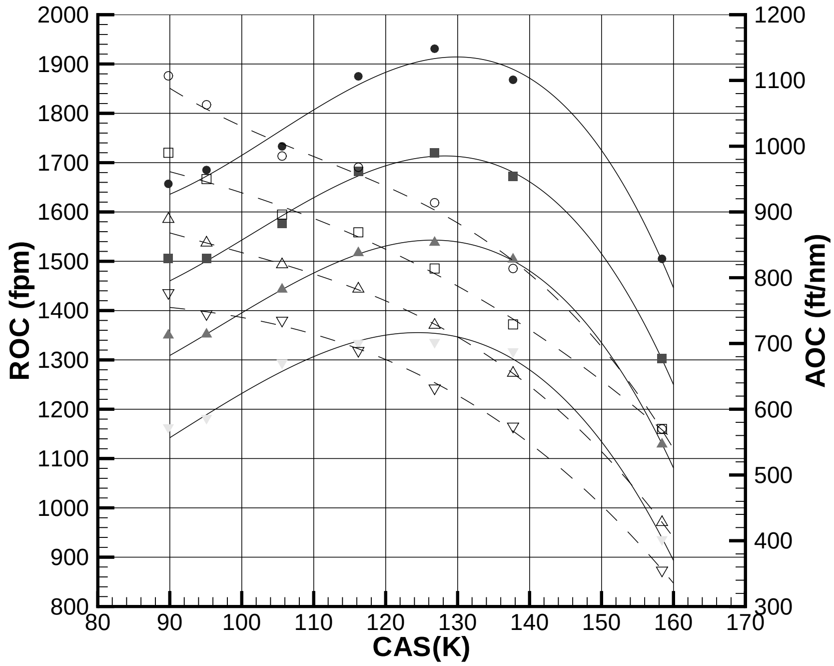

Figure 3. Rate and angle of climb vs calibrated airspeeds at constant density altitudes. Circles = 3850 ft DA, squares = 6500 ft DA, deltas = 7850, gradients = 9900 ft DA. Solid lines and filled points represent ROC. Broken lines and unfilled symbols represent AOC. All data were taken at full throttle and 3800 engine RPM.

The points in this figure for CAS $< 137.3k$ were taken from the linear curve fits of Figure 2 and hence represent somewhat smoothed data. To better define the best rate of climb airspeed I conducted an additional test at 158.4k CAS. This data was collected on a much warmer day at the actual density altitudes shown in Figure 3, hence these points involve no smoothing. The data show reasonably good agreement with 3^{rd} order polynomial fits curve fits (the lines). The peak of the solid curves give the best rate of climb and corresponding airspeed V_x. From 3800 ft to 9900 ft DA V_x decreases slightly, from about 130k to about 125k. The peak of the AOC lines defines the maximum climb angle and the corresponding airspeed V_y. As mentioned previously, I did not test low enough airspeeds to obtain a well defined peak. Judging by the slope of the AOC lines at 90k CAS, it appears that $V_y \leq 90K$ for all density altitudes examined. It appears likely that the maximum angle of climb at sea level would be obtained at or very near the stall speed. At 9900 ft DA, the slope of the AOC curve is nearly zero, so you could take V_y to be approximately 90k at this altitude. V_x and V_y must approach one another as density altitude increases, merging at the service ceiling. The trends in Figure 3 indicate that the service ceiling is far in excess of 10000 ft. I did no flying above 15000 ft due to low temperatures and the lack of a tested oxygen system, but I was still getting good climb performance at that point. The service ceiling is clearly above 18000 MSL, where flight was prohibited by the Phase I operating limitations of the airworthiness certificate.

Figure 4 shows the best rate of climb and corresponding airspeed as a function of density altitude, taken from the curve fits of Figure 3. Filled points and the solid line show the ROC. Open circles and the broken curve correspond show V_x. Extrapolating the ROC to zero DA suggests an initial climb rate from a sea level airport of about 2250 fpm under standard conditions. This is very consistent with what I observed at the Oxnard, CA airport just after the aircraft was cleaned up and the engine rpm reduced to 3800. According to the dynamometer data, this would correspond to about 86 % power.

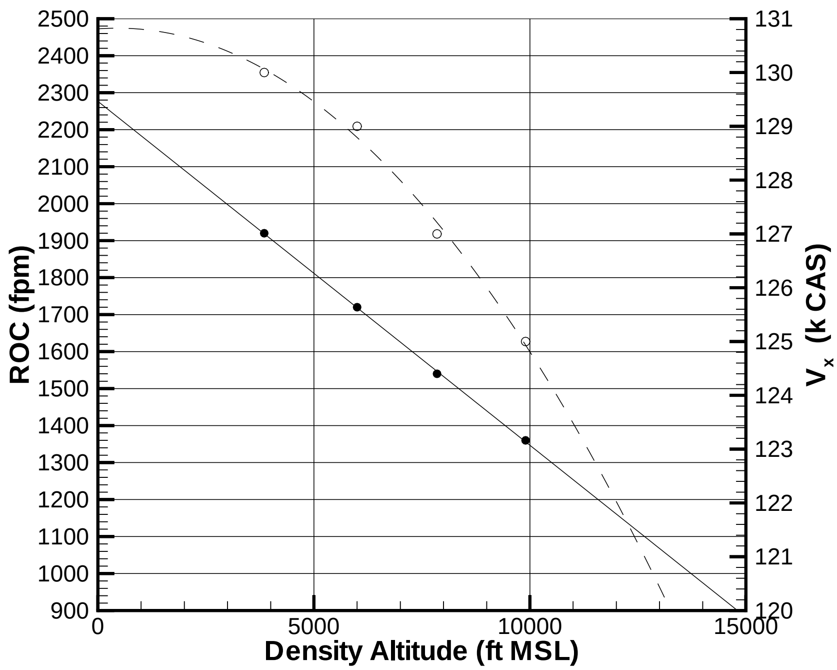

Figure 4. Best rate of climb and corresponding airspeed vs density altitude. The solid line and filled circles represent ROC. The broken line and open circles represent the best rate of climb calibrated airspeed (V_x). All data are for full throttle and 3800 engine RPM.

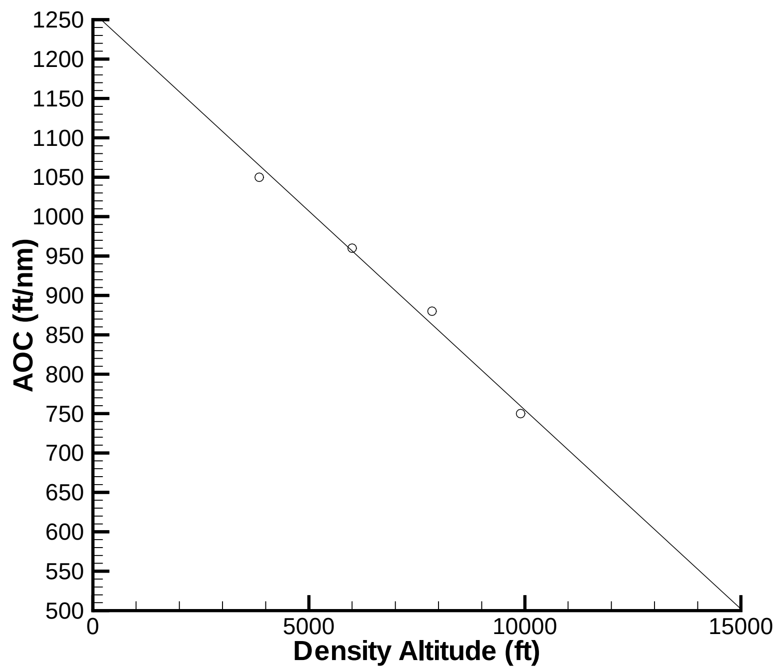

Figure 5. The maximum observed angle of climb as a function of density altitude. Data for full throttle, 3800 engine RPM and 90k CAS.

Glide data

Figure 6 shows glide ratio and sink rate as functions of CAS with aircraft clean (flaps and gear up), the propeller at the coarse pitch limit (the prop speed control at the low RPM stop) and the throttle at idle. The data was taken at a density altitude of 8400 ft, gross weight of 3325 lbs and a center of gravity at 20.9% mean aerodynamic cord (89.4 inches aft of the spinner tip).

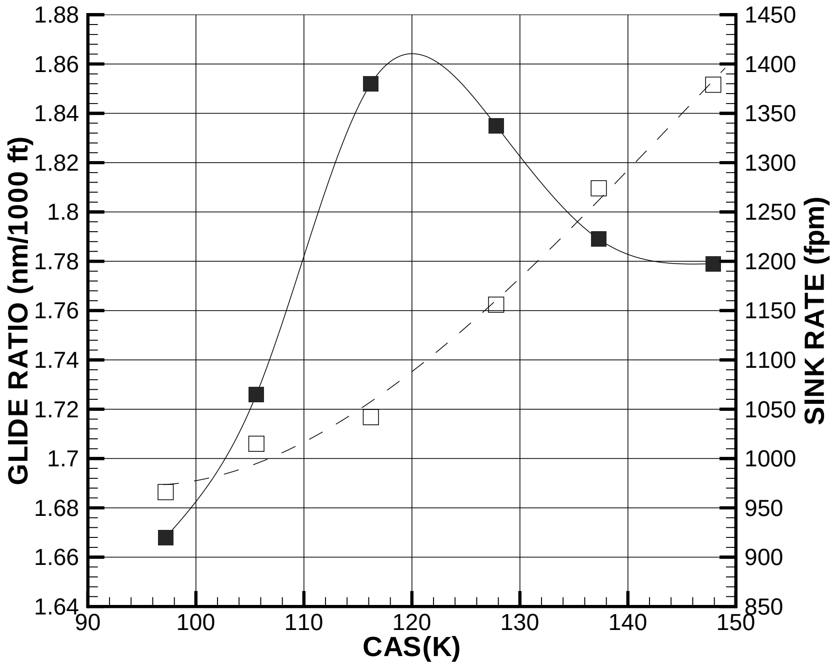

Figure 6. Glide ratio and sink rate vs CAS in a clean configuration, idle power and at maximum propeller pitch. The filled points and solid line represent the glide ratio, expressed as nautical mile over the ground (with zero wind) per thousand feet of altitude loss. The open points and broken line represent the sink rate in ft/min.

The best glide speed under these conditions is 120k CAS, which yields a glide ratio of 11.3:1 or 1.86 nm glide per 1000 ft altitude loss. This is about the same glide ratio as most light aircraft. Note that the glide ratio decreases rapidly at lower airspeed, so for maximum glide distance it's much better to be a little too fast than too slow.

The minimum sink rate occurs at a lower airspeed than the minimum tested. As for the best angle of climb airspeed, it appears to be a few knots above stall, perhaps around 90k CAS.

Figure 7 displays glide ratio data with the propeller speed control set to a typical cruise condition (3800 engine rpm). Conditions were the same as in Figure 6 except for the lower rpm propeller speed setpoint, which at the test conditions probably put the propeller against the low pitch stop. In comparison to Figure 6, the data show somewhat more scatter because the descent rate was determined over 1000 ft altitude blocks. Both the glide ratio and sink rate are very sensitive to propeller pitch. In comparison with Figure 6, the maximum glide ratio decreases by nearly a factor of two and the best glide speed increases to over 150k CAS. The very high minimum sink rate (about

2300 fpm) illustrates dominant effect of propeller drag. To achieve a reasonable glide ratio, it's imperative that the propeller speed control against the low speed stop. Very likely the glide ratio of Figure 6 could be substantially improved by stopping the propeller, but I did not try that as it would require shutting down the engine in flight. Someone should try this test.

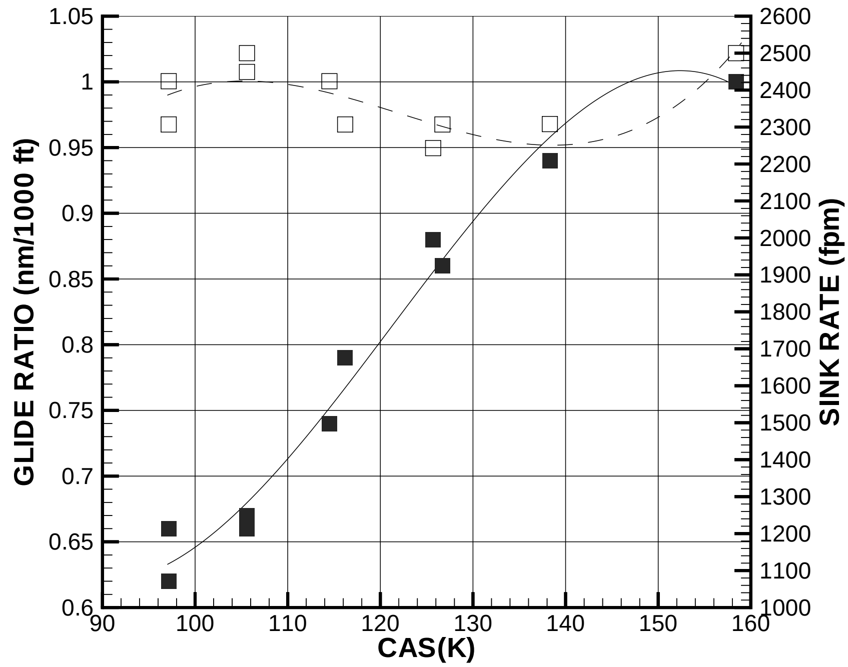

Figure 7. Glide ratio and sink rate vs calibrated airspeed in a clean configuration and with the propeller speed setpoint at 3800 engine rpm. The filled points and solid line represent the glide ratio. The open points and broken line represent the sink rate in ft/min.

I conducted some tests to estimate the minimum altitude from which a 270 degree turn to landing could be accomplished. The military refers to this position as "high key". These tests were done at Fox Field in Lancaster, CA, with a field elevation of 2335 MSL. The field density altitude during the tests was approximately 2340 ft. The aircraft weight and CG were as reported for the glide tests. The high key tests were performed by approaching the runway mid-point normal to the centerline at 4000 MSL, 110k IAS (116k CAS) and the prop pitch control set for 3800 engine rpm, lowering the gear and retarding the throttle to idle over the mid-point, then initiating a 270 degree descending left turn to a landing. No flaps were deployed. By very aggressive maneuvering involving bank angles of at least 60 degrees and sink rates well in excess of 3000 fpm, it was possible to arrive over the runway centerline at about 200 ft AGL, corresponding to an altitude loss of about 1470 ft. I maintained approximately 110k IAS on the approach. According to the relatively imprecise angle of attack instrumentation aboard the aircraft, the resulting angle of attack was within 10% of stall at the steep bank angles utilized. It's extremely difficult to make a good power-off landing from such highly unstabilized approach (I seriously damaged the aircraft trying it). The temptation is to break the descent too early, in which case you will run out of energy and redevelop a very high sink rate. The glide tests shown in Figure 7 suggest a steady-state, wings level sink rate of about 2400 ft/min with idle power and the prop set at 3800 engine rpm, and that's with the gear up. It will obviously be higher with the gear down, but perhaps not as much as might be expected since the prop is apparently responsible for most of the drag.

Several less aggressive approaches (at higher airspeeds and/or less steep bank angles) would have resulted in an off-airport landing.

Although I did not try it, I'm confident you could make a successful approach from a high key position with 1300-1400 ft altitude loss with the prop pitch set at the low rpm stop, provided you are willing to fly aggressively. I think this is worth trying, but first should be simulated at high altitude because it would be very easy to get into an accelerated stall, especially without AOA instrumentation aboard. If you have AOA instrumentation you can then try out the procedure in the pattern, where the ground will grow in the windscreen at an impressive rate. I'd advise against trying to take it all the way to the ground. In fact, I think it's risky to try it in the pattern without AOA instrumentation.

Best range and endurance

Before a lot of flight test data at cruise conditions were collected, some preliminary tests were conducted to determine airspeeds for best range and best endurance. These tests were flown at 8500 ft pressure altitude with an outside air temperature of -3C, yielding a density altitude of 8364 ft. The aircraft gross weight was 3393 lbs and the engine speed was constant at a typical high cruise condition of 3800 rpm. Figure 8 shows an example of how the airspeeds for best range and endurance were determined. Here engine power (represented by fuel flow) is plotted against TAS and CAS at constant altitude. The mixture was leaned aggressively for these tests. The lowest point on the power curve (the bottom of the bucket) represents the minimum fuel burn rate, so the corresponding airspeed (125k TAS and, in this case, about 110k CAS) results in maximum endurance. The corresponding fuel burn rate is 12.1 gph. With 68 gal usable fuel and allowing 10 gal for takeoff and climb, that gives about 4.8 hr endurance.

To the right of the bucket (the front side of the power curve) the usual procedure is to set the power at an altitude slightly above the target, then dive to the target altitude and hold a pitch attitude that results in level, steady-state flight. Here the target airspeed is approached from above. The data to the left of the bucket (e.g. on the back side of the power curve) can be considerably more difficult to obtain. Again, you're near stall here where the handling qualities are not the best. To establish level, steady-state flight under these conditions the target airspeed must be approached from below the maximum endurance airspeed. This is done by establishing a slow climb at the target airspeed and a slightly higher power setting than was used for the lowest point on the front side of the bucket, then slowly reducing power until steady, level flight is achieved. If you let the airspeed decay too far you may not be able to achieve level flight by adding power, and if the airspeed gets above the maximum endurance airspeed you will need to start over. In short, you will probably spend more than half your time taking the points on the back side.

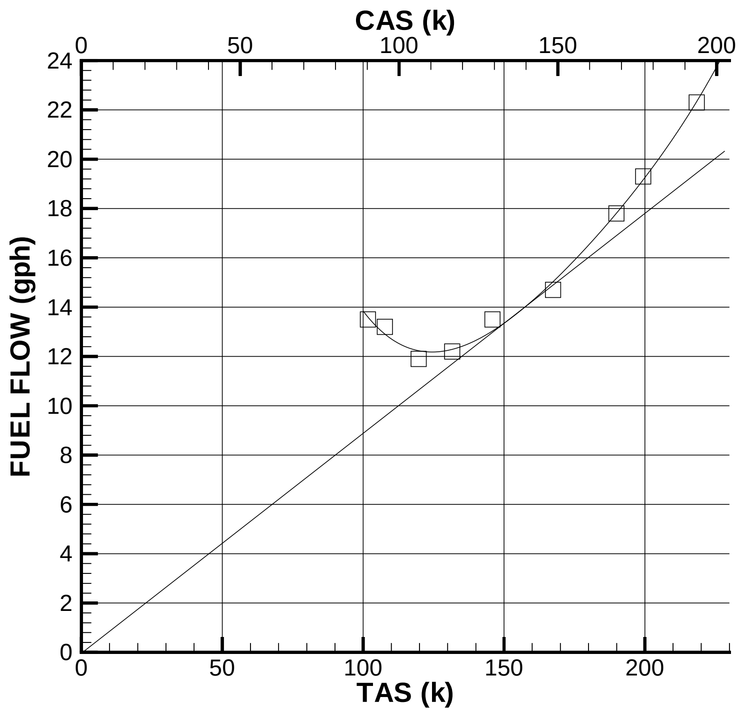

Figure 8. Plot for determination of maximum range and endurance. 3800 RPM and 8500 ft PA, OAT = -3 C. Gross weight = 3393 lbs.

The inverse of the slope of a straight line through the origin (zero power and airspeed) gives the specific range in nm/gal, so the best range condition corresponds to the line with the lowest slope. The best range airspeed is then determined by the point where the power curve is tangent to a straight line drawn from the origin. In this figure it's about 155k TAS. Flight at 155k TAS with a fuel burn of 13.5 gph would result in a range of 588 nm, assuming no wind, 30 minutes reserve and allowing 10 gal for takeoff and climb. That would be a 3.8 hr flight. I like to go faster, and typically plan maximum leg lengths of about 400 nm.

I have taken a little data like this under other conditions. Generally the best range and endurance speeds are about the same as indicated above, but of course the time aloft and range will change. Likely both range and endurance could be improved by operating at higher altitudes and lower engine rpm. Probably these speeds would actually be used only out of necessity, for example on a diversion due to weather or a long flight over an unpopulated area that has to be made without stops. Otherwise, I've found the cockpit of the S51 to be uncomfortable enough that 2 hour legs are about all I want to deal with.

Cruise performance data

Figures 9-12 show some cruise performance data at pressure altitudes of 4500, 6500, 8500 and 10500 ft. The mixture was leaned at each altitude to near peak exhaust gas temperature. Leaning by EGT is somewhat more difficult and less precise than is usual, probably because the EGT probes are located a short distance from the end of the exhaust stacks. This appears to cause considerable temporal variation in the EGT readings and makes it more difficult to determine peak EGT. From the data it appears that I leaned more aggressively at some altitudes than others. Although the EGT with respect to peak may be a little different among the tests, the same mixture setting applies to all the points in a given figure (e.g. at a given altitude). The data are generally similar to that in Figure 8, but now show a range of engine speeds and throttle settings, all on the front side of the power curve. The points on each curve range from 2800 to 4000 rpm in increments of 200. On each curve the highest point is at 4000 rpm. The right axis shows percent power, which is based on a full power fuel flow of 38.6 gph. This was measured at a density altitude of 1830 ft, full throttle and 4750 engine rpm. The corresponding airspeeds were 232.4k CAS and 239.1k TAS.

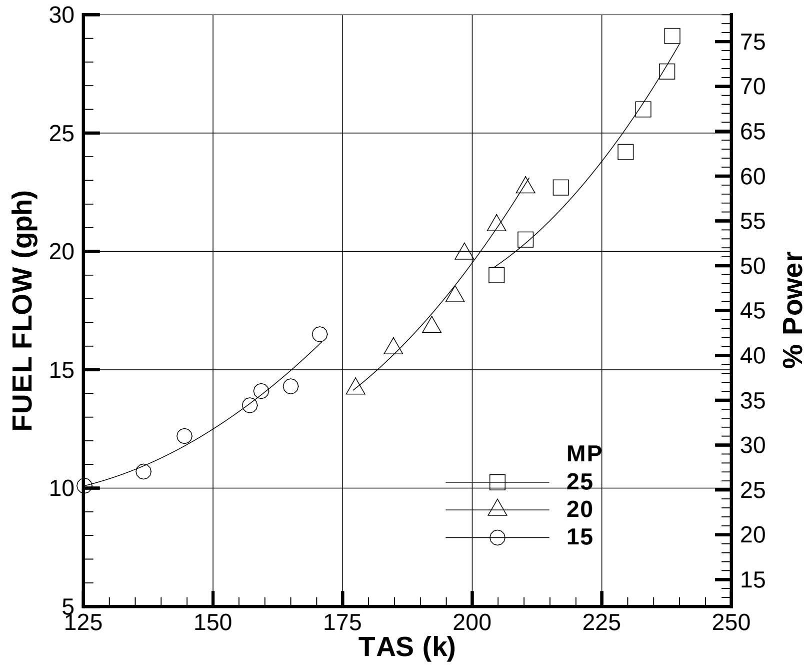

Figure 9. Power necessary to maintain airspeed in level flight at 4500 ft PA. Square = 25"MP (full throttle), Delta = 20"MP, Circle=15"MP. Points on each curve represent engine speeds from 2800 - 4000 RPM in increments of 200.

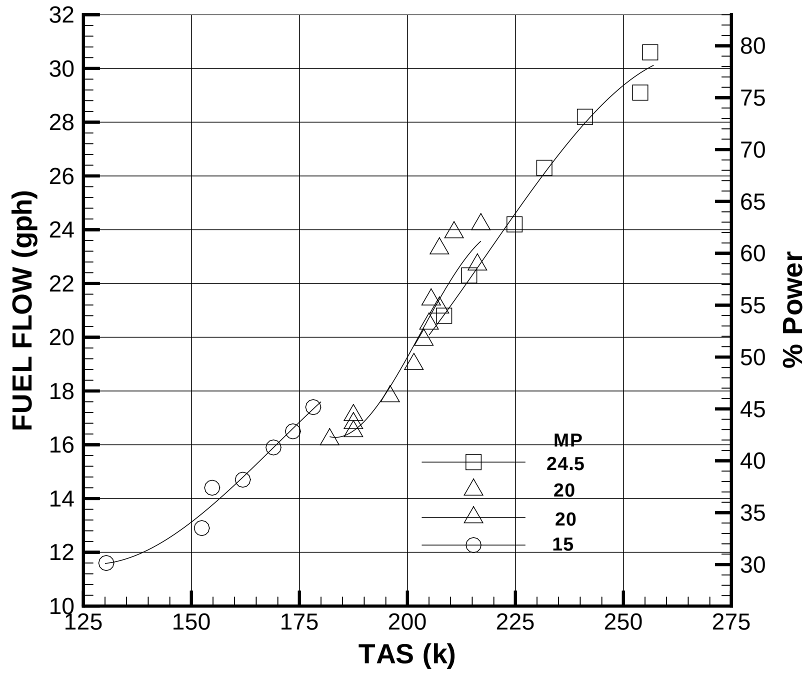

Figure 10. Power necessary to maintain airspeed in level flight at 6500 ft PA. Square = 24.5”MP (full throttle), Delta = 20”MP, Circle=15”MP. Points on each curve represent engine speeds from 2800 - 4000 RPM in increments of 200.

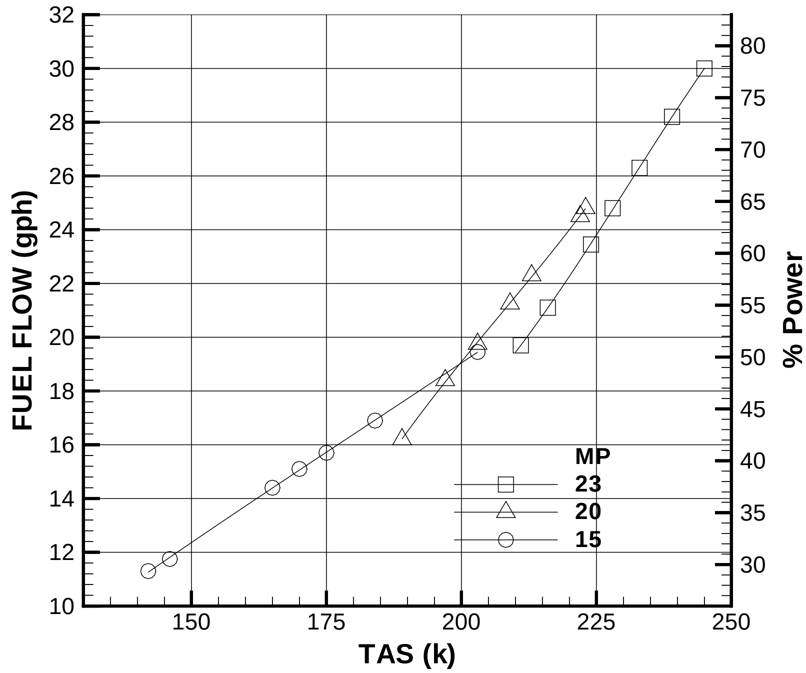

Figure 11. Power necessary to maintain airspeed in level flight at 8500 ft PA. Square = 23"MP (full throttle), Delta = 20"MP, Circle=15"MP. Points on each curve represent engine speeds from 2800 - 4000 RPM in increments of 200.

Taking Figure 10 at 8500 ft PA as an example, the highest curve represents full throttle, about 23" MP. The middle curve is for 20" MP and the lowest is for 15" MP. Density altitudes in these tests varied between 8300 and 9200 ft. If you want to use 4000 rpm and full throttle you can expect get 245 K TAS at 8500 ft. It's going to cost you 30 gph though. I typically cruise at full throttle and 3000 rpm. That gets you 215K and 21 gph at 55% power. At that speed you can cover a 400 nm leg in about 2 hrs, start to finish.

Figures 9, 10 and 12 show similar data for pressure altitudes of 4500, 6500 and 10500 ft.

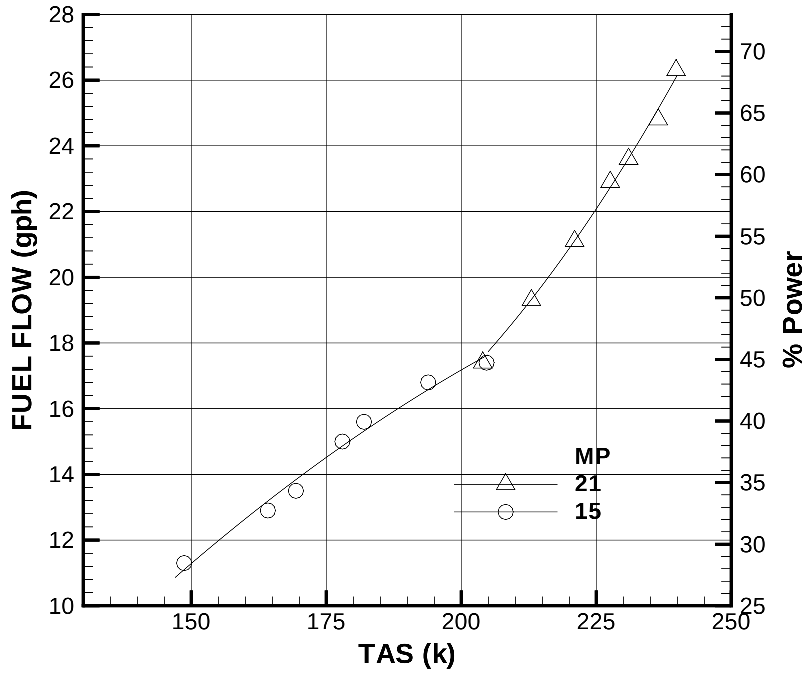

Figure 12. Power necessary to maintain airspeed in level flight at 10500 ft PA. Delta = 21"MP (full throttle), Circle = 15"MP. Points on each curve represent engine speeds from 2800 - 4000 RPM in increments of 200.